AERONAUTICAL DESIGN STANDARD

HANDBOOK

CONDITION BASED MAINTENANCE SYSTEM FOR US ARMY AIRCRAFT

FUNCTIONAL DIVISION

MICHAEL S. McCALL
Technical Director (Acting)
Aviation Engineering

SUBMITTED BY:

DAVID G. STEPHAN
Director (Acting)
Aviation Engineering

APPROVED BY:

JAMES B. LACKEY
AMCOM and PEO Aviation
Standards Executive

AMSC N/A

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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 Aircraft 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 detail 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 reference 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 is 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 component's 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 database http://www.redstone.army.mil/amrdec/rdmr-se/tdmd/StandardAero.htm is important.

5. The US Army Aviation and Missile Research, Development and Engineering Center would like to recognize the support provided by the Vertical Lift Consortium (VLC) TAJI group. This organization was instrumental in providing the draft Verification and Validation of CBM Processes Appendix K. Their help and guidance in this work is greatly appreciated.

6. 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 4484, Room 202
Attn: Ms. Gail Cruce
gail.e.cruce@us.army.mil
Redstone Arsenal, AL 35898-5000
Telephone:
Desk: 256-313-8996
Mobile: 256-975-3516

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1. SCOPE

1.1 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 Debits and Credits (modified inspection and removal criteria of components based on measured condition and actual usage) resulting from CBM implementation as well as Airworthiness Debits and Credits. The document is organized with its main body associated with general overarching guidance, and appendices governing more detailed 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 shall be technically reviewed to ensure there has been no adverse impact to baseline risk. This document provides specific technical guidance for 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 UAS.

2. REFERENCES AND 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.

MILITARY STANDARDS	
MIL-STD-1553B CHG Notice 4	Digital Time Division Command/Response Multiplex Data Bus, 15 Jan 1996.
MIL-STD-1760E	Interface Standard Aircraft/Store Electrical Interconnection System, 24 October 2007.
MIL-STD-882E	DOD Standard Practice for System Safety, 11 May 2012.

(Copies of these documents are available online at https://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.

ARMY REGULATIONS	
Army Regulation 70-62	Airworthiness Qualification of Aircraft Systems. 21 May 2007.
Army Regulation 750-43	Army Test, Measurement, and Diagnostic Equipment, 3 Dec 2006.

(Copies of these documents are available online at http://www.apd.army.mil/.)

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

AERONAUTICAL RADIO, INCORPORATED (ARINC)	
ARINC-429	Mark 33 Digital Information Transfer System (DITS), Avionics Bus Interface.

(Copies of this document are available from http://www.arinc.com/ or 6767 Old Madison Pike, Suite 300 Huntsville, AL 35806, phone 256.922.1022.)

INSTITUTE OF ELECTRICAL AND ELECTRONICS ENGINEERS (IEEE)	
IEEE 802.3	Standard for Information Technology Wireless Local Area Network
IEEE 802.11	Wireless Local Area Network
IEEE 802.15	Wireless Personal Area Networks (WPAN)

(Copies of this document are available from www.ieee.org or IEEE Service Center, 445 Hoes Lane, Piscataway, NJ 08854-1331.)

INTERNATIONAL ORGANIZATION FOR STANDARDIZATION (ISO)	
ISO 11898-1:2003	Controller Area Network (CAN)
ISO 13374-3:2012	Condition Monitoring and Diagnostics of Machines
ISO 9001:2008	Certified Organization

(Copies of these documents are available at http://www.iso.org/iso/iso_catalogue.htm International Organization for Standardization, ISO Central Secretariat, 1, ch. de la Voie-Creuse, CP 56, CH-1211 Geneva 20, Switzerland, E-mail: central@iso.org, Tel. : +41 22 749 01 11 Fax : +41 22 733 34 30)

OTHER	
MIMOSA Standard	MIMOSA Open Systems Architecture for Condition Based Maintenance, v3.2. 19 August 2011
Felker, Douglas	"PM/FM Matrix & CBM Gap Analysis in Reliability Centered Maintenance." Presented to the 2006 DoD Maintenance Symposium.
Canaday, Henry.	"Hunting for Productivity Gains." Aviation Week and Space Technology. September 10, 2004

(Copies of this document are available from http://www.mimosa.org MIMOSA, Administrative Office, 204 Marina Drive Ste 100, Tuscaloosa, AL 35406, Phone 1-949-625-8616.)

Radio Technical Commission for Aeronautics (RTCA)	
RTCA DO-178B	Software Considerations in Airborne Systems and Equipment Certification.
RTCA DO-200A	Standards for Processing Aeronautical Data

(Copies of this document are available from http://www.rtca.org/ or RTCA, Inc., 1150 18th Street, NW Suite 910, Washington, DC 20036, Tel: 202-833-9339, Fax: 202-833-9434 info@rtca.org)

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

ARMY REGULATIONS	
AR 25-2	Information Management: Information Assurance. 24 October 2007. Rapid Action Revision (RAR) Issue Date: 23 March 2009.
AR 750–1	Army Materiel Maintenance Policy. 20 Sep 2007.
AR 750–43	Army Test, Measurement, and Diagnostic Equipment. 3 November 2006.
Department of the Army Pamphlet DA PAM 738–751	Functional Users Manual for the Army Maintenance Management System—Aviation, (TAMMS-A). 15 March 1999.

(Copies of this document are available from http://www.apd.army.mil/ http://www.apd.army.mil/jw2/xmldemo/p738_751/head.asp or customer service @314-592-0910 or email us at usarmy.stlouis.106-sig-bde.mbx.dolwmddcustsrv@mail.mil)

DEPARTMENT OF DEFENSE DOCUMENTS	
DoDI 4151.22	Condition Based Maintenance Plus (CBM+) for Materiel Maintenance. Department of Defense Instruction Number 4151.22. 2 December 2007. http://www.dtic.mil/whs/directives/corres/pdf/415122p.pdf
	DOD Guidebook for CBM+. May 2008. http://www.acq.osd.mil/log/mpp/cbm+/CBM_DoD_Guidebook_May08.pdf

(Copies of this document are available from https://www.dtic.mil. or Defense Technical Information Center, 8725 John J. Kingman Road, Suite 0944, Fort Belvoir, VA 22060-6218.)

DEFENSE ACQUISITION UNIVERSITY (DAU)	
US Army CBM+ Roadmap.	13 Dec 2007.
US Army AMCOM Condition Base Maintenance (CBM) Systems Engineering Plan (SEP)	May 2008

(Copies of this document are available from https://acc.dau.mil/cbm-guidebook or Defense Acquisition University, DAU-GLTC, 9820 Belvoir Road, Ft. Belvoir, VA 22060-5565)

SOCIETY OF AUTOMOTIVE ENGINEERS (SAE) INTERNATIONAL	
SAE Standard AS5391A	Health and Usage Monitoring System Accelerometer Interface Specification. 12 Dec 2002.
SAE Standard AS5392A	Health and Usage Monitoring System, Rotational System Indexing Sensor Specification. 12 Dec 2002.
SAE Standard AS5393	Health and Usage Monitoring System, Blade Tracker Interface Specification. 12 Dec 2002.
SAE Standard AS5394	Health and Usage Monitoring System, Advanced Multipoint Interface Specification. 22 Feb 2002.
SAE Standard AS5395	Health and Usage Monitoring System, Data Interchange Specification. 23 June 2006.
SAE Aerospace Information Report AIR5113	Legal Issues Associated with the Use of Probabilistic Design Methods. 7 June 2002.
SAE JA1011	Evaluation Criteria for Reliability Centered Maintenance Processes. 26 Aug 2009.
SAE JA1012	A Guide to Reliability Centered Maintenance Standard. 22 Aug 2011.

(Copies of this document are available from http://www.sae.org/standards/ or SAE World Headquarters, 400 Commonwealth Drive, Warrendale, PA 15096-0001 USA. Phone (US) 1-877-606-7323)

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

CBM Debit: The approval of any unfavorable change, from the perspective of the maintainer, to the maintenance specified for a specific end item or component, such as an increase in inspections or reduction in Component Retirement Time established for the baseline system that is based on the incorporation of CBM as the approved maintenance approach. For example, a legacy aircraft with a 2,000 hour Component Retirement Time (CRT) for a drive system component may mandate a decreased CRT for an installed component for which CBM Health Indicator (HI) values go above specified limits and the component is removed from the monitored aircraft. CBM Debits are mandated through a Safety of Flight/Aviation Safety Action Message or AWR restriction/limitation.

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 CRT for a drive system component can establish a change to the CRT for an installed component for which CBM Condition Indicator (CI) values remain below specified limits and the unit remains installed on a monitored aircraft.) Often, CBM Credits may be authorized through an 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.

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

Condition Monitoring: The technique of monitoring equipment parameters during operating conditions to look for signal behavior anomalies and long term trends in signal behavior.

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 should be assumed to equal 0.9 (or 90%).

Credible Failure: A failure that is supported by engineering test, probabilistic risk analysis, or actual occurrences of failure.

Critical Safety Item (CSI): A part, assembly, installation equipment, launch equipment, recovery equipment, or support equipment for an aircraft or aviation weapons system that contains a characteristic any failure, malfunction, or absence of which could cause a catastrophic or critical failure resulting in the loss or serious damage to the aircraft or weapons system, an unacceptable risk of personal injury or loss of life, or an uncommanded engine shutdown that jeopardizes safety. Damage is considered serious or substantial when it would be sufficient to cause a "Class A" accident or a mishap of severity category I. The determining factor in CSIs is the consequence of failure, not the probability that the failure or consequence would occur. For the purpose of this instruction "Critical Safety Item", "Flight Safety Critical Aircraft Part", "Flight Safety Part", "Safety of Flight Item", and similar terms are synonymous. The term Critical Safety Item should be the encompassing term used throughout this handbook.

Data integrity: Data Integrity refers to the provisions taken so the data is unchanged (not missing or corrupted) from when it was initially acquired by the CBM system as reflected in RTCA DO-201A, Section 2.

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.

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 as actions ranging from down sampling (volume reduction) to filtering (smoothing).

Diagnosis: The process of analyzing parameter data associated with a suspected fault and postulating the cause of the fault.

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

Exceedance: An event in which the equipment operates outside of its specified limits.

False Negative: A fault is not indicated by the Digital Source Collector but found to exist by inspection.

False Positive: A fault is indicated by the Digital Source Collector but not found to exist by inspection.

Failure: The loss of function of a part, component, or system caused by the presence of a fault.

Fault: An undesired anomaly in an item or system.

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 System: Equipment / techniques / 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, or procedures used to measure the loads (forces or moments) experienced by an aircraft component during operational flight.

Mission Profile: A time-based description of engine operating conditions experienced in the course of a nominal mission.

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

Prognosis: The prediction of life/degradation of a component or the time before failure based on the current and accumulated parameter data.

Regimes: Aircraft load event categorized by aircraft configuration, flight environment, and operating condition type and severity.

Regime Recognition: The process of using flight data to identify historical flight regime occurrences and durations.

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

Remaining Useful Life (RUL): The actual or predicted useful life left on a component at a particular time of operation. .

Service Life: The number of hours or cycles a life-limited component may remain in service before it must be removed.

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

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

True Negative: A fault is not indicated by the Digital Source Collector nor found to exist by inspection.

True Positive: A fault is indicated by the Digital Source Collector and found to exist by inspection.

Usage Monitoring System: Equipment / techniques / procedures by which selected aspects of service [flight] history can be determined.

Usage Credit: Credit awarded to equipment maintenance practices (i.e. inspections, overhaul, on wing life limit) based on selected aspects of actual service usage history of specific monitored equipment versus design usage of general equipment population.

Usage Spectrum: Operating condition distribution used in fatigue analysis which allocates time or number of occurrences over a period of operation based on mission profile, theater, and unit.

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.

3.1 Acronyms

1P	once per revolution
a/c	aircraft
ADC	analog-to-digital converter
ADS	Aeronautical Design Standard
AED	Aviation Engineering Directorate
AES	Atomic Emission Spectroscopy
AG	Advisory Generation
AMCOM	Aviation and Missile Command
AN	Acid Number
ANN	artificial neural networks
AOAP	Army Oil Analysis Program
AOB	angle of bank
APU	Auxiliary Power Unit
AR	Army Regulation
ARINC	Aeronautical Radio, Incorporated
ASMET	Accelerated Simulated Mission Endurance Tests
ASTM	American Society for Testing and Materials
ATEDS	Aviation Turbine Engine Diagnostic System
AWR	Airworthiness Release
BAMO	Battalion Aviation Maintenance Officer
BIT	built-in test
BITE	Built-In Test Equipment
BPFI	inner race ball pass frequency
BPFO	outer race ball pass frequency
BSF	ball spin frequency
CAD	Component Advanced Design
CADAT	Central Aviation Data Analysis Tool

CBM	Condition Based Maintenance
CDR	Critical Design Review
CFF	cage fault frequency
CG	center of gravity
CIs	Condition Indicators
CLOE	Common Logistics Operating Environment
COTS	Commercial off-the-shelf
CRT	Component Retirement Time
CSI	Critical Safety Item
CVS	Comma Separated Values
DA	Data Acquisition
DA1	Data Algorithm 1
DAD	Detection Algorithm Development
DAL	design assurance level
DA PAM	Department of the Army Pamphlet
DAU	Defense Acquisition University
DBA	database administration
DBCC	database consistency check
DCS	Deputy Chief of Staff
DM	Data Manipulation
DMWR	Depot Maintenance Work Requirements
DoD	Department of Defense
DOS	Direct Operating System
DOT	Department of Transportation
DRGW	Denver, Rio Grande and Western Railway
DSC	Digital Source Collector
DUS	Design Usage Spectrum
EDM	Electronic Discharge Machine
EGT	Exhaust Gas Temperature
EHMS	Engine Health Monitoring System
EMI	Electromagnetic Interference
EMS	Engine Monitoring System
EOF	end-of-file
ERITS	Equivalent Retreating Indicated Tip Speed
ESU	Electrical Sequencing Unit
FAA	Federal Aviation Administration
FADEC	Full Authority Digital Engine Control
FCC	Failure Condition Categorization
FDA	Filter Debris Analysis
FHA	Functional Hazard Assessment

FLM	Fatigue Life Management
FLS	Flight Load Survey
FMECA	Failure Modes Effects Criticality Analysis
FN	False Negative
FOD	foreign object damage
FP	False Positive
FPG	Flat Pitch Ground
FTIR	Fourier Transform Infrared
GAG	Ground Air Ground
GW	gross weight
HA	Health assessment
HCF	High Cycle Fatigue
HDBK	Handbook
HIs	Health Indicators
Hr	Hour
HUMS	Health and Usage Monitoring System
IAW	in accordance with
ICP	Inductively Coupled Plasma
IEEE	Institute of Electrical and Electronics Engineers
IETM	Interactive Electronic Technical Manuals
IGB	Intermediate Gearbox
I/P	Input
IR	Infrared
ISO	International Organization for Standardization
I/TDA	Inspection/Tear Down Analysis
ITT	Interturbine Temperature
IVHMS	Integrated Vehicle Health Management System
JSSG	Joint Service Specification Guide
KFT	Karl Fischer Titration
KOH	potassium hydroxide
LAR	Logistics Assistance Representative
LCF	Low Cycle Fatigue
LE	Logistics Engineer
LIMSS GAS/MAST	Longbow Integrated Maintenance Support System Ground Analysis Software
LIW	Logistics Information Warehouse
LME	Loads Monitoring and Estimation
LNF	LaserNet Fines
LRM	Line replaceable module
LRU	Line replaceable unit
LUIs	life usage indicators

MAC	Message Authentication Code
MDR	Maintenance Data Recorder
MEC	Maintenance Engineering Call
MES	main engine start
MIC0	Message Integrity Code (
MIL-STD	Military Standard
MIMOSA	Machinery Information Management Open Standards Alliance
MPA	Module Performance Analysis
MSPU	Modern Signal Processing Unit
MTBF	Mean Time Between Failure
MTTR	mean time to repair
NDE	Non Destructive Equipment
NDI	Non-Destructive Inspection
NGB	nose gearbox
OASIS	Oil Analysis Standard Interservice System
OAT	Outside Air Temperature
ODM	oil debris monitoring
O/P	Output
OUS	Operating Usage Spectra
PA	Prognostics assessment
P_D	Probability of Detection
PDR	Preliminary Design Review
PEO	Program Executive Officer
P_{FN}	Probability of False Negative
P_{FP}	Probability of False Positive
PHM	Prognostics and Health Management
POD	Probability of Detection
PODF	Probability of Detecting a Fault
PPM	Part Per Million
PSAC	Plan for Software Aspects of Certification
PSSA	Preliminary System Safety Assessment
PTIT	Power Turbine Inlet Temperature
RCM	Reliability Centered Maintenance
RDBMS	Relational Database Management System
RE	rolling elements
RIMFIRE	Reliability Improvement through Failure ID and Reporting
RFP	Request for Proposal
RFS	rotrode filter spectroscopy
ROA	Reduced Order Algorithm
ROC	Receiver Operating Characteristic
ROM	reduced order models

RUL	Remaining Useful Life
SAE	Society of Automotive Engineers International
SARSS	Standard Army Retail Supply System
SD	State Detection
SEM	scanning electron microscope
SFT	Seeded Fault Testing
SHM	Structural Health Monitoring
SIRB	Software Internal Review Board
S/N	empirical stress/cycle
SOW	Statement of Work
SPC	Statistical process control
SQL	Standardized Query Language
STA	Synchronous Time Averaging
STAMIS	STandard Army Management Information System
SUMS	structural usage monitoring system
TAJI	Technical Area of Joint Interests
TBO	Time Between Overhauls
TCP/IP	Transmission Control Protocol/Internet Protocol
TDA	Tear Down Analysis
TEI	Total electrical impedance
TGT	Turbine Gas Temperature
TM	Technical Manual
TN	True Negative
TOS	Top of Scatter
TP	True Positive
TRR	Test Readiness Review
T/SF	Time of Inspection/Service Life
UAS	Unmanned Aircraft System
ULLS-A	Unit Level Logistics System-Aviation
V&V	Verification and Validation
V_h	maximum level flight airspeed
V_{ne}	“never exceed” airspeed
VLC	Vertical Lift Consortium
VMS	Vibration Monitoring System
WUC	work unit code
XRF	X-Ray Fluorescence

3.2 Condition-based maintenance probability parameter definitions

Complementary Probability: The probability of an event that can be expressed as a binomial probability if the event's outcomes can be broken down into two probabilities of events *A* and *B*. When *A* and *B* are complementary, the sum of their probabilities of occurrence is one (i.e. $A + B = 1$).

+ : CBM system provides an alarm indicating an unhealthy component and maintenance is required.

- : CBM system does not provide an alarm, indicating a healthy component and no maintenance is required or is optional.

F: Unhealthy component and maintenance is required.

H: Healthy component and maintenance is not required or is optional.

PF: Proportion of population that are unhealthy components = 1-PH. PF is complementary to PH. Therefore, PF + PH = 1 and PF = 1-PH.

P(F): Probability of an unhealthy component.

P(+): Probability that a Health and Usage Monitoring System (HUMS) indicates an alarm = [P(+|F)*P(F)] + [P(+|H)*P(H)] = [(PF)(PODF)] + [(PH)(PFP)]. P(+) is complementary to P(-). Therefore, P(+) + P(-) = 1.

P(-): Probability that a HUMS does not provide an alarm = [P(-|F)*P(F)] + [P(-|H)*P(H)] = [(PF)(PFN)] + [(PH)(PODH)]. P(+) is complementary to P(-). Therefore, P(+) + P(-) = 1.

PFA or POFA: Probability of a false alarm, also known as probability of false positive (PFP).

PFN: Probability of a false negative.

PFP: Probability of a false positive also known as probability of false alarm (PFA).

PH: Proportion of population that are healthy components = 1-PF. PH is complementary to PF. Therefore, PH + PF = 1 and PH = 1-PF.

P(H): Probability of a healthy component.

POD: Probability of detection (also known as detection reliability of a CBM system).

P(A|B): If A and B are events, then P(A|B) is the probability of event A occurring given that event B has occurred.

P(F|+): Probability of having an unhealthy component given the system provides an alarm = $P(F|+) = \frac{[P(+|F)*P(F)]}{[P(+|F)*P(F)+P(+|H)*P(H)]} = (PF)(PODF)/[(PF)(PODF) + (PH)(PFP)]$

P(F|-): Probability of having an unhealthy component given the system does not provide an alarm =

$$P(F|-) = \frac{[P(-|F) * P(F)]}{[P(-|F) * P(F) + P(-|H) * P(H)]}$$

$$= (PF)(PFN)/[(PF)(PFN)/(PF)(PFN) + (PH)(PODH)]$$

P(H|+): Probability of having a healthy component given the system indicates an alarm

$$= P(H|+) = (PH)(PFP)/[(PH)(PFP) + (PF)(PODF]$$

P(H|-): Probability of having a healthy component given the system indicates a healthy component

$$= (PH)(PODH)/[(PH)(PODH) + (PF)(PFN)]$$

P(+|F): Probability of a system indicating an alarm given there is an unhealthy component requiring maintenance. P(+|F) is also called "Probability of Detection" (POD) or "Probability of Detecting a Fault" (PODF) which is specified in this document to equal or exceed 90% or 0.9 for critical components. POD is complementary to "Probability of False Negative" (PFN). Therefore, POD + PFN = 1 and PODF = 1 – PFN.

P(-|F): Probability of a system not indicating an alarm given there is an unhealthy component requiring maintenance. P(-|F) is also called "Probability of False Negative" (PFN) and is complementary to the Probability of Detection (POD). Therefore, PFN = 1-PODF and 1-POD. If POD is specified as 90% or greater, then PFN is less than or equal to 10% or 0.1.

P(+|H): Probability of a system indicating an alarm for a healthy component = 1-PODH. P(+|H) is also called "Probability of False Positive" (PFP) which is specified in this document to be equal to or less than 10% or 0.1 for critical components. PFP is complementary to "Probability of True Negative" (PODH). Therefore, PFP + PODH = 1 and PFP = 1 - PODH. Also, PFP = Significance level = (1 – Confidence).

P(-|H): Probability of a system not indicating an alarm for a healthy component. P(-|H) is also called "Probability of True Negative" or "Probability of Detecting Healthy" components (PODH) which is specified in this document to be equal or greater than 90% or 0.9 for critical components. PODH is complementary to "Probability of False Positive" (PFP). Therefore, PODH + PFP = 1 and PODH = 1 – PFP.

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 (Ref DOD CBM+ Guidebook, SAE JA1011, & SAE JA1012) 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]

The purpose of Condition Based Maintenance is to take maintenance action on equipment where there is evidence of need. Maintenance guidance is based on the condition or status of the equipment instead of specified calendar or time based limits while preserving the system baseline risk. The key to implementing CBM is to 'right size' CBM for the targeted platform. This is achieved by defining what is practical to implement vs. attempting to implement condition based maintenance on all possible equipment. This Design Handbook describes the elements that enable the issuance of CBM Credits, or

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

modified inspection and removal criteria of components based on measured condition and actual usage utilizing systems engineering methods.

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. 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 materiel to standards that ensure 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 Aircraft Systems.

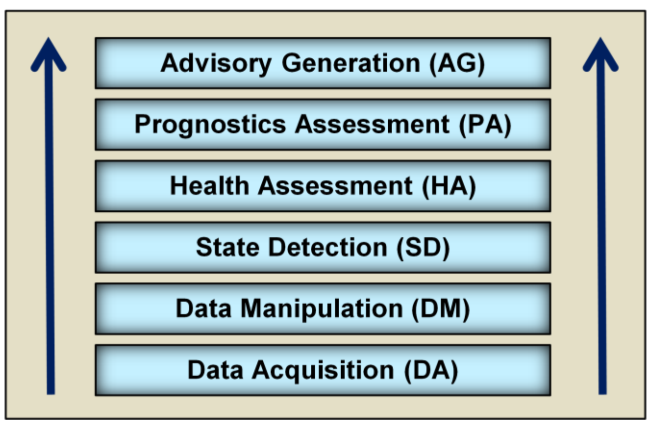

FIGURE 1. ISO-13374 Defined data processing and information flow

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

a. Embedded diagnostics/prognostics 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). See Figure 2 for Ideal sensor behavior.

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 accumulated. Within this context, detail guidance is provided where benefits can be derived.

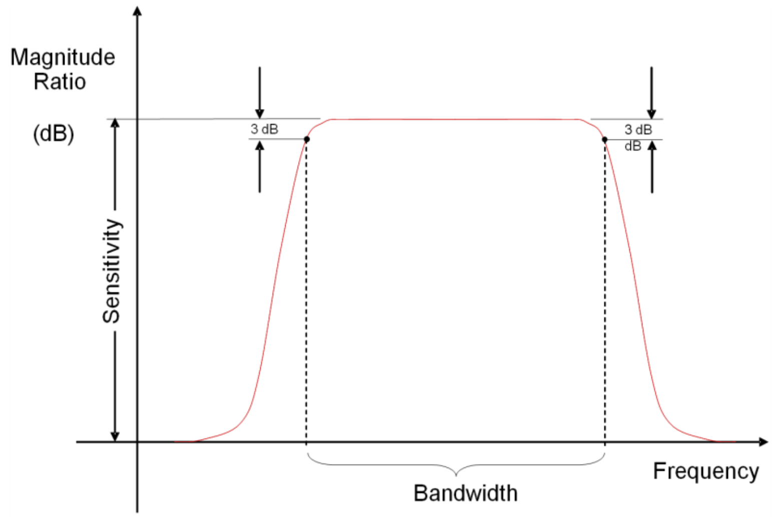

FIGURE 2. Ideal sensor behavior

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.

Health and Usage Monitoring Systems (HUMS) operation during flight is essential to gathering data for CBM system use, but is not typically flight critical or mission critical as long as the HUMS does not provide actionable input to the pilot or control the aircraft during flight. When the HUMS is not flight critical, 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 bears the same priority as mission or flight critical equipment relative to the requirement to restore its proper operation and requires the same level of software qualification as all flight critical systems.

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-1553, 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.

4.2.1 Embedded diagnostics/prognostics. HUMS has 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 records 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/prognostics”) represents a capability to provide the

[2] MIL-STD-1553B. Digital Time Division Command/Response Multiplex Data Bus. 15 January 1996.

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. There is no intent for recurring calibration of these sensors.

b. Data Acquisition: Onboard data acquisition hardware should have high reliability.

c. Sensor Selection: Sensors should be selected and/or designed in such a way that the predominate failure mode does not affect operational performance.

d. Algorithms: Fault detection algorithms are applied to the basic acquired data to provide condition and health indicators. Validation and verification of the Condition Indicators (CIs) and Health Indicators (HIs) included in the CBM system are required to establish maintenance and airworthiness credits/debits. Basic properties of the algorithms are: (1) sensitivity to the faulted condition, and (2) insensitivity to conditions other than the faulted condition. The algorithms and methodology should demonstrate the ability to account for exceedances, missing, or invalid data. Once verified and validated, there should be the presence of continuous assessment of algorithm performance. Algorithms utilized as maintenance practice enhancements (versus a maintenance practice replacement) with reliability not verified and validated need only demonstrate a level of reliability acceptable to the platform manager and maintainer (i.e. acceptable level of maintenance associated with false positives).

4.2.2 Fatigue damage monitoring. Fatigue damage is estimated through calculations which use loads on aircraft 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)), external stores, and aircraft state parameters related to maneuvering (i.e.: air speed, aircraft attitudes, power applied, and accelerations). To establish these loads, 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 retirement time adjustment, the algorithms should be validated through flight testing. Detail guidance for validation of regime recognition algorithms is contained in Appendix B.

Legacy aircraft operating without CBM capabilities typically use assumed usage, test established fatigue strength, 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 aircraft sensors, typically performed onboard an aircraft by a Digital Source Collector (DSC) enables methods that improve accuracy of the previous detection and assessment methods. The improvement is due to the use of actual service usage or measured loads rather than calculations based on assumptions made during the developmental design phase of the acquisition.

4.2.3 Regime recognition (usage detection) Accurate detection and measurement of flight regimes experienced by the aircraft 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) individual component fatigue damage assessment estimates can be based on specific aircraft flight history instead of the baseline "worst case design estimate" for the total aircraft population. To perform individual component fatigue 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 retirement intervals 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 individual component fatigue 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 individual component fatigue 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 individual component fatigue 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 Remediation of fatigue sensitive components. 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 or debits (changes to scheduled maintenance) based on usage/loads monitoring and damage accrual or CI/HI values 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-Aviation (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 use the predetermined "thresholds" or CI/HI values to allow for anticipatory supply actions, optimized maintenance planning, and enhanced safety by avoiding a precautionary landing/recovery/launch.

c. Extending/overflying the CRT based on individual component fatigue 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 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 extending/overflying a fatigue-related retirement or inspection interval can be given to groups of aircraft or parts, as long as they can be tracked. CBM credits extending/overflying a fatigue-related retirement or inspection interval cannot be applied to individual items based on individual component fatigue 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.

d. Lack of complete serialization of monitored components

5. DETAILED GUIDANCE

Detail guidance for the CBM system is grouped by the functionality shown in Figure 1to link the guidance to the overarching International Standards Organization (ISO) and Data Acquisition (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 twelve 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: Structural Health and Loads Monitoring

d. Appendix D: Minimum Guidance for Determining CIs/His

e. Appendix E: Vibration Based Diagnostics

f. Appendix F: Rotor Track and Balance

g. Appendix G: Turboshaft Engine and Auxiliary Power Unit Condition Based Maintenance (CBM)

h. Appendix H: Embedded Diagnostics/Prognostics and Health Management of Electronic Components

i. Appendix I: Sample Sizes for Maintenance Credits Using Vibratory CBM on Propulsion Systems

j. Appendix J: Seeded Fault Testing

k. Appendix K: Verification and Validation of CBM Processes

l. Appendix L: Data Integrity

m. Appendix M: Oil Condition and Debris Monitoring

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 information presentation. Technical displays and information presentation to support CBM should be developed for compatibility with software operating systems and DoD style guides. These operating systems are identified by the Logistics Information

Systems (LIS) for desktop systems and include other standards for portable maintenance aids and 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 Appendices 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 E 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 Data Collection 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 exists 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.

The level of criticality of the HUMS information recorded should determine the capabilities of the recorder to prevent data loss or degradation between downloads, as well as the requirements for scheduling maintenance or repair of the HUMS components. The storage sampling rates are also determined by the level of criticality.

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 National 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) National Level data link; (4) Web site and (5) Other user info 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 National 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, National Level 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 detailed 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 Appendices B and E.

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 processed data to State Detection to Health Assessment, 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 State Detection, Health Assessment, Prognostics Assessment, and Advisory Generation is dependent upon the software modules used. Traceability of this software is essential for configuration management and confidence in the result. Detailed guidance for data integrity and data management is referenced in the Data Integrity Appendix (Appendix L).

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 for flight critical applications should provide a 90% probability in detection of incipient faults and also have no more than a 10% false positive rate (indications of faults that are not present). Further details are found in Appendices D and I. For non critical applications, the probability of detection and false positive rate may vary significantly lower than 90% POD and higher than 10% PFA depending on what additional maintenance associated with false positives is acceptable to the maintainer and platform manager.

c. Fault Validation: Detection algorithms should be validated to ensure that they are capable of detecting the intended faults. One common method of algorithm validation is to create (i.e. "seed") a fault in a new or overhauled component or to simply use a known faulted component, and collect data on the fault's progression to failure in controlled testing which simulates operational use. Data collected from this test are used as source data for the detection algorithm as described in Appendix J. Another common method of algorithm validation is to formally inspect components removed from service though normal operations and maintenance practices. If the component is determined to have a fault of interest that is desired to be detected, the field data can be used as source data for the detection algorithm. In either case, the algorithm's results are compared to actual component condition through direct measurement.

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 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 responds to a maintenance error rather than the presence of a fault. 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. CBM can also be used to control the conditions that cause the vibrations, which prevents the failures from occurring.

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

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 detailed guidance regarding regime recognition, refer to Appendix B.

5.6 Health assessment (HA). 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: green (nominal – no action required), yellow (elevated advisory – watch/prepare for maintenance), orange (caution/remaining life limited - schedule and perform maintenance when optimal for operations), and red (warning/increased risk - ground aircraft/maintenance 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.

HIs should integrate with the existing maintenance and logistics information systems. This integration extends to Interactive Electronic Technical Manuals (IETMs) where applicable.

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 flight critical applications. For non critical applications, the RUL estimate accuracy may vary significantly lower than 90% depending on what additional maintenance and costs associated with early removals are acceptable to the maintainer and platform manager. 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 10% for critical applications at the time of component removal. Again, for non critical applications, the false positive rate may vary significantly higher than 10% depending on what additional maintenance and costs associated with false positives are acceptable to the maintainer and platform manager.

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 sustain system operation

The interval between download of data and health assessment is affected by operational use and tempo or conditions noted by the flight crew. Download intervals should consider the intended use of the CI/HI implemented by the system. If the goal of the system is to enhance maintenance, download intervals should be set by the Platform Management Office. If the intent of the system is to replace current maintenance practices, the download interval should be sufficient to diagnose whether the system is operating properly, to avoid loss of data, and to identify damage prior to failure in any case.

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 and alternatives for modifying maintenance on legacy aircraft. A robust and effective CBM system can provide a basis for maintenance and airworthiness credits and debits that modify legacy maintenance practices and intervals. As part of the continuous analysis of CBM data provided by the fielded systems and or seeded fault testing, CBM applications to scheduled maintenance intervals for servicing and inspection can enhance current maintenance practices to increase aircraft availability and optimize safe operations and maintenance cost. Similarly, validated CBM data can be used to modify the Time Between Overhauls (TBO) for affected components. Finally, validated 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. Involved in each of these approaches are alternatives to CBM which should be considered to realize the most beneficial gain in maintenance and airworthiness for the using service. The following subsections discuss each of the aforementioned CBM approaches and related alternatives to CBM. It is important to note that the number of test specimens necessary to validate each alternative approach will vary across the different methods. The details for determining the necessary sample size to refine a CI are outlined in Appendix D. Guidelines to determine the appropriate Sample Size for replacing current maintenance practices are outlined in Appendix I.

5.9.1 Enhancing current maintenance with CBM on legacy aircraft. Validated and unvalidated CI/HI algorithms may be utilized for both data gathering and aircraft maintenance diagnostics/ prognostics while legacy maintenance practices remain in place. Data gathering permits time to adjust maintenance alert levels built into the algorithms (CI/HI refinement) while not degrading any aspect of continued airworthiness associated with the legacy maintenance of the aircraft. In fact, adding sensors and associated algorithms to the aircraft legacy maintenance practices can actually increase the reliability of the overall aircraft system provided the sensor hardware reliability is not mission or flight critical and does not cause unscheduled maintenance impacting aircraft readiness. Note also, the sensors and associated algorithms may be applied to focus on specific component failure modes versus all failure modes of complex components (e.g. transmissions and engines). Since baseline risk is not degraded by the sensors, validated as well as unvalidated algorithms may be employed to enhance current maintenance practices and develop new diagnostic/prognostic maintenance.

Advantages to this approach are: relatively low initial cost approach that does not require a large number of sample specimens (see Appendix D) to demonstrate algorithm reliability for each failure mode; ability to enhance maintenance by focusing on specific failure modes versus all failure modes of complex components; relatively short timeframe involved prior to field implementation due to the reduction in test requirements; and the ability to gather data during normal aircraft operation to facilitate verification/validation efforts. Disadvantages to this approach are: number of false positives indications with unvalidated algorithms along with the associated maintenance increases; and the determination of return on investment may only be conducted after field data is collected, the tear down analyses confirm sensor indications, and the diagnostics/prognostics are matured.

5.9.2 Modifying or replacing overhaul intervals on legacy aircraft. Prior to considering modifications to or replacing legacy aircraft maintenance, it is important to understand what initial specification design, maintenance, and reliability requirements were placed on the legacy platform as well as the engineering rigor utilized to verify, validate, and establish legacy maintenance practices. Consequently, any maintenance modifications or replacement should be validated as good as, or better than, legacy maintenance practices. For US Army aircraft propulsion and drive systems this involves: User requirements for aircraft usage, maintenance, and reliability; bearing B10 analyses; bearing endurance testing; lubrication shelf life analyses; gear tooth bending fatigue life analyses; gear tooth contact fatigue life analyses; thousands of hours of endurance testing; engine component life testing/analyses; wear rate analyses to ensure component reparability; and TBO/On-Condition establishment. Therefore, any CBM system implemented to modify or replace legacy maintenance practices should undergo similar analytical and testing rigor. In the case of vibration monitoring, CBM algorithms implemented to accurately depict actual hardware condition and replace current maintenance practices should be required to be validated. CBM algorithm validation will require both faulted and unfaulted components. The statistically significant sample size (see Appendix I) for faulted and unfaulted components should take into account the required confidence and reliability guidelines within this document.

In addition to understanding legacy requirements to verify and validate legacy maintenance practices, it is important to note that TBO interval extensions are generally limited by the repair limits and calculated fatigue lives of components within a system under consideration for maintenance modification. An exception to the fatigue life limit is to employ CBM monitoring if the fatigue failure mode is detectable utilizing a validated detection system and will not result in the failure mode progressing or manifesting into a failed state within 2 data download intervals of the monitoring system.

Results of teardowns should be involved in validating the measured detection value to ensure that it is representative of the actual hardware condition. An example would be Hertzian Contact Fatigue for bearings. This type of fatigue generally results in spalling, which is usually easily detected (through chip detection or vibration monitoring) and also is usually associated with significant operational capability remaining from the onset of spalling. Again, component sample sizes for validating CBM algorithms to detect faulted and unfaulted bearings should take into account the required confidence and reliability guidelines within this document when the components are flight critical.

An alternative to TBO extensions employing CBM monitors and algorithms is to extend TBOs based on actual hardware condition from the field. This may be achieved by using a minimum of 5 detailed teardown inspections of components that reached the original TBO in the field. 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. Based on the US Army's past experience, teardown inspections on actual field hardware, involving dimensional analysis and comparison to production and depot repair limits, ensures confidence in capturing the inherent variability that may occur with actual field usage. If the parts are determined to still be within acceptable dimensional limits (for operation and repair), a corresponding wear rate may be analyzed and a basis for a new TBO limit established with final approval of the airworthiness activity. Therefore, it is possible to obtain TBO extensions on unmonitored aircraft based on field experience. US Army historical TBO extensions have been between 200 and 500 hours depending on the analytical results.

The advantages to this alternative to CBM are: there are no additional material costs incurred to purchase components for sampling; only the costs to perform the evaluation are required since teardowns must already be conducted on fielded TBO components at the depot; and part reparability with a TBO extension is relatively easy to quantify based on the current depot information. The disadvantages to this alternative are the time incurred to obtain components from the field that are at, or near, the TBO interval and the relatively small return on investment with the incremental maintenance benefit.

5.9.3 Transitioning to on-condition maintenance for legacy aircraft. Prior to transition to On-Condition for legacy aircraft components/assemblies, incremental TBO extensions discussed in 5.9.2 should be pursued to ensure that wear rates and failure modes associated with on-condition status are fully captured and understood. Guidelines for obtaining on-condition status for components on monitored systems having performed data acquisition via field faults / seeded fault tests are outlined in paragraphs 5.9.3.1 and 5.9.3.2, respectively. Achieving on-condition status via field faults could take several years, therefore, incremental TBO extensions on monitored aircraft will be instrumental in increasing the 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 seeded fault tests required to ensure each credible, critical failure mode can be detected. Credible critical failure modes are obtained through Failure Modes Effects Criticality Analysis (FMECA) and actual field data. Damage limits should be defined for specific components in order to classify specific hardware condition to CI/HI limits through the use of Reliability Improvement through Failure Identification and Reporting (RIMFIRE), Tear Down Analysis's (TDA), 2410 forms, and other available data sources. Implementation plans should be developed for each component clearly identifying goals, test requirements and schedule, initial CI/HI limits, and all work that is planned to show how the confidence and reliability levels delineated in paragraph 5.9.4 will be achieved.

The advantages of the on-condition transition approach include: providing the highest reliability (probability of detection and true negatives) since monitoring hardware and software are tested to capture all failure modes of a component; and providing the fewest false positives/negatives as a result of the validated reliability and confidence levels in the on-condition design. The disadvantages of the on-condition transition approach involve: relatively higher (if not highest) costs and lengthiest schedule required to test the sample sizes necessary for validation limitations of current condition monitoring designs which may not be capable of, or optimal for, capturing the onset of component failure modes for legacy aircraft; and the potential limited application for components with life limited parts not tracked via regime recognition monitoring.

5.9.3.1 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 also be used as a data point to help reduce the required testing. Test plans will be developed, identifying each of the credible, critical failure modes and corresponding seeded fault tests required to reliably show that each credible, critical failure mode can be detected. The seeded fault test plan should include requirements for ensuring 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. To be eligible for on-condition status using seeded fault testing for critical components, a statistically significant sample size for faulted and unfaulted components should take into account the required confidence and reliability guidelines within this document (see Appendix I). TDA's will be ongoing for components exceeding initially established CI/HI limits. Once the capability of the monitoring system has been validated, based on successful test results from the sample specimens for each credible, critical failure mode, increased TBO intervals may be modified to on condition status and approved for use by the airworthiness authority.

5.9.3.2 Field fault analysis. The guidance for achieving on condition status via the accumulation of field faults is essentially the same as those identified in paragraph 5.9.3.1. Incremental TBO extensions will play a bigger role utilizing this approach based on the assumption that fault data will take much longer to obtain if no seeded fault testing is performed. To be eligible for on-condition status using field fault analysis, a statistically significant sample size (see Appendix I) for faulted and unfaulted components should take into account the required confidence and reliability guidelines within this document for flight critical components. TDA's will be ongoing for components exceeding initially established CI/HI limits. Once the capability of the monitoring system has been validated by successful test results from the sample specimens for each credible, critical failure mode, increased TBO intervals may be modified to on condition status and approved for use by the airworthiness authority.

5.9.3.3 Alternatives to transitioning to on-condition for legacy aircraft**.** During the Reliability Centered Maintenance analysis for justifying pursuit of an On Condition CBM approach, it may become evident that other alternatives to the on condition transition approach are more feasible and should be considered. Two of these alternatives are discussed, herein.

One alternative is component redesign and requalification using current on-condition designs that could be built into the component. For example, an unmonitored, grease filled gear box, that is time limited in the field by the life of the grease, may benefit from a redesign using an oil filled gear box and incorporating a chip detector. Sometimes a simple redesign of a seal may be all that is needed to increase time on wing for a gearbox versus implementing an on-condition maintenance approach.

The advantages of a redesign and requalification alternative include: being able to design a specific form of monitoring tailored to capture all failure modes versus a limited number of failure modes (e.g. chip detector versus vibration monitor on a complex gearbox); sustaining fewer false positives (less maintenance) and false negatives (less safety issues) with the on-condition redesign; and realizing cost and schedule synergies in testing the performance of the component and monitoring device concurrently during requalification. Disadvantages of a redesign and requalification using on condition designs are associated with relatively higher costs than pursuing TBO enhancements (5.9.1) and TBO extensions (5.9.2) as well as relatively longer schedule than TBO enhancements.

A second alternative is to do nothing. Doing nothing sometimes may be the most logical alternative if a TBO is not being attained on components in the field due to reasons not appropriate for CBM resolution. No additional investment cost or schedule is needed to maintain the status quo. However, the status quo may be unacceptable to current readiness rates. Other disadvantages to doing nothing are not being able to realize any operations and sustainment savings or provide proactive maintenance.

5.9.4 Statistical considerations. There is interest in the likelihood that the monitoring system will detect a significant difference in signal when such a difference exists. To validate the target detection and confidence levels (target detection = 90%, target confidence = 90 to 95% depending on component criticality), a statistically significant sample size (see Appendix I) for faulted and unfaulted components should take into account the required confidence and reliability guidelines within this document for flight critical components.

Since a probabilistic approach is a recommended method that can be utilized to validate CBM algorithms using confidence and reliability factors, it is important to maintain a high level of quality in the probabilistic design. It should be noted the only way to successfully attack a probabilistic design or analysis is to undermine confidence in its quality. For information addressing legal implications when employing a probabilistic approach, reference SAE AIR5113 for a compilation of experience and past precedent.

If at least one of the detections in the sample size 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 is necessary to complete validation of the CIs/HIs.

A false negative occurrence for a critical component will impact safety, and should be assessed to determine the impact on future TBO extensions or On-Condition status. 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 is necessary to continue validation of the CIs/HIs.

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 detail 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. This Management Plan should provide the following:

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

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

c. Specifically describe what CBM credits are sought. (examples are extended operating time between maintenance, overhaul / inspection)

d. 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, integrated logistics support, and user training. The complexity and scope of the undertaking can cause uncertainty as to where or how to begin the process. The following guidance in Figure 3 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.

6.1 CBM for 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 reasonably achieve the four CBM goals.

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 (Figure 4). 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.

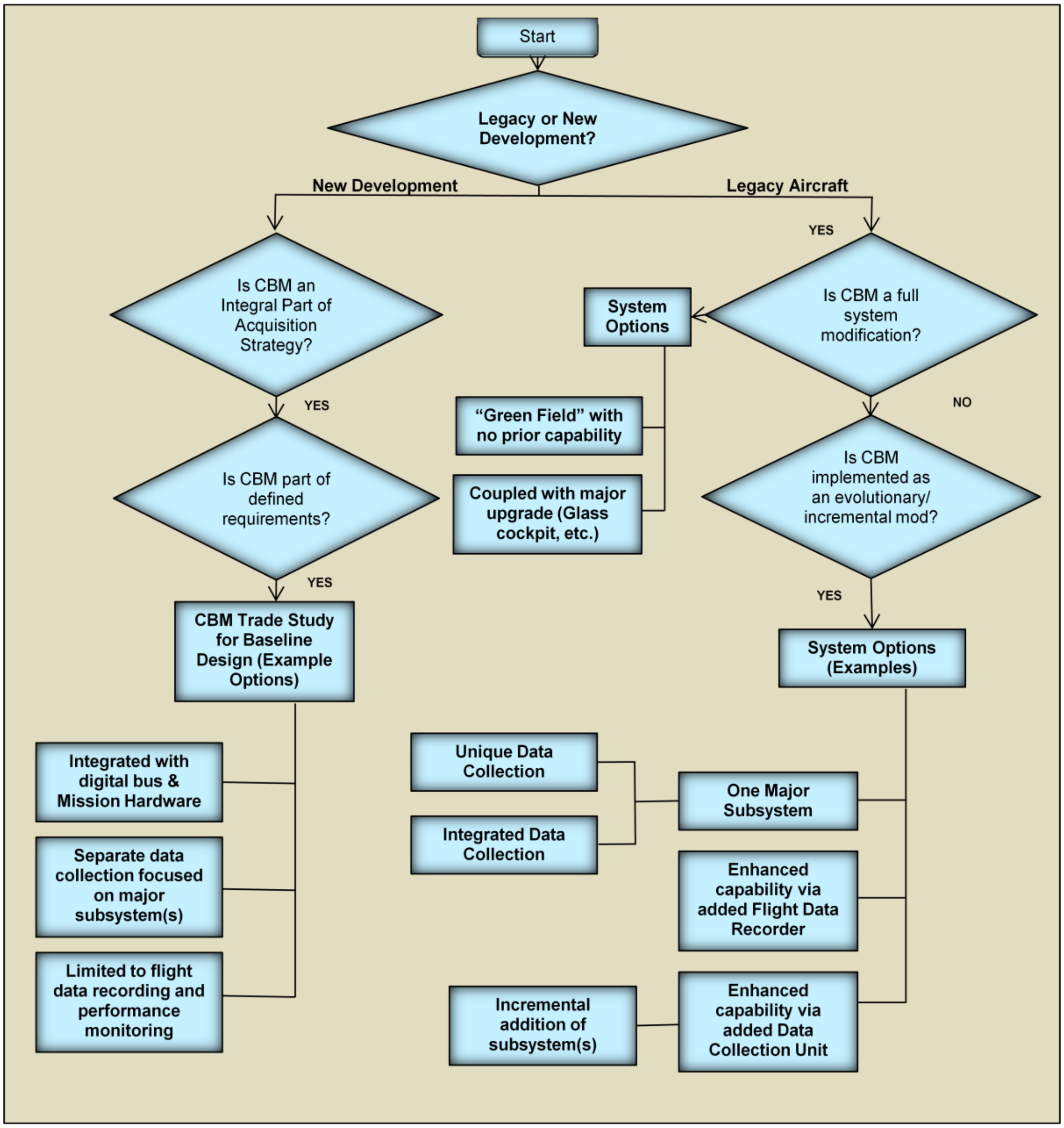

FIGURE 3. Mindmap of how to use ADS-79D-HDBK

Figure 4 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, 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 Appendices 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.

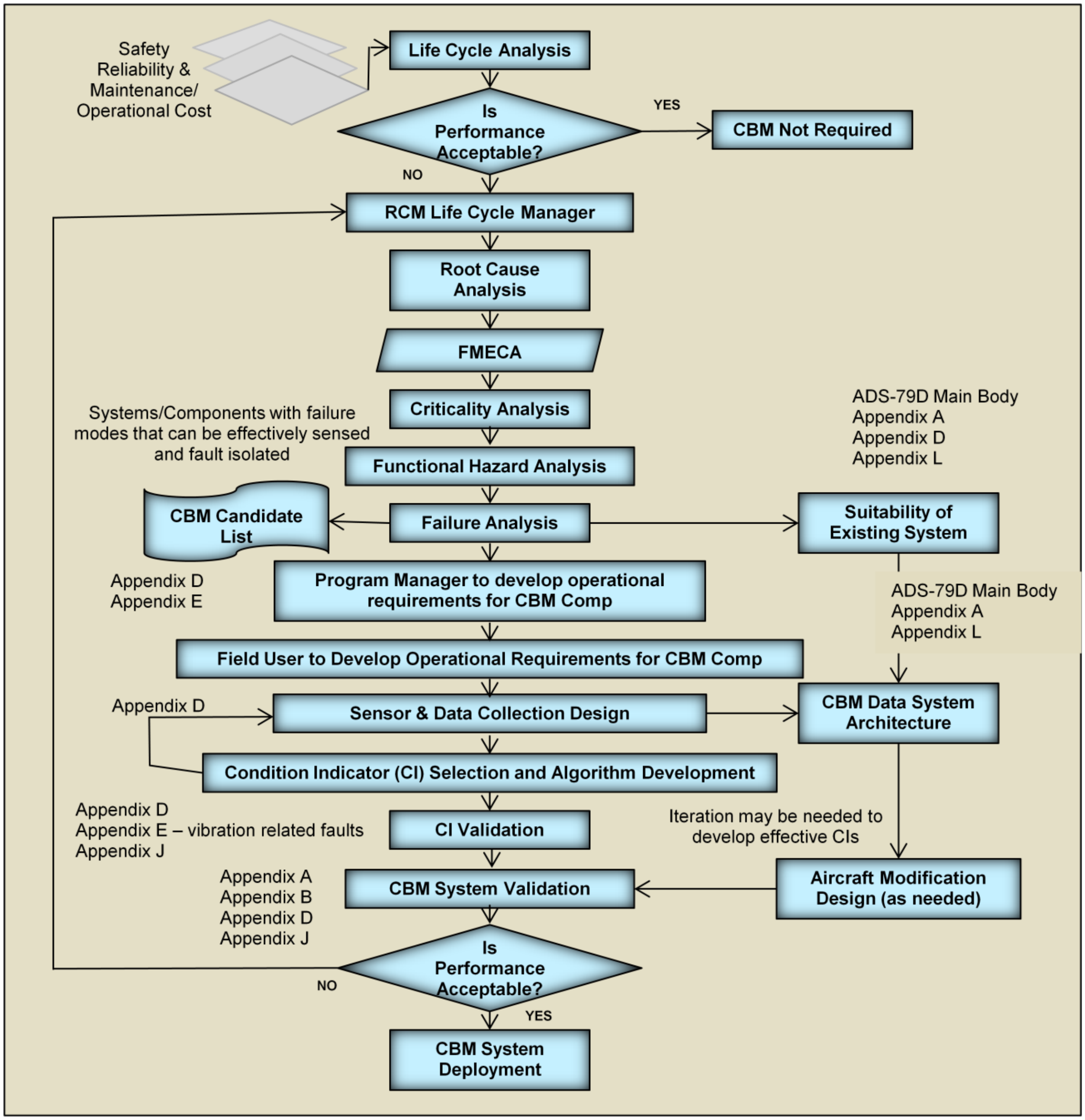

FIGURE 4 CBM development for legacy aircraft

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 H.

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 4, 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.

6.2 CBM for 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 5 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 FMECA. The FMECA documents the failure modes and effects of the system. Upon completion of the FMECA a 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 H contain additional guidance for this part of the analysis.

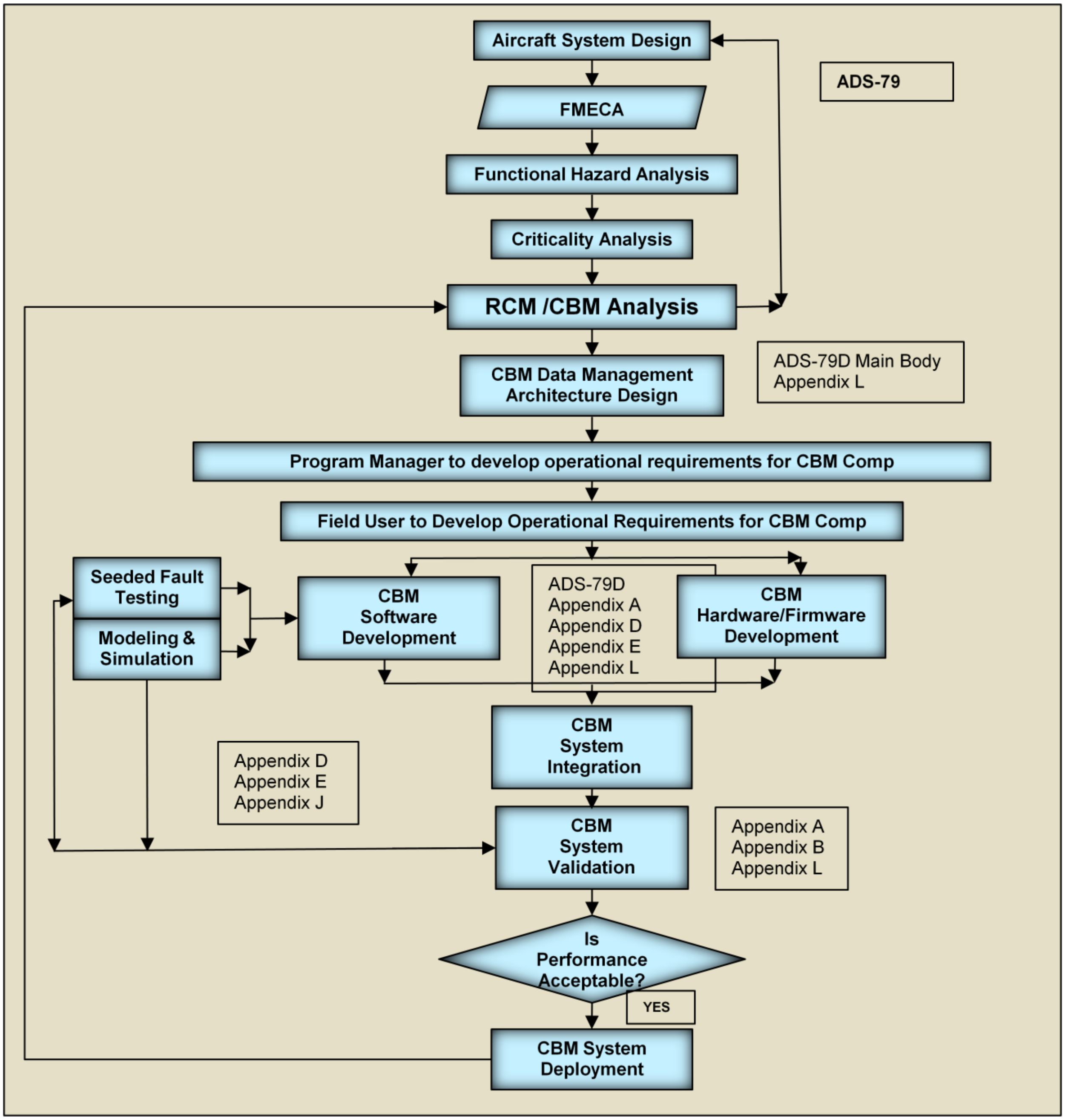

FIGURE 5. CBM development for new acquisition

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 F. 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.

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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 CBM Credit/Debit for incorporation of CBM into Army aircraft systems from a Fatigue Life Management (FLM) point of view. This appendix also documents potential applications of FLM.

A.2 APPLICABLE DOCUMENTS

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

(Copies of this document are available at http://www.redstone.army.mil/amrdec/rdmr-se/tdmd/StandardAero.htm)

JOINT SERVICE SPECIFICATION GUIDE	
JSSG-2001	Department of Defense Joint Service Specification Guide, Air Vehicle, 29 Jan 2009.
JSSG-2006	Department of Defense Joint Service Specification Guide, Aircraft Structure, 30 October 1998.

(Copies of this document are available from https://assist.daps.dla.mil/quicksearch/ DLA Document Services, Building 4/D, 700 Robbins Avenue, Philadelphia, PA 19111-5094. 215-697-6396.)

VARIOUS REFERENCES	
Benton, Robert E, Jr.	"Further Advances in a Recently Developed Cumulative-Damage Reliability Method", *American Helicopter Society 66th Annual Forum Proceedings*, Phoenix, AZ, May 11-13, 2010.
Collins, J. A.	Failure of Materials in Mechanical Design: Analysis, Prediction, Prevention. Wiley & Sons: New York, 1993.
Adams, D. O. 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.
Zhao, J. and D. O. Adams	"Achieving Six-Nine's Reliability Using an Advanced Fatigue Reliability Assessment Model", Presented at the American Helicopter Society 66th Annual Forum, Phoenix, AZ, May 11-13, 2010

(Copies of these documents are available from sources as noted.)

A.3. DEFINITIONS

Hazard: A real or potential condition that could lead to an unplanned event or series of events (i.e., mishap) resulting in damage to the system, death, or injury.

System: The organization of hardware, software, and data needed to perform a designated function within a state environment with specified results.

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), external stores, 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. Fatigue Life Management (FLM) and usage/load monitoring, using flight recorder data, will provide the information needed to determine and track this relationship.

An 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-79D-HDBK), the goals of the CBM system are to minimize burdensome maintenance tasks, increase aircraft availability, improve flight safety, and reduce maintenance cost. The primary mechanism of FLM is to enable updating of the usage spectrum required for maintaining airworthiness of Army aircraft systems.

The secondary mechanisms 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 in 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 Component Remediation. The Reliability Criteria for establishing maintenance actions based on a FLM system are provided in section A.6.

A.5 POTENTIAL APPLICATIONS

A.5.1 Updating service usage spectrums. The FLM system enhances the capability to update service 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 service usage. The number of aircraft required to participate in a usage survey should be statistically significant (as calculated using the statistical principals in section 3 of Appendix I).Likewise, a survey should be conducted at sufficient locations to ensure inclusion of all missions, including training locations, to ascertain appropriate usage severity. Ongoing usage monitoring (via regime recognition) is used to assess the need for an updated usage spectrum as well as the need for additional structural flight testing. Valid regime recognition data is used during planning and conduction of the pilot interview process (it should be noted that pilot interview data is required to update a US Army aircraft usage spectrum). Usage spectrum updates are accomplished based on valid field-representative data. This data includes required pilot interview data and available regime recognition data.

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 Ground Air Ground (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 claiming to maintain required 0.999999 (six nines) reliability using updated usage spectrum from HUMS is given in reference Adams and Zhao, AHS 2009[3] for the case where:

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

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

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

[3] 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.5.2 Managing service life of Safe-Life Structural components. The service life of structural CSIs 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 cumulative damage hypothesis.[4] 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.

a. 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.

b. 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.6.

c. 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.

d. 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 failures per 100,000 flight hour's system hazard) is met.

A.5.3 Component 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 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.

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

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 (based upon available field data).

b. Investigate regime recognition data for causal 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 Technical Manuals (TMs) and Depot Maintenance Work Requests (DMWRs) allowing the component to stay on the aircraft longer. Remediation enhances the four goals of FLM and can be considered a subset of analysis and correlation of data to component fatigue strength.

A.5.4 Managing service life of damage tolerant structure. FLM 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.[5] 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. When coupled with appropriate flight load survey data, the FLM derived actual usage, a direct load measurement or an updated usage spectrum will provide the load spectrum 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.[6] The inspection requirements for the repair/redesign must be substantiated to the reliability requirements of section A.6 to prevent a catastrophic failure. The FLM database will be utilized in the evaluation of existing structure, repairs, reinforcements, and redesigns.

Also, the FLM system has the potential to provide input to the user 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 FLM has the potential of significant improvements in readiness and reduction of sustainment costs for Army aircraft systems.

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

[6]JSSG-2001B, Department of Defense Joint Service Specification Guide, Air Vehicle, 29 Jan 2009.

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 the Program Executive Officer (PEO), Aviation System Safety Risk Management Process IAW MIL-STD-882. 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 system. Incremental incorporation should require allocation of risk.

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

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 system 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 in order to reduce operation and support costs, and hence, to improve readiness. FLM 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 (individual component fatigue damage assessment based on regime recognition) FLM usage monitoring tracks 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 appropriate distributions. Experimental data from fatigue characterization and 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 strengths are statistically modeled (individual component fatigue damage assessment based on loads monitoring) 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 and design damages applied (alternate individual component fatigue damage assessment based on regime recognition and design damage.) 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 detailed probabilistic analysis is not available for determination of component reliability, maximum accumulated damage should be tracked to no more than 0.5. Baseline retirement times are based on composite worst case design spectrum which is assumed to add one nine of reliability. The adjustment of the accumulated damage is to ensure baseline reliability is maintained when component damages are accumulated using the actual flight maneuvers[7]. . 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 Evaluation of reliability for usage spectrum update For the case of updating usage spectrums for legacy aircraft, statistical analysis of the usage data is used to determine a statistical approximation of an updated composite worst case spectrum as discussed in section A.5.1. Consider use of a "mean plus two sigma" spectrum[8] to avoid the need for additional fatigue strength working curve reductions or probability analysis. As an alternative a "mean plus sigma" spectrum may be applied with appropriate probability analysis. Use of a mean spectrum is not appropriate.

[7] Benton ,Robert E., Jr., "Further Advances in a Recently Developed Cumulative-Damage Reliability Method", *American Helicopter Society 66th Annual Forum Proceedings*, Phoenix, AZ, May 11-13, 2010.

[8] Note that the rule of thumb that "a factor of 2 in life is worth a factor of 10 in reliability" is verified via a probabilistic analysis example in J. Zhao and D. O. Adams, "Achieving Six-Nine's Reliability Using an Advanced Fatigue Reliability Assessment Model", Presented at the American Helicopter Society 66th Annual Forum, Phoenix, AZ, May 11-13, 2010.

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APPENDIX B

REGIME RECOGNITION/FLIGHT STATE CLASSIFICATION WITH VALIDATION OF REGIME RECOGNITION ALGORITHMS

B.1 SCOPE

B.1.1 Scope. This 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.

Various References	
Cronkhite, J., B. Dickson, W. Martin, and G. Collingwood DOT/FAA/AR-97/64	Operational Evaluation of a Health and Usage Monitoring System (HUMS), April 1998
McCool, K. and B. Barndt.	"Assessment of Helicopter Structural Usage Monitoring System Requirements," DOT/FAA/AR-04/3, April 2004.
Thompson, Audbur E. and David O. Adams	"A Computation Method for the Determination of Structural Reliability of Helicopter Dynamic Components", Presented at the AHS Annual Forum, May 1990. http://toc.proceedings.com/11824webtoc.pdf
Vaughan, Robert E., J. Chang, M. Rogers	"Obtaining Usage Credits from Monitoring of Helicopter Dynamic Components without Impacting Safe Life Reliability", Presented at the AHS 63rd Annual Forum, May, 2007. http://toc.proceedings.com/11807webtoc.pdf

(Copies of these documents are available from sources as noted.)

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.

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

US ARMY AERONAUTICAL DESIGN STANDARD	
ADS-51 HDBK	Rotorcraft and Aircraft Qualification Handbook, 21 Oct 1996.

(Copies of this document are available at http://www.redstone.army.mil/amrdec/rdmr-se/tdmd/StandardAero.htm)

ASTM INTERNATIONAL (AMERICAN SOCIETY FOR TESTING AND MATERIALS)	
ASTM E1049-85(2011)e1	Standard Practices for Cycle Counting in Fatigue Analysis.
ASTM D664-11A	Standard Test Method for Acid Number of Petroleum Products by Potentiometric Titration.

(Copies of these documents are available online at http://www.astm.org or from the ASTM International, 100 Barr Harbor Drive, P.O. Box C700, West Conshohocken, PA 19428-2959.)

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.

DEPARTMENT OF TRANSPORTATION	
DOT/FAA/AR-04/19	Hazard Assessment for Usage Credits on Helicopters Using Health and Usage Monitoring System

(Copies of this document are available from http://www.tc.faa.gov/its/worldpac/techrpt/ar04-3.pdf) or write Office of Aviation Research, Washington, D.C. 20591.

B.3 DEFINITIONS

Structural Usage Monitoring: Managing fatigue lives via Usage Monitoring

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 a design mission spectrum determined by the class of aircraft. In a CBM system, however, component life calculations can be refined through knowledge of the actual service amount of operational time spent in each flight regime. CRTs can be extended when an aircraft is actually exposed to less severe mission profiles. Alternatively, in the interest of safety, CRTs can be reduced in the presence of more severe mission profiles than accounted for 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 strains/loads are determined during the flight load survey performed during the development phase of the aircraft. Next, analysis is performed to determine the rate of life expenditure 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 to enable accurate classification of 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 based on 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.

Although various CBM system architectures may be used to implement CBM processes such as Structural Usage Monitoring, the CBM system architecture planned for a given aircraft platform should be consistent with US Army infrastructure and outlined in the CBM management plan (see, for example Figure B-1).

Figure B-2 and Table B-I describe the processes and data necessary for regime recognition with usage spectrum update. Because all items in the fleet of the same part number are affected by a usage spectrum update, this is often referred to as a "part number" methodology . Similarly, Figure B-3 and Table B-II describe the processes and data necessary for regime recognition with individual component fatigue damage assessments. Because individual component fatigue damage assessments are performed individually for each serial number item based on its unique usage history, this is often referred to as a "serial number" methodology. Finally, the processes and data necessary for loads monitoring and estimation are described in Figure B-4 and Table B-III.

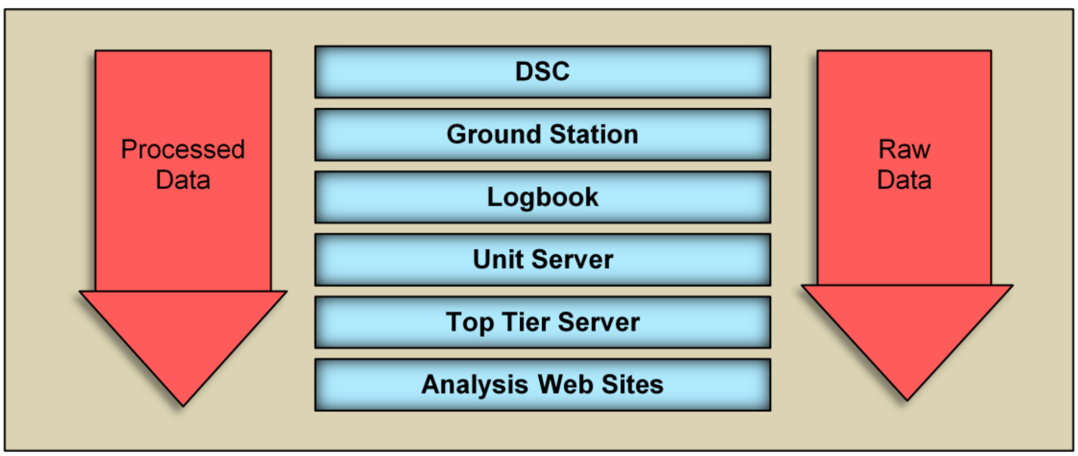

FIGURE B-1. Army flow of data for regimes

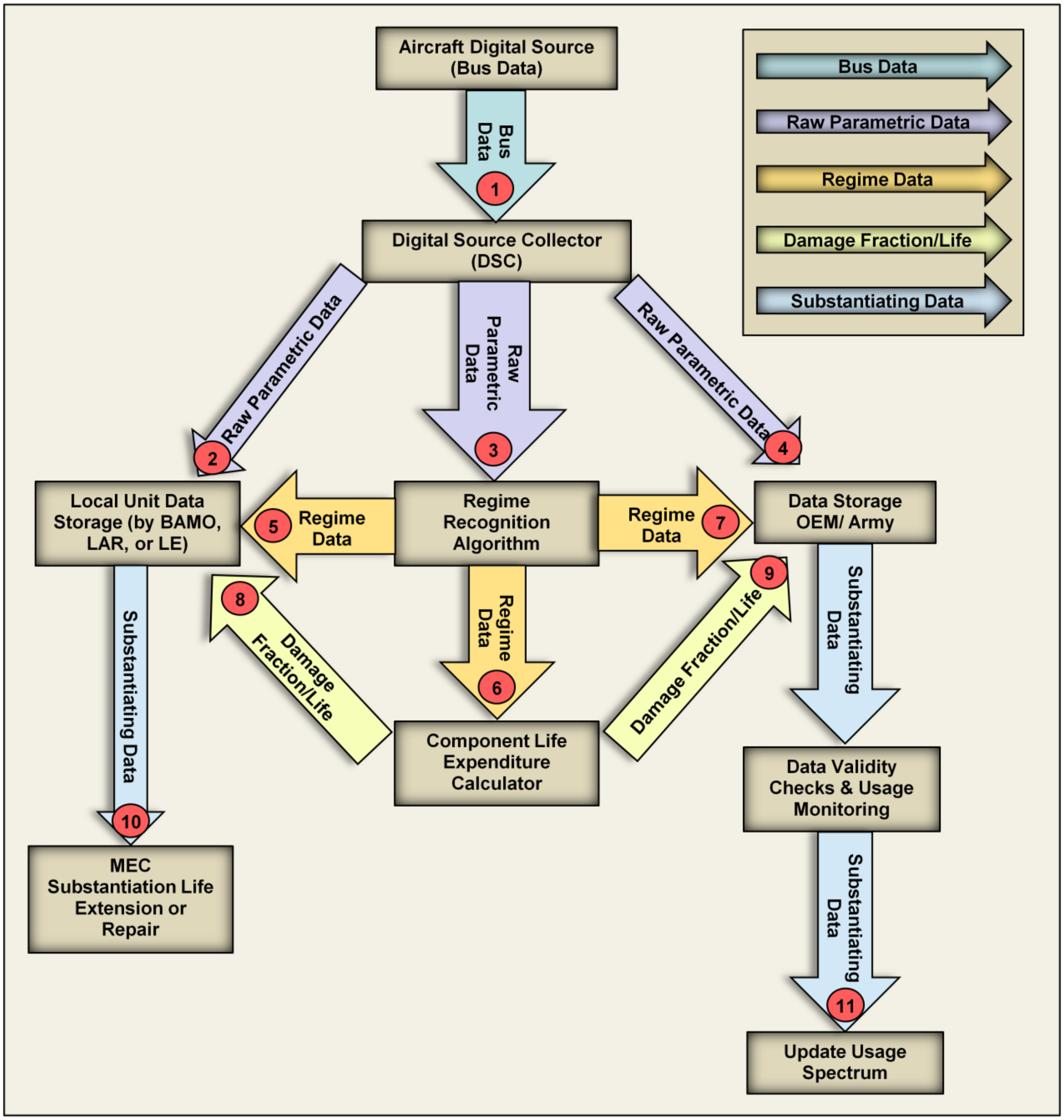

FIGURE B-2. Regime recognition processes with usage spectrum update

TABLE B-I. Data streams used in regime recognition processes with usage spectrum update

Number	Type of Data	Purpose	Category
1	Bus Data	Stored by DSC to Enable Regime Recognition	Inherent
2	Raw Parametric Data	Stored at Local Unit (by Battalion Aviation Maintenance Officer (BAMO), Logistics Assistance Representative (LAR), Logistics Engineer (LE), etc.) for Unit Purposes	Unit Discretion
3	Raw Parametric Data	Processed by Regime Recognition Algorithm	Inherent
4	Raw Parametric Data	Troubleshooting Regime Recognition Algorithm for Unidentified Intervals	On Condition
		Auditing Regime Recognition Algorithms	Statistical Sampling
5	Regime Data	Stored at Local Unit (by BAMO, LAR, LE, etc.) for Unit Purposes	Unit Discretion
6	Regime Data	Processed by Component Life Expenditure Calculator	Inherent*
7	Regime Data	Usage Monitoring and Storage for Potential Usage Spectrum Updates	Required
8	Damage Fraction/Life Data	Stored at Local Unit (by BAMO, LAR, LE, etc.) for Unit Purposes	Unit Discretion
9	Damage Fraction/Life Data	Auditing Component Life Expenditure Calculator	Statistical Sampling
10	Substantiating Data	Maintenance Engineering Call (MEC) Substantiation	Unit Discretion
11	Substantiating Data	Usage Spectrum Monitoring	Periodic
		Usage Spectrum Updates	On Condition

*although not strictly inherent, developers should treat this type of data as if inherent to the process.

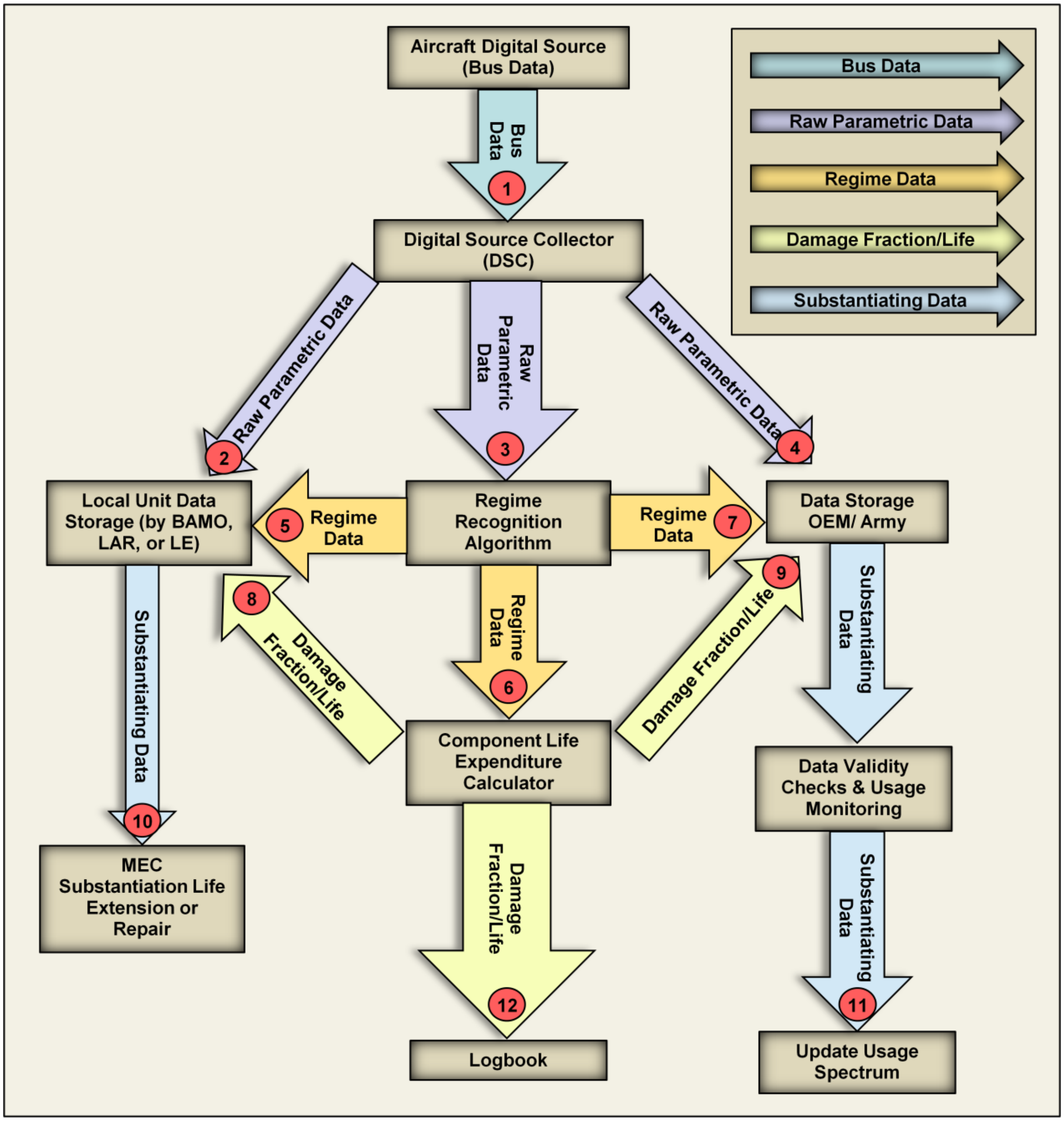

FIGURE B-3. Regime recognition processes with individual component fatigue damage assessment

TABLE B-II. Data streams used in regime recognition processes with Individual component fatigue damage assessment

Number	Type of Data	Purpose	Category
1	Bus Data	Stored by DSC to Enable Regime Recognition	Inherent
2	Raw Parametric Data	Stored at Local Unit (by BAMO, LAR, LE, etc.) for Unit Purposes	Unit Discretion
3	Raw Parametric Data	Processed by Regime Recognition Algorithm	Inherent
4	Raw Parametric Data	Troubleshooting Regime Recognition Algorithm for Unidentified Intervals	On Condition
		Auditing Regime Recognition Algorithms	Statistical Sampling
5	Regime Data	Stored at Local Unit (by BAMO, LAR, LE, etc.) for Unit Purposes	Unit Discretion
6	Regime Data	Processed by Component Life Expenditure Calculator	Inherent
7	Regime Data	Usage Monitoring and Storage for Potential Usage Spectrum Updates	Required
8	Damage Fraction/Life Data	Stored at Local Unit (by BAMO, LAR, LE, etc.) for Unit Purposes	Unit Discretion
9	Damage Fraction/Life Data	Auditing Component Life Expenditure Calculator	Statistical Sampling
10	Substantiating Data	Maintenance Engineering Call (MEC) Substantiation	Unit Discretion
11	Substantiating Data	Usage Spectrum Monitoring	Periodic
		Usage Spectrum Updates	On Condition
12	Damage Fraction/Life Data	Tracked in Logbook	Inherent

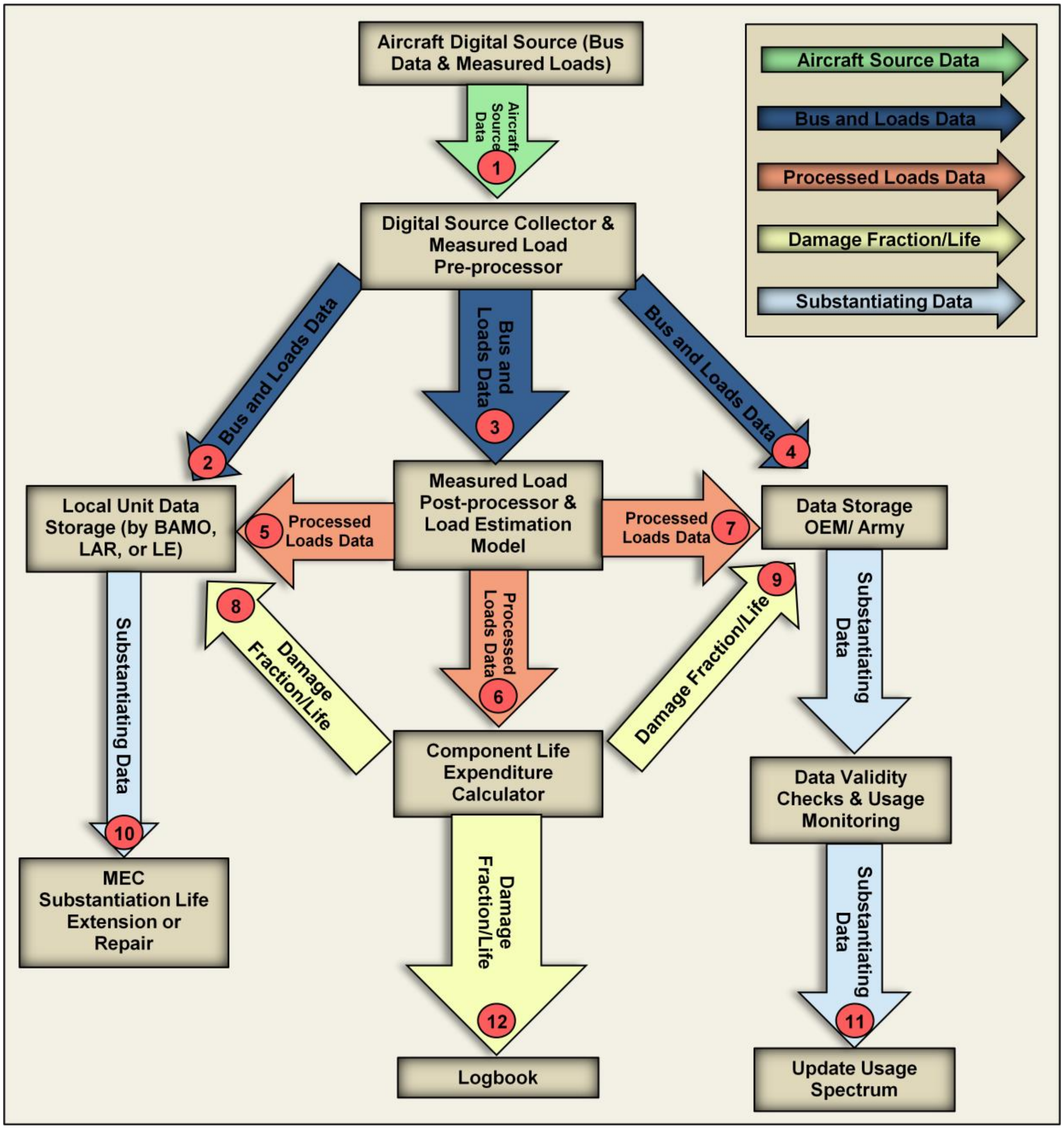

FIGURE B-4. Loads monitoring and estimation processes

TABLE B-III. Data streams used in load monitoring and estimation processes

Number	Type of Data	Purpose	Category
1	Aircraft Source Data	Stored by DSC to Enable Loads Monitoring and Estimation	Inherent
2	Raw Bus and Loads Data	Stored at Local Unit (by BAMO, LAR, LE, etc.) for Unit Purposes	Unit Discretion
3	Raw Bus and Loads Data	Processed by Measured Load Post-processor and Load Estimation Model	Inherent
4	Raw Bus and Loads Data	Troubleshooting Load Monitoring and Estimation	On Condition
		Auditing Load Monitoring Post-processor and Load Estimation Model	Statistical Sampling
5	Processed Loads Data	Stored at Local Unit (by BAMO, LAR, LE, etc.) for Unit Purposes	Unit Discretion
6	Processed Loads Data	Processed by Component Life Expenditure Calculator	Inherent
7	Processed Loads Data	Usage/Loads Monitoring and Storage for Potential Usage/Loads Spectrum Updates	Required
8	Damage Fraction/Life Data	Stored at Local Unit (by BAMO, LAR, LE, etc.) for Unit Purposes	Unit Discretion
9	Damage Fraction/Life Data	Auditing Component Life Expenditure Calculator	Statistical Sampling
10	Substantiating Data	Maintenance Engineering Call (MEC) Substantiation	Unit Discretion
11	Substantiating Data	Usage/Loads Spectrum Monitoring	Periodic
		Usage/Loads Spectrum Updates	On Condition
12	Damage Fraction/Life Data	Tracked in Logbook	Inherent

B.5 DETAIL 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 components and structural elements 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, or 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). As depicted in Figure B-5, these regimes form the basis of fatigue calculations and should also form the basic requirement for regime recognition algorithms. Two validation loops are shown in Figure B-5 and described in Table B-IV. In addition to the regime recognition system's ability to identify usage in an operational environment and a key factor in the successful implementation of a regime recognition algorithm is whether the regime recognition matches the flight loads survey test points including consideration of flight test maneuver descriptions and tolerances used during the flight load survey. A series of flights should be performed with a test aircraft that is fully equipped with the regime measurement package (such as the DSC) and additional recording systems for capturing data needed to evaluate and tune the algorithms. One may establish that a 0.5% under-prediction of damage fraction could introduce a 5% increase in probability of failure. To avoid under predicting damage fraction by more than 0.5%, it is recommended that the regime recognitions algorithms be required to demonstrate that they can identify sufficient regimes, such as 97% or greater of the actual flight regimes, including all highly damaging maneuvers and benign maneuvers with high frequencies of occurrence. Targeted regimes should be selected such that any unrecognized regimes would introduce less than 0.5% under-prediction of fatigue damage fraction based on the design usage spectrum. Also, for misidentified or unrecognized flight regimes, the system should demonstrate that it errs on the side of selecting a more severe regime. This ensures that a component is not allowed to receive maintenance credit where it is not due and therefore prevents a component from being flown beyond its margin of safety.

Cronkhite, *et al.* (1998) provides an example which cautions against the temptation to identify overly-broad flight conditions.[9] Although broad flight conditions would allow one to claim credit for identification of a high percentage of flight times with little effort, the lack of correlation between broad categories and the certification spectrum may result in fatigue damage accrual rates that do not sufficiently represent those corresponding to more refined regime categories.

[9] Cronkhite, J., B. Dickson, W. Martin, and G. Collingwood DOT/FAA/AR-97/64 Operational Evaluation of a Health and Usage Monitoring System (HUMS), April 1998.

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

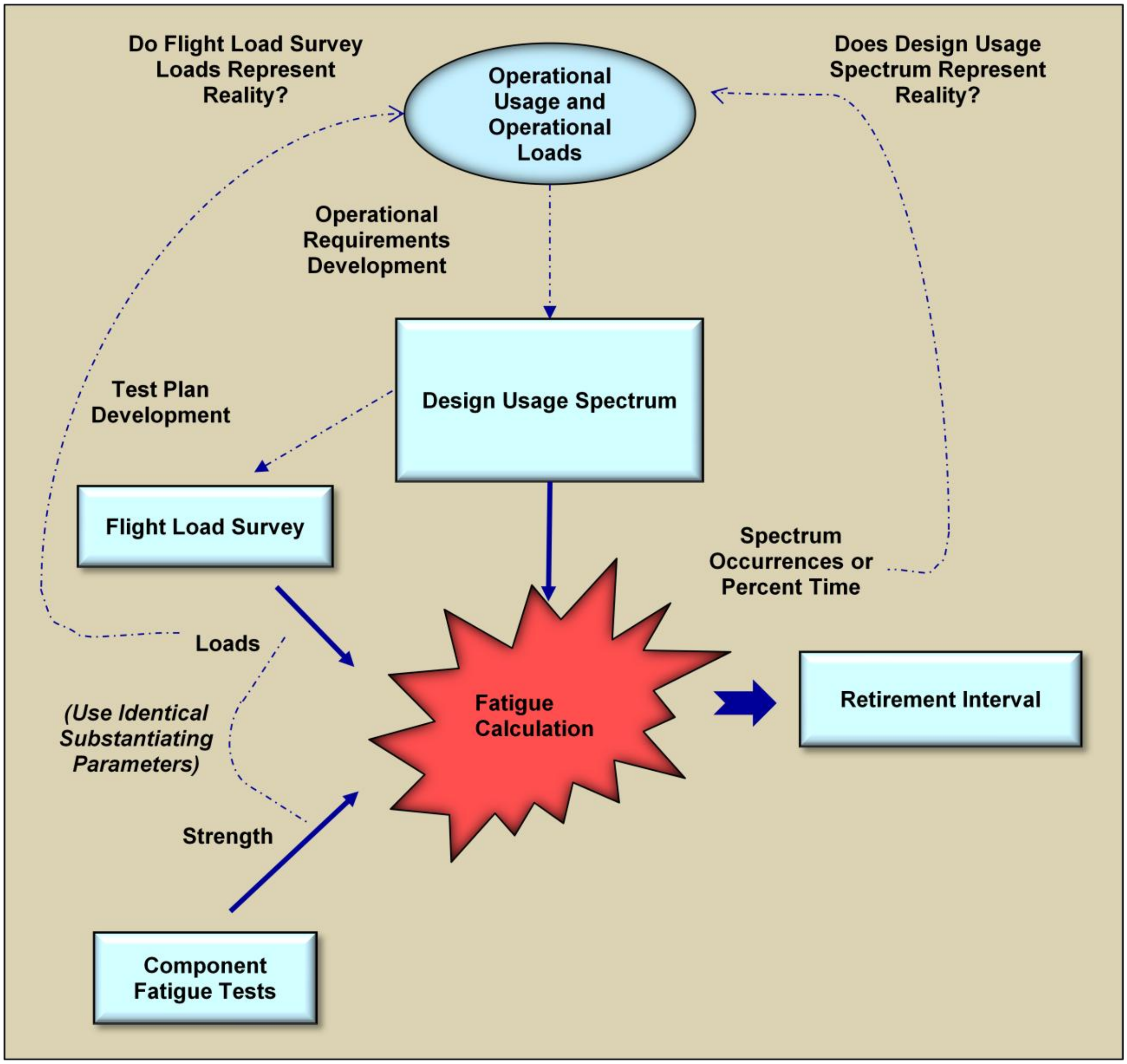

FIGURE B-5: Fatigue life management usage and load validation loops

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-V 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-IV. Fatigue life management validation challenges

Validation Loop	Primary Question	Challenges to Overcome
Usage Validation	How well does the Design Usage Spectrum represent operational usage?	Sampling issues (must represent all units, missions, theaters, and threat environments)
		Maneuver standards (does regime recognition cover flight load survey regime?)
Loads Validation	How well do Flight Load Survey loads represent operational loads?	Loads variability (pilots, air quality, gusts, aircraft manufacturing tolerances)
		Load binning effects
		Maneuver standards (does regime recognition cover flight load survey loads?)

The sample list of components in Table B-V 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.

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

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

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

TABLE B-VI. Typical military helicopter flight environment parameters (EXAMPLE ONLY)	
	Local Base Environment – Off Board Data Collection
1	Geographic Description of Theater (Desert, Mountains, etc)
2	Shipboard Operations (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 Digital source collector design for structural usage monitoring.

B.5.2.1 Onboard flight state sensing. Flight state parameters are used as inputs to the regime classification algorithms. According to McCool and Barndt (2004), Gross Weight, Airspeed, Altitude, and Outside Air Temperature are four key parameters. These parameters represent very important measures of aircraft usage and loads and are likely to characterize the flight test maneuver load database and fatigue calculations for most platforms. Although these and other important state parameters may be estimated or derived from various other sources of input, the resulting accuracy and fidelity should be consistent with the range of operational load conditions and configurations intended to be covered by each flight test regime and its associated description and tolerances. The set of flight state inputs provided in Table B-VII is intended to serve as an example of the type of parameters which a hypothetical regime classification algorithm may use. The digital source collector for a particular aircraft should be designed to collect information necessary to either directly record or indirectly estimate/derive the required input parameters for the aircraft's particular regime classification algorithm.

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 ensuring that flight state sensors are guaranteed to be operational and calibrated as part of normal aircraft maintenance procedures.

The blade stall indication listed in the Table B-VII example could represent recording a Cruise Guide Indicator (CGI) signal such as for Boeing Philadelphia products, estimation/ calculation of Sikorsky's Equivalent Retreating Indicated Tip Speed (ERITS), or some novel approach to indicating blade stall. It is noted that the example lists various notional derived parameters which may be of use for similar purposes, including Referred Gross Weight, Blade Load ("C_T/σ"), and advance ratio ("μ"). Airspeed ratios to the maximum level flight airspeed (V_h) or the "never exceed" airspeed (V_{ne}) are often used to characterize airspeed in fatigue calculations and may be based on tabulated values for V_h or V_{ne} as a function of Referred Gross Weight. Use of these blade stall indications or tabulated airspeed characterizations should be considered when determining the fidelity requirements for Altitude or Gross Weight estimation. Mission planning may provide supplemental data to assist with selecting appropriate V_h or V_{ne} values for the flight.

In addition to the parameters shown in Table B-VII, regime classification algorithm designers may also consider whether the potential usefulness would justify the additional expense of requiring the digital source collector to monitor control input rates, flight control actuator loads, blade flapping, swashplate tilt, aircraft longitudinal/lateral CG accelerations, parking brake indication, trim ball indication, ground speed, ground track, and miscellaneous strain measurements.

TABLE B-VII: Typical state parameters required for structural usage monitoring, including measured and derived parameters (EXAMPLE ONLY)

PARAMETER			PARAMETER	
1	Aircraft Tail Number		20	Rotor Mast Torque (if available)
2	Date or Unique Flight Sequence Number		21	Engine Torque (for each engine)
3	Time Indication or Elapsed Time		22	Longitudinal Cyclic Position
4	Outside Air Temperature (OAT)		23	Lateral Cyclic Position
5	Pressure Altitude		24	Collective Position
6	Density Altitude (possibly derived)		25	Pedal Position
7	Radar Altitude		26	Heading
8	Indicated Airspeed		27	Pitch Attitude
9	True Airspeed (possibly derived)		28	Roll Attitude
10	Calibrated Airspeed (possibly derived)		29	Yaw Rate
11	Main Rotor Speed		30	Pitch Rate
12	Rate of Climb/Descent		31	Roll Rate
13	Gross Weight (possibly derived/estimated)		32	Weight on Wheels/Gear Indication
14	Referred Gross Weight (derived)		33	Rotor Brake Indication
15	Long/Lat CG Position (possibly derived)		34	Percent VH (derived)
16	Fuel Quantity (for GW/CG derivation)		35	Percent VNE (derived)
17	External Load (cargo hook – for GW/CG)		36	Blade Stall Indication (derived/measured)
18	Weapon Stores Indication (for GW/CG)		37	Advance Ratio (derived)
19	Normal CG Load Factor		38	Blade Load CT/SIGMA (derived)

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-6 from the referenced report shows the maximum load factor that would be recorded for a pull-up maneuver at 2 different sample rates.[11] Figure B-6 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.

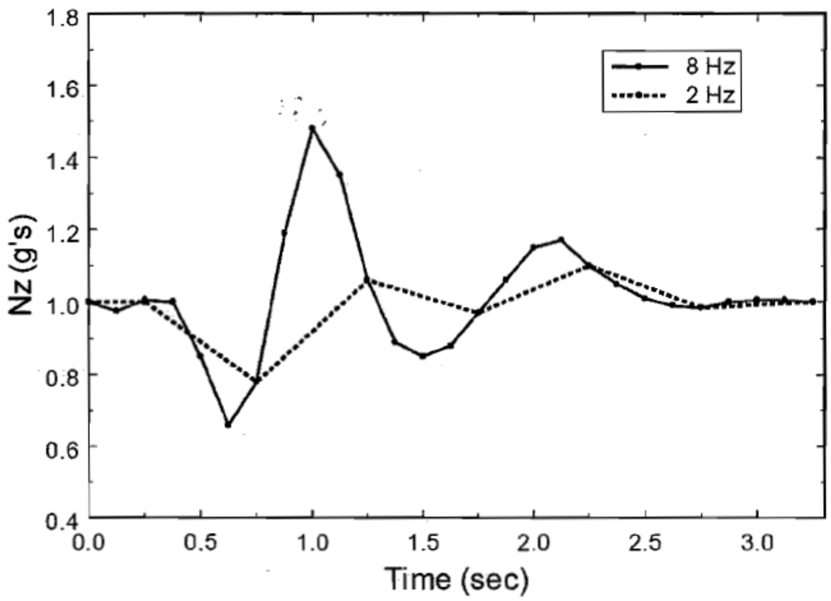

FIGURE B-6. Effect of data rate on vertical acceleration[12] (EXAMPLE ONLY)

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-VIII shows the example data rates for military helicopters for each parameter. Using the example rates in Table B-VIII should not be considered a substitute to performing the validation described in paragraph B.5.4

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 events to each regime should be developed. Although one may elect to perform regime classification and allocated flight recording in real-time

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

[11] *Ibid.*

[12] *Ibid.*

onboard the aircraft, storage of raw unprocessed flight state measurements for later processing is preferred, provided sufficient available onboard data storage capacity for selected sample rates.

B.5.2.4 Component lifecycle tracking. To enable individual component fatigue 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 be maintained as part of the maintenance logistics process which incorporates appropriate quality assurance processes to avoid duplicate or nonsensical data entries and contains indentured parts lists with component serial numbers for each aircraft tail 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 ensures that a component serial number is not orphaned without any means of determining its retirement time.

TABLE B-VIII. McCool and Barndt proposed data rates[13] (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

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 individual component fatigue damage assessments.

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

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 each component part number 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 Digital source collector validation for structural usage monitoring. 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. Maneuver descriptions and tolerances should match those used during the flight load survey. 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, after completion of optimization to achieve the designated accuracy, a comprehensive flight card should be developed which incorporates all of the maneuvers for which the algorithms have been developed. Without being provided knowledge of the flight card content, the regime recognition design team should demonstrate the ability to identify the maneuvers flown, their severity and duration. Sufficient flight time and flight conditions should be properly identified such that any unrecognized regimes would introduce less than 0.5% under-prediction of fatigue damage fraction based on the design usage spectrum.

B.5.3.2 Accuracy. To avoid under predicting damage fraction by more than 0.5%, it is recommended that the CBM regime recognitions algorithms should demonstrate that they can identify sufficient regimes, such as 97% or greater of the actual flight regimes, including all highly damaging maneuvers and benign maneuvers with high frequencies of occurrence. Maneuver descriptions and tolerances should match those used during the flight load survey. Targeted regimes should be selected such that any unrecognized regimes would introduce less than 0.5% under-prediction of fatigue damage fraction based on the design usage spectrum. Also, for misidentified or unrecognized flight regimes, the system should demonstrate that it errs on the side of selecting a more severe regime. This ensures 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 objective of the following is to provide guidelines for the qualification of a Structural Usage Monitoring System (SUMS) that will establish the basis for maintaining the reliability for the entire lifecycle of the aircraft in accordance with Appendix A. Fully validated SUMS should be considered an intimate part of the airworthiness process throughout the aircraft's lifecycle. Accordingly, the SUMS process should be included in the airworthiness qualification process for the aircraft, including consideration of SUMS diagnostics and alternative means of achieving reliability in the event of a SUMS failure.

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 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 two peak loads from within the same maneuver, 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. Any Parametric, Regime, and Damage Fraction/Life data stored on the ground station should be stored using a common non-proprietary binary format.

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 documented as will a procedure to address a condition of an inoperative monitoring system. Any Parametric, Regime, and Damage Fraction/Life data stored on the ground station should be stored using a common non-proprietary binary format which is clearly specified within appropriate interface control documentation to allow third parties to build data conversion routines, as necessary to meet changing or future joint-platform requirements.

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 with comparison to maneuver descriptions and tolerances used during the flight load survey. The purpose of these flight tests is to verify that the usage monitoring system can identify the significant regimes of the usage spectrum including highly damaging maneuvers and benign maneuvers with high frequencies of occurrence. 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 rainflow cycle counting[14] 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 rainflow cycle counted loads from flight start to flight end for comparison to the usage based damage sum and the maneuver load based damage sum.

[14] "ASTM E1049-85 Standard Practices for Cycle Counting in Fatigue Analysis"

APPENDIX C

STRUCTURAL HEALTH AND LOADS MONITORING

C.1 SCOPE

This Appendix provides guidance for incorporation of Structural Health Monitoring (SHM) and Loads Monitoring and Estimation (LME) systems into an aircraft system's Condition Based Maintenance (CBM) Management Plan. Structural Health Monitoring is a fleet management concept that allows evaluation of the structural health of an aircraft throughout its life cycle based on measured data. The purpose of evaluating structural health is the prognosis of future performance. Future performance predictions are based on comparing the current state of the structure with initial and degraded system states. The initial state is the "as manufactured" system where structural capability is substantiated by analyses and tests. The reference degraded state corresponds to the minimum structural capability required for the aircraft to perform its intended function. Sensors are utilized to monitor structural degradation due to the service environment experienced by the aircraft in assessing its real time capability for the maintainer, inherent material degradation, and component wear-out. Typical monitoring includes strains/loads and for the presence of structural damage.. Detailed guidance included in this appendix addresses the Structural Health Monitoring system, integrated Non-Destructive Inspection (NDI) methods, load monitoring, load estimation, and limitations on use of vibration measurements.

C.2 REFERENCES AND APPLICABLE DOCUMENTS

C.2.1 References.

VARIOUS REFERENCES	
Rummel, Ward D.	"Recommended Practice for a Demonstration of Nondestructive Evaluation (NDE) Reliability On Aircraft Production Parts", Materials Evaluation, Volume 40, No. 9, 1982.
Zhao, J., Justin Wu, M. Urban, and Douglas Tristch.	"Optimization of Inspection Planning for Probabilistic Damage Tolerance Design", presented at the American Helicopter Society 67th Annual Forum, Virginia Beach, Virginia, May 1-3, 2011.

(Copies of these documents are available from sources as noted.)

C.2.2 Other Government and Non-Government guidance documents. The following documents should be used to complement the guidance of this handbook.

US ARMY AERONAUTICAL DESIGN STANDARD	
ADS-51 HDBK	Rotorcraft and Aircraft Qualification Handbook, 21 Oct 1996.

(Copies of this document are available at http://www.redstone.army.mil/amrdec/rdmr-se/tdmd/StandardAero.htm)

ASTM INTERNATIONAL (AMERICAN SOCIETY FOR TESTING AND MATERIALS)	
ASTM E2862-12	Standard Practice for Probability of Detection Analysis for Hit/Miss Data, February 2012.
ASTM E1316-11b	Standard Terminology for Nondestructive Examinations, January 2012.

(Copies of these documents are available online at http://www.astm.org or from the ASTM International, 100 Barr Harbor Drive, P.O. Box C700, West Conshohocken, PA 19428-2959.)

ASM INTERNATIONAL The Materials Information Society	
ASM Handbook, Volume 17	Nondestructive Evaluation and Quality Control

(Copies of these documents are available online at http://www.asminternational.org or from the ASM International, 9639 Kinsman Rd., Materials Park, OH 44073-0002.)

C.2.3. Government Documents. The following specifications, standards, and handbooks form a part of this document to the extent specified herein.

MILITARY STANDARDS (MIL-STDs)	
MIL-STD-810	Department of Defense Test Method Standard for Environmental Engineering Considerations and Laboratory Tests, 31 October 2008
MIL-STD-461	Department of Defense Interface Standard Requirements for the Control of Electromagnetic Interference Characteristics of Subsystems and Equipment, 10 December 2007.
MIL-HDBK-1823	Department of Defense Handbook – Nondestructive Evaluation System Reliability Assessment, 07 April 2009.

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

C.2.4 Non-Government Documents. The following specifications, standards, and handbooks form a part of this document to the extent specified herein.

OTHER	
Rummel, Ward	Recommended Practice for a Demonstration of Nondestructive Evaluation (NDE) Reliability on Aircraft Production Parts, Materials Evaluation Volume 40, No. 9, 1982.

(Copies of these documents are available from sources as noted.)

C.3 DEFINITIONS

Structural health monitoring (SHM). Structural Health Monitoring is a fleet management concept that allows evaluation of the structural integrity of an aircraft throughout its life cycle based on measured data. SHM uses one of many technologies to monitor aircraft structural capabilities, including integrated NDI methods (algorithms, instruments, software procedures).

Loads estimation. Equipment, techniques, or procedures to estimate the loads (forces or moments) experienced by an aircraft component during operational flight.

Non-destructive inspection (NDI). Methods used to check the soundness of a structural element or component without impairing or destroying the serviceability of the structural element or component. Methods include visual, magnetic particle, liquid penetrant, eddy current, ultrasonic, radiographic, etc.

Integrated NDI. Methods that may be integrated into the design of a structural element or component which are used to check the soundness of the structural element or component without impairing or destroying the serviceability of the structural element or component.

C.4 GENERAL GUIDANCE

C.4.1 Aircraft mission performance impact**.** SHM and LME systems have the potential to enhance aircraft availability and should be incorporated into CBM Management Plans as soon as practical. However, aircraft availability should never be allowed to depend on operability of the SHM or LME system. Alternate means of monitoring structural health, such as traditional NDI techniques with appropriate inspection intervals, maintenance schedules, and component retirement intervals, should be available for use in cases where the SHM system experiences a loss of function. Plans for SHM/LME system development and fielding should incorporate appropriate logistics support with provisions for diagnostics, component replacement, and repair.

C.4.2 Airworthiness qualification guidance**.** SHM and LME systems should be fully qualified in accordance with the aircraft system specification, including operation in specified environmental conditions. In accordance with guidance contained in ADS-51-HDBK, qualification requirements should be documented in an Airworthiness Qualification Plan (AQP) and an Airworthiness Qualification Specification (AQS).

C.5 DETAIL GUIDANCE

C.5.1 Monitoring system. The SHM /LME system includes, as a package; sensor elements, processing and communication chips, and a power supply. The SHM/LME system must be designed to be both reliable and durable. In addition, SHM/LME component installations should not expose aircraft components to foreign object debris hazards. The design life of the system should match the design service life of the structure being monitored. Likewise, the design environment and requirements for the SHM/LME system must be compatible with the structure. Qualification of the system for environmental and electromagnetic compatibility will be performed using the latest versions of MIL-STD-810 and MIL-STD-461 as standards, respectively. The ability of the system to identify the health of the structure and its ability to account for the variability of the “as manufactured” system must be validated.

Sensor selection will be based on the structure being monitored, its potential failure modes, and the data required to establish structural health. Identification of sensor requirements may require detailed evaluation of the structure and review of its service history in order to identify critical failure modes and potential hot spots. A special application exists in cases where the rate of structural deterioration is known to increase under certain aircraft regimes or Ground Air Ground (GAG) conditions. For this application, counter and timers will provide useful data in establishing structural health. A second special application is the use of thermal sensors to monitor the health of structural joints such as the lead lag joint of a main rotor system.

Details of the validation of the SHM/LME system should be documented in the CBM Management Plan. The capability of the system to evaluate structural health or monitor/estimate loads must be substantiated by analysis and test. Likewise, the durability of the system must be demonstrated including demonstration of any calibration/recalibration necessary to ensure that the system continues to perform in accordance with appropriate measurement standards. The SHM/LME system must be substantiated to meet the reliability requirements presented in Appendix A for Fatigue Life Management as specified in section A.6 to prevent a catastrophic failure in order to achieve airworthiness credit that expands or eliminates current maintenance actions. The structure must be restored to ultimate strength capability when the SHM/LME system and follow on inspections reveal that the structure can no longer support ultimate loads.

C.5.2 Integrated NDI methods. Alternative NDI methods are being developed based on various arrays or patterns of micro-sensors and actuators designed to detect structural health issues such as corrosion or crack detection. Many of these technologies depend on analysis, test, or field experience to determine the most likely location for crack initiation. Others claim to detect changes (such as growing cracks) in larger structures. In either case, the probability of detection must be substantiated for each critical crack initiation site being protected (see MIL-HDBK-1823 and ASTM E2862-12 for details on establishing appropriate probabilities of detection).

Every NDI system must be qualified and validated on a case by case basis. The design of experiments approach may be best to quantify various influencing variables, particularly the human factors. These CBM systems must be evaluated for their particular human factors. Rummel provides guidelines for demonstration of NDI reliability on aircraft production parts and contains information for development of a valid repeatable NDI demonstration program. [15]

Although these alternative NDI methods may not necessarily improve on the probability of detection of established NDI methods, the ability to integrate NDI methods into structure has the potential to increase structural reliability based on the frequency of inspections. Inspection intervals should be based on reliability analysis incorporating probability of detection, material property variation, and load/usage variation. (see guidance in paragraphs A.5.4 and A.6 of this ADS, as well as the example methodology discussed in Zhao, et al., AHS Forum 67, 2011[16]).

Diagnostics should include identification of any faults in the integrated NDI method which would reduce the probability of detection from the baseline capability. To avoid establishing integrated NDI as a mission critical function, incorporation of integrated NDI should not impede the ability to perform manual NDI methods for periods of time where the integrated NDI system is inoperable. For example, future acquisition airframes developed with integrated NDI capabilities as a requirement should continue to incorporate inspection access panels to allow for manual NDI.

Validation of integrated NDI methods as applied to a particular structure should include the effects of geometry and, as applicable, the nearby presence of built up structure, joints, or fittings. By definition, integrated NDI methods must not damage or cause any change in the characteristics of the structure or component.

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[15] Rummel, Ward D. "Recommended Practice for a Demonstration of Nondestructive Evaluation (NDE) Reliability On Aircraft Production Parts", Materials Evaluation, Volume 40, No. 9, 1982.

[16] J. Zhao, Justin Wu, M. Urban, and Douglas Tristch, "Optimization of Inspection Planning for Probabilistic Damage Tolerance Design", presented at the American Helicopter Society 67th Annual Forum, Virginia Beach, Virginia, May 1-3, 2011.

C.5.2.1 Example of the implementation of an Integrated NDI method. An integrated NDI system can be considered as an alternative to current NDI methods. It has potential to improve the probability of detection over established NDI methods. The benefits of implementing an integrated NDI system into structures includes detecting cracking in hard to access areas, eliminating complex and time-intensive procedures, reducing human factors and improving the inspection techniques from faulty detection using hand held inspection tools.

An integrated NDI system consists of a suite of sensors designed to monitor for damage of structural components. The system could take advantage of sensors already installed on the aircraft or new ones may need to be integrated. The following framework illustrates the process for selecting, qualifying and validating an integrated NDI system for Army Aviation.

1. Identify structural failure modes and failure locations:

   a. Review fielded data and design requirement to identify component(s) of interest.

   b. Review part service history and determine the potential failure modes.

   c. Perform detailed structural analysis to confirm failure mode(s) and identify critical location(s).

2. Identify and Select Integrated NDI System

   a. Specify sensor requirement through the information collected in Step 1

   b. Identify candidate sensor(s) and sensor system and perform sensor screen test or study for further down-selection

   c. Perform optimization study to determine the most suitable sensor or sensor network

   d. Perform hardware integration assessment. Analysis should determine if current on-board systems sufficiently meet the system requirements. If not, analysis should fully define integration requirements to include power requirements.

   e. Develop requirements and technical path for data transmission.

   f. Evaluation of efficiency and accuracy of the software for health monitoring of in-service structures

3. Qualification/Validation

   a. Qualification of the system for environment and electromagnetic compatibility will use the latest version of MIL-STD-810 and MIL-STD-461 as standards, respectively.

   b. Sensor Requirement – Identification of sensor requirement requires detail evaluation of structure, review of its history, and existing maintenance procedures to have better understanding of the failure mode.

   c. Capability Development/Demonstration –

i. Requires establishing probability of detection (POD) curve for each failure modes. The PODs may be established using sub-element of structures.

ii. Evaluate frequency of inspection based on worst case field spectrum, 90% POD/95% confidence flaw size, and requires damage tolerance inspection interval factor for structural reliability improvement

d. Validation - Validation of the system should demonstrate that the integrated system is at least as reliable at detecting the structural damage as the legacy techniques. If the goal of implementing the integrated system is to eliminate complex inspection procedures, the output of the integrated system should be compared against the output of the current system to determine efficacy.

i. Validation of the system as applied to particular structures should include the effect of geometry and, as applicable, the nearby presence of buildup structure, joints, or fittings. In this example, two tailboom configurations exist: one 0.040" skin, the other with 0.063" skin. Implementation of the integrated NDI system on the tailboom for crack detection would require evaluation of the skin thickness effect. Also, response waveform of the embedded sensor system may be sensitive to degree of interference fit on joints. Implementation of the integrated NDI system on joint or fittings would require evaluation of the interference fit effect.

ii. The analysis should include the identification of deficiencies that could cause performance shortfalls

e. Durability of the system should be demonstrated.

C.5.2.2 Example Inspection Interval Adjustment. Although current engineering practice commonly used in rotorcraft industry doesn't require damage tolerance based approach for fatigue life design of dynamic components, the concept of damage tolerance has been employed from time to time to establish appropriate inspection intervals for components subject to potential damage beyond their intended fatigue initiation stage. The concept of a widely adopted inspection planning methodology for in-service damage detection is depicted in Figure C-1. In this approach, a fatigue crack growth analysis is performed first. The typical material data, in terms of estimated value of equivalent initial flaw size, expected value of crack growth rate, anticipated usage and applied load are used in the analysis. The average crack growth behavior is predicted as the outcome of the analysis. To incorporate effect of NDE, a characteristic value representing inspection capability, a_{NDE}, is considered. In general, the a_{NDE} intends to represent high reliability of detection. If a POD model for the NDE exists, $a_{90/95}$ is often used. The $a_{90/95}$ represents defect size with which there is 90% of chance of detection with 95% of confidence. Otherwise, a conservative value of estimated crack size with very high detectability is used.

Similar to the concept of P-F interval, a damage growth life is defined as the amount of time that a crack growth from a_{NDE} to a pre-determined critical crack size. The damage growth life represents the window of opportunity to reliably detect damage before it reaches unstable growth and yields final failure. Due to the inherent randomness associated with damage progression, the damage growth life also fluctuates. To address the associated variability, the estimated average damage growth life is further divided by a damage tolerance inspection interval factor. The adjusted value is defined as the inspection interval. In theory, the first inspection should be started at time corresponding to a_{NDE}. In practice, the inspection begins as the induction of the inspection plan. Some applications require quarter life or half life as the time of performing the first inspection.

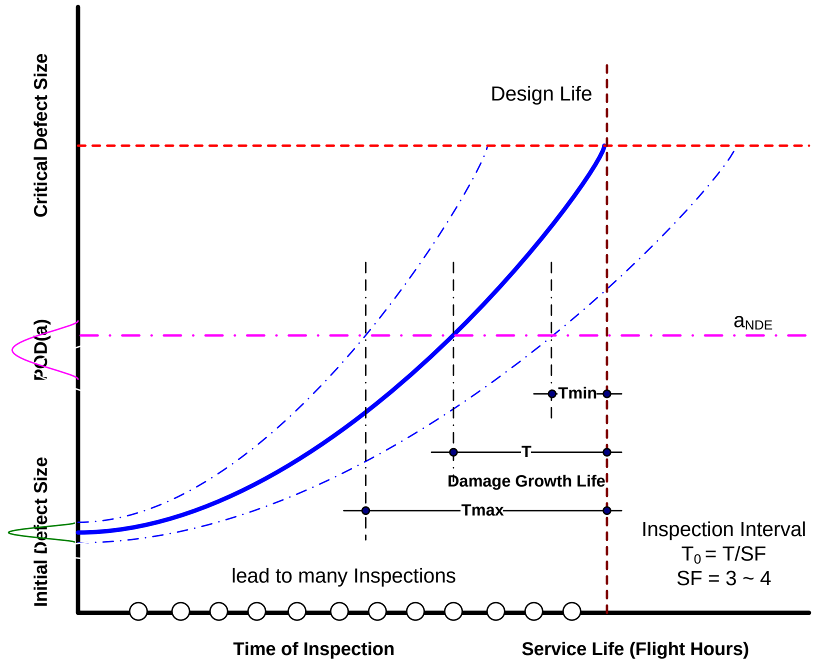

FIGURE C-1. Current approach of determining inspection interval

Although the outlined approach is well received in the industry for the purpose of inspection planning, there are several folders of drawbacks. The approach is often based on average behavior of fatigue crack growth, which is unconservative. While a damage tolerance inspection interval factor is employed to make further adjustment, its value may not be fully justified but is often selected to add conservatism. In addition, there is no justification to use a single characteristics value representing the capability of an NDI. Therefore, the lack of rigorous statistics to address uncertainties in damage progression and inspection capability limits the applicability and creditability of the current approach. In general, the inspection plan obtained from the aforementioned approach results in many unnecessary inspections.

Damage tolerance inspection interval factors to be employed in practice depend on allowable probability of structural element failure and the NDI POD. For example, a 90% POD would require six inspections to maintain six nines reliability in critical structural elements (designed for slow crack growth). It should also be understood that the allowable probability of structural element failure depends on criticality of the failure mode. For example, six nines reliability may be retained in redundant fail-safe structure with less reliability for each element failure mode. Damage tolerance inspection interval factors employed for fail safe failure modes typically range from 3 to 4, but substantiation via reliability analysis should consider changes in load path which would occur as a result of the various element failure modes.

Due to the aforementioned shortcomings for the current approach to determine inspection interval, a technical approach that addresses the inherent scatter of damage progression and incorporates the reliability model of specific inspection is highly desirable. As depicted in Figure C-2, there are two major aspects of uncertainty involved in risk assessment of damage inspection, namely the scatter of damage progression (as a function of time) and the variability associated with inspection capability. To establish an efficient inspection plan, the overlap between the distributions representing these two

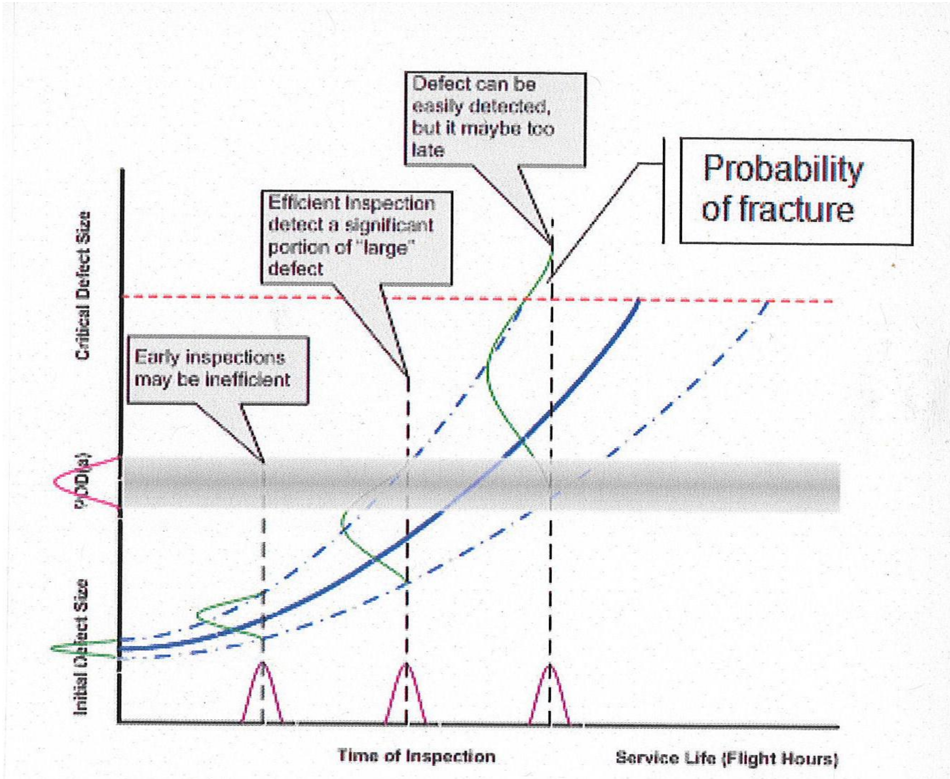

FIGURE C-2. Effect of inspection time on risk reduction

controlling factors should be maximized while the area of left tail of damage size crossing the critical crack size should be tightly limited to ensure meeting desirable level of reliability. Therefore, the effectiveness of inspection depends on capability of inspection and time of inspections. A late inspection would have great chance to detect damage, but risk of failure is high, while an early inspection may not capture damage growth in structural components.

To achieve the most efficient inspection within target reliability constraints, an ideal inspection plan should yield the minimum number of inspections while the underlying risk of failure resulting from misdetection doesn't exceed the maximum acceptable risk. Therefore, there is a window of opportunity to conduct inspections to achieve optimal solutions.

As a further elaboration of the previous discussion, a technical approach which overcomes the aforementioned shortcomings associated with the current approach is proposed and further discussed herein. The concept of the new approach is mainly based on a risk-based optimization to determine the best timings to perform inspections while satisfying the underlying reliability constraint and other CBM logistic requirements. Figure C-3 contains a notional sketch illustrating the effect of inspection and related repair / replacement on alleviation of damage progression and associated risk management. As depicted in the figure, the damage progression can be effectively reduced through a well planned inspection scheme. Detection of excessive damage progression triggers a CBM decision in repairing or replacing damaged components if the detected damage exceeds a threshold. Therefore, the anticipated risk reduction can be achieved through effectively detecting premature damage during the inspections and restoring desired structural integrity by fixing or removing damaged components.

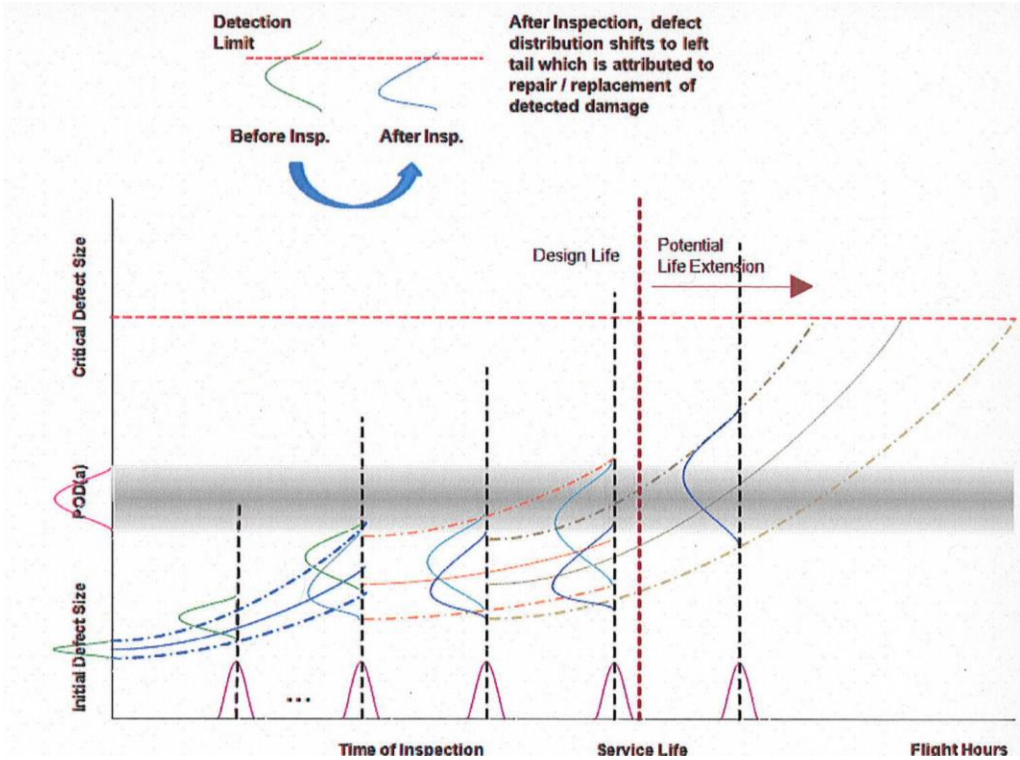

FIGURE C-3. Illustration of damage progression and effect of inspections

During repair or replacement, the damage in the structural component will be reduced or removed. The process essentially removes a significant portion of right tail of the crack size distribution obtained prior to the repair which results in the left shift of statistical distribution of damage population. This effect is illustrated in Figure C-3.

C.5.3 Load monitoring. For certain critical components and structural elements for which traditional safe-life fatigue methodologies or scheduled application of NDI methods are not able to provide cost-effective retirement intervals, calibrated strain-gage bridges and other load sensors may be considered as an alternative to component redesign. Use of actual service loads measured in the field removes pilot-induced loads/usage uncertainties from traditional application of composite worst case design usage and top of scatter flight test loads.

It is recommended that load monitoring be considered for non-rotating components and airframe structural elements. Although it is possible to incorporate load monitoring for rotating components, there are a number of technical challenges which are likely to increase the cost of implementing a robust load monitoring system.

Diagnostics tools for use with load monitoring should incorporate historical trending of maximum, minimum, vibratory, and steady loads for certain common steady-state flight conditions in consistent density altitude ranges. Typically, flight conditions of primary interest may include hover, forward flight, climbs, and steady turns. Sensitivity analysis can be used to identify additional flight conditions that may have a significant impact on fatigue reliability. Automated checking of trends against predetermined thresholds could provide notices to maintainers. Automatic zeroing procedures should be considered, as well as a generous number of backup gages.

Retirement methods based on load monitoring should be documented in detail in the CBM Management Plan. Reliability analysis of the resulting retirement scheme should be based on statistical evaluation of fatigue strength. (see section A.6.3 of this ADS for additional reliability guidance).

C.5.4 Load estimation. Load estimation, which is also known as "virtual loads modeling", is a method of processing aircraft state and control data per rotor cycle and estimating maximum, minimum, vibratory, and steady loads. If possible, the variance in each aspect of load should also be estimated. Various models may be formulated, including purely phenomenological, purely statistical, empirical, or neural-network based. Model validation should include comparisons to available flight test loads per rotor cycle. However, validation of statistical, empirical, or neural-network models should not rely on the same test data used to develop, or train, the model.

Diagnostics tools for load estimation should incorporate diagnosis of the repeatability and accuracy of the reported state and control data used by the model. In addition, exceedances of the tested state and controls envelope used as a basis for the load estimation models should be indicated to allow for predetermined special maintenance procedures, as appropriate.

Retirement methods based on load estimation should be explained in detail in the CBM Management Plan. Reliability analysis of the resulting retirement scheme should be based on statistical evaluation of load estimation and fatigue strength.

C.5.5 Limitations on use of vibration measurements for structural health monitoring. Vibration measurements, including processed signals such as vibration based "condition indicators" or "health indicators," may provide an indication of structural damage, such as rod end bearing free play and airframe cracks. However, the vibration frequencies and load magnitudes measured in airframe structure during flight are directly related to applied loads. The most appropriate use of vibration measurements as an indication of structural deterioration may be to monitor changes in natural frequencies due to stiffness changes. This approach would require periodic dynamic inputs and the measurement of frequency response. However, this approach would not be suitable for structures which exhibit an immeasurable change in dynamic response prior to primary structural load paths being severed, including non-redundant structure and any structure prone to wide-spread fatigue damage. Partially through cracks will not produce a significant change in structural load paths or stiffness. Hence, vibration measurement may not be sensitive to structural deterioration and are considered a poor indicator of structural health for airframe structure. Therefore, when any such vibratory indication of structural damage is provided, the structure near the vibratory indication should be inspected using established NDI methods. Prior to burdening maintenance personnel with required but potentially unnecessary inspections, the physics of structural health should be clearly linked to load path and vibration indication. Vibration measurements are not considered a viable approach for achieving airworthiness credits for use in place of scheduled airframe structural inspections.

APPENDIX D

MINIMAL GUIDANCE FOR DETERMINING CIS/HIS FOR PROPULSION SYSTEMS

D.1 SCOPE

This Appendix provides guidance for the development and testing of all Condition Indicators (CIs) and Health Indicators (HIs) used in the CBM system for Propulsion Systems. 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.

D.2 REFERENCES AND APPLICABLE DOCUMENTS

D.2.1 References.

OTHER	
Girdhar, Paresh and Cornelius Scheffer.	*Practical Machinery Vibration Analysis and Predictive Maintenance*, p. 112. Elsevier, 2004
Vachtsevanos, G., F.L. Lewis, M. Roemer, A. Hess, and B. Wu	*Intelligent Fault Diagnosis and Prognosis for Engineering Systems*. Wiley & Sons: New York, 2006.

(Copies of these documents are available from sources as noted.)

SOCIETY OF AUTOMOTIVE ENGINEERS (SAE) INTERNATIONAL	
Society of Automotive Engineers Aerospace. Aerospace Recommended Practice 5783.	Health and Usage Monitoring Metrics Monitoring the Monitor. 19 Feb 2008.

(Copies of this document are available from http://standards.sae.org/arp5783 or contact Kerri Rohall kerrir@sae.org (724) 772-7161.)

D.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.

D.2.2.1 Government documents.

INTERNATIONAL ORGANIZATION FOR STANDARDIZATION (ISO)	
ISO 13374:2003 **NOTE:** **Part 1: 2003** **Part 2 2007** **Part 3 2012**	Condition monitoring and diagnostics of machines.

(Copies of this document are available from http://www.iso.org/iso/ catalogue_detail?csnumber=21832 or contact International Organization for Standardization ISO Central Secretariat 1, ch. de la Voie-Creuse CP 56 CH-1211 Geneva 20 Switzerland.)

MIMOSA	
MIMOSA Standard	MIMOSA Open Systems Architecture for Condition Based Maintenance, v3.2. 19 August 2011.
MIMOSA Standard	OSA CBM for Enterprise Application Integration. v 3.2, 19 August 2011.

(Copies of this document are available from http://www.mimosa.org MIMOSA, Administrative Office, 204 Marina Drive Ste 100, Tuscaloosa, AL 35406, Phone 1-949-625-8616.

DEFENSE ACQUISITION UNIVERSITY (DAU)	
US Army CBM+ Roadmap.	Revised Draft 20 July 2007.
US Army AMCOM Condition Base Maintenance (CBM) Systems Engineering Plan (SEP),	May 2008.

(Copies of this document are available from https://acc.dau.mil/cbm-guidebook or Defense Acquisition University, DAU-GLTC, 9820 Belvoir Road, Ft. Belvoir, VA 22060-5565)

D.2.3 Process Description 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)

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.

D.3 PROCESS GUIDANCE

Detailed FMECA, often completed as a part of 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 be the development of 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, to understand the presence and progression of a fault mode, the modeling of crack size propagation 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 justify 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 with *features* or characteristics that can be obtained from signal processing of sensor data and used to accurately detect the presence of fault modes. 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 prioritized such that:

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 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 positive rate, such as 10% false positives (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:

a. Accuracy: The proportion of all healthy and faulted components which were diagnosed correctly. Accuracy represents the most fundamental metric of an algorithm's performance. [Reference c]

b. Detectability: The extent to which a diagnostic measure is sensitive to the presence of a particular fault. Detectability should relate the smallest fault signature that can be identified at the prescribed false positive rate.

c. Identifiability**:** The extent to which a diagnostic measure distinguishes one fault from another that may have similar properties.

d. Separability: The extent to which a diagnostic measure discriminates between faulted and healthy populations.

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 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 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” where a single value is provided (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. 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:

Prognostic Accuracy[17]: 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:

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

where $D_i = \left| t_{pf}(i) - t_{af}(i) \right|$ 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

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

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 D-1 illustrates the fault evolution and the prognosis, the actual and predicted failure times, and the prognostic accuracy.

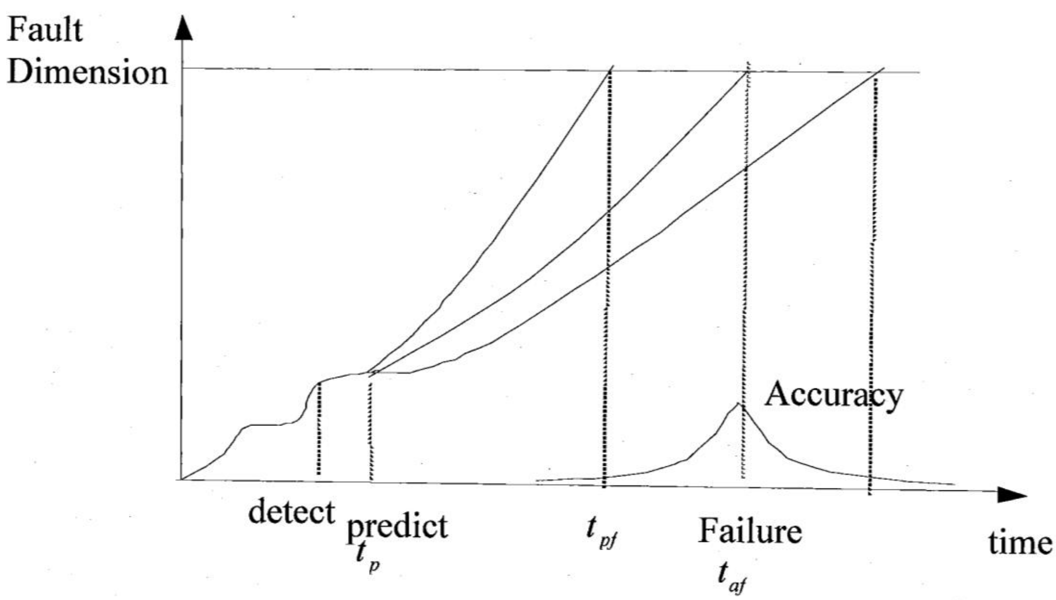

FIGURE D-1. Schematic of prognostic accuracy

Three evolution curves split from the prediction 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 prognostic accuracy 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 CIs and HIs 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 the 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.

D.4 SPECIFIC GUIDANCE

D.4.1 Condition indicator (CI) and health indicator (HI) behavior. CIs and HIs included in the CBM system for a particular Army aircraft are based on the following criteria:

a. They are identified through RCM methods including 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. All Category 2 faults should have diagnostic coverage unless the forecast rate or occurrences is less than 1 per 1 million flight hours. The diagnostic 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 Program Manager (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 Principal 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 performance.

d. The CI/HI is analyzed with respect to the feasibility of detecting the fault, the repeatability of gathering accurate fault data through the sensor, and 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 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.

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

g. 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 severity specified.

h. Redundancy between CI/HI algorithms is discouraged, but correlation between non-redundant algorithms may be used in specific cases, e.g. symptom or fault cascades. 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/HI 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.

i. 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.

j. 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 CI values for criticality and severity

ii. Bayesian Reasoning

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

iv. Fuzzy Logic Inference

k. 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 “linear trend analysis.”

D.4.2 CI and HI Confidence To ensure confidence in fault detection, CIs should be characterized by accuracy, detectability, separability, and identifiability. A class is representative of a specific failure mode or the base class of normal operation. To meet this confidence requirement, the following guidelines are recommended.

D.4.2.1 False positive rate. CI and HI based maintenance actions on the aircraft should have a false alert rate of no more than 10%. A false positive is a warning that results in the unnecessary removal of a component or other unnecessary maintenance actions.

D.4.2.2 False negative rate. CI/HIs designed to monitor flight critical failure modes must have a missed detection rate of not more than 1 in 1,000,000 occurrences of a fault.. 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.4.2.3 Fault isolation rate. Once a fault has been detected, the fault should be correctly identified 95% of the time.[18] Since a component may fail in several ways, the system should identify the particular type of failure specifically within that component. Maintenance actions resulting from HI exceedances should restore component condition with a success rate of at least 95%.

D.4.2.4 Software development. Diagnostic software should be developed, at the minimum, to the integrity level required by the system criticality assessment using RTCA DO-178B. A Functional Hazard Assessment (FHA) should determine the Design Assurance Level (DAL) of the CBM system. The Safety Assessment Report (SAR) should define the DAL and rationale. This system-determined level should be a result of the end-to-end criticality assessment.

D.4.2.5 Recommended maintenance actions. A reliable alert generation process should be developed to provide maintenance recommendations and requirements. This will provide maintenance personnel information needed to perform recommended maintenance actions, to perform troubleshooting activities to isolate a fault, or to review data and determine what maintenance actions are required.

D.4.2.6 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 prediction of future condition. Prognostics based on this characterization will be updated with usage experience.

D.4.2.7 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 (and subsequently data download intervals) for these components should be performed more frequently than for other components.

D.4.3 CI and HI Confidence Level Requirements. CIs and HIs can be classified into several confidence strata based on the ground truth evidence associated with any particular implementation. For example, levels of evidence required for maintenance credits are of course higher than levels required for enhanced maintenance. The following are examples of how to classify the different CIs and HIs respective to their ground truth data base size.

Levels 1, 2, and 3 are considered advanced and require limited to no oversight. Levels 4 and 5 are part of engineering development, and Level 6 is essentially the diagnostic graveyard.

D.4.3.1 Level 1: Verified and Validated CIs/HIs. CIs and HIs that are certified through the process outlined in Appendix I for maintenance credit are Verified and Validated. They typically

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

require 90% probability of detection and 90% confidence. The number of ground truth data samples is determined by Weibull or other similar analysis.

D.4.3.2 Level 2: Mature CIs/HIs. CIs and HIs that are supported by a minimum of 5 TP ground truth data samples and have a minimum accuracy performance of 90% are Mature and can be used to supplement standard maintenance practices in the field. Upon achieving a mature status, it is only necessary to increase the size of a CI/HI's ground truth dataset if maintenance credits are sought. The majority of CI/HIs will not need to pass beyond this level of certification. Mature CI/HIs can be used by the Project Manager to define maintenance logistics (yellow/orange thresholds) and do not exceed limits (red thresholds).

D.4.3.3 Level 3: Established CIs/HIs. CIs and HIs that are supported by a minimum of 1 TP ground truth data sample and have a minimum accuracy performance of 60% are Established. These CI/HIs can be used by the Project Manager to provide maintenance guidance that is recommended or optional. These indications are usually not associated with mandatory maintenance (red thresholds).

D.4.3.4 Level 4: Developmental CIs/HIs. CIs and HIs that are supported by a minimum of 1 TP ground truth data sample are Developmental. This level of classification has no minimum accuracy requirement and is generally not associated with field usage. It is recommended that if the Project Manager places developmental CI/HIs into the field environment, the thresholds be set arbitrarily high or not at all.

D.4.3.5 Level 5: Nascent CIs/HIs. CIs and HIs that are only supported by a physics of failure model and may be taken directly from literature for deployment into a ground station for post processing or on-board with no thresholds are Nascent. These CIs and HIs are added to the data collection system so that a ground truth dataset can be built over time.

D.4.3.6 Level 6: Retired CIs/HIs. CIs and HIs that are either obsolete, exhibit poor accuracy performance (lower than 50%), or have an unacceptable FN/FP rate as determined by the Project Manager/Engineer are Retired.

D.4.4 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 should be directly correlated to a maintenance action that can be accomplished by the maintainer and should convey the immediacy of the maintenance action. In their simplest form, HIs can be binary (i.e. "No Maintenance Required" and "Maintain Immediately"); however, in order to achieve CBM goals they should be given a range of values and meanings. (See Figure D-2) For example:

a. Green – No Maintenance required/Monitor frequently

b. Yellow – Maintain as soon as practical

c. Orange – Maintain as soon as practical/Non-flight critical maintenance

d. Red – Maintain as soon as possible

HIs may have any combination of these statuses as determined by the failure mode monitored or the redundancy of the component.

Platform: ____________________ Name: ____________________
Component: ____________________ Date: ____________________

Score Card

Color Code	Operational Capability	Maintenance Action Required	Time Horizon for Maintenance	Impact to Components	Color Determination
	Fully Functional	No Maintenance Required	Form 2410 Remaining Life	No perceptible impact to Components/ Mating Parts	Green
	Functional with Degraded Performance	Monitor Frequently	>100 Hrs	Eventual Component/ Mating Part degradation from Light Metal	Green
	Reduced Functionality	Maintain as soon as Practical (Similar to a Red Diagonal* Logbook Entry)	10 Hrs < X < 100 Hrs	Moderate Metal Contamination resulting in accelerated component/ Mating Part degradation	Yellow
	Non Critical and Non Mission Aborting Failure Mode	Non Urgent Maintenance (Similar to a Red Diagonal* Logbook Entry)	0 < X < 10 Hrs	Immediate Component/ Mating Part degradation	Orange
	Critical or Mission Aborting Failure Mode	Maintain Immediately (Similar to a Red X* Logbook Entry)	None	Heavy Metal contamination resulting in Catastrophic Potential	Red

* As defined in DA PAM 738-751

FIGURE D-2. Example color code score card

The maintenance descriptions associated with the colors yellow, orange, and red for "Maintenance Action Required" were obtained from Department of the Army Pamphlet (DA PAM) 738-751 "Functional Users Manual for the Army Maintenance Management System – Aviation" to directly correlate the CBM information to actionable maintenance information understood by the field maintainers.

The "Time Horizon to Maintenance" is associated with the estimated Remaining Useful Life (RUL). RUL is very important to the prognostics and diagnostics process for engines and transmissions. With regard to lifing, there are generally two fatigue life approaches to metal design life for critical aircraft components: "safe life" and "damage tolerant" designs. Fixed wing aircraft typically employ the "damage tolerant" design approach wherein there exists an attribute of a metal component that permits it to retain residual strength for a period of usage without repair after the component has sustained damage from specified levels of fatigue, corrosion, or accident. Damage is allowed to progress: (1) to an inspectable flaw size detectable within a specified probability and confidence; or (2) when the component fails in a safe manner due to a redundant load path or crack stopping design.

Rotorcraft and their propulsion systems are typically designed using a "safe life" approach for the metals in critical components to resist fatigue without ancillary damage and thus ensure safety. The safe life of a metal component is that usage period in flight hours when there is a low probability the strength will degrade below its design ultimate value due to fatigue cracking. The determination of the safe life of aircraft metal components depends primarily on the results of full scale fatigue tests that do not introduce other types of damage such as corrosion. The number of simulated flight hours of operational service successfully completed in the laboratory is the "test life" of the metal component. The safe life also depends on the expected distribution of failures. The distribution of failures provides the basis for factoring the test life. The factor is called the "scatter factor." The distribution of failures may be derived from past experience from similar aircraft or from the results of design development testing preceding the full scale fatigue test. The test life is divided by the scatter factor to determine the safe life. The scatter factor is supposed to account for material property and fabrication variations in the population of aircraft.

An exception to using critical metal components beyond the "safe life" fatigue limits for bearings and gears is to employ vibratory CBM monitoring if the onset of the fatigue failure mode (accompanied by surface/subsurface cracking, spalling, flaking, chipping, and pitting) is detectable utilizing a validated detection system and will not result in fatigue cracking or malfunction progressing into a failed state within 2 data download intervals of the monitoring system. An example of such an exception is Contact Fatigue in transmission gears, bearings, and shafts.

Contact fatigue generally initiates as microscopic surface and subsurface cracks which develop into surface pits subjected to alternating Hertzian stresses due to concentrated loads repeated many times during normal operation. This then leads to spalled surfaces and significant subsurface cracking if the critical metal components are left to degrade to failure. Contact fatigue is usually associated with significant operational capability remaining (time on wing / remaining useful life) from the onset of spalling depending on the speeds, loads, temperatures, and lubrication present. Bearings are especially vulnerable to this type of fatigue failure onset as they are designed using an L10 lifing approach which accepts 10% of the bearings to fail at any point prior to reaching the design life. As a result, it is accepted design and operational practice to employ chip detectors in aircraft transmissions to provide sufficient indication of impending gear, bearing, and shaft failures due to Contact Fatigue. In other words, there is some very limited "damage tolerance" accepted in the "safe life" design of the rotorcraft engines and transmissions which allows the components to operate until sufficient quantities and/or size of material is captured by the chip detectors denoting an impending failure.

It is an ongoing CBM objective to increase the sensitivity of sensors tracking contact fatigue progress that generates the metal degradation of the gears, bearings, and shafts. This, in turn, provides earlier impending failure indication than the chip detectors historically employed in aircraft engines and transmissions. As a result, the US Army is able to provide an increased time estimate to the user before failure using vibration based diagnostics. This time estimate is characterized as either a time horizon to maintenance (time on wing) or RUL.

Since each gear, bearing, and shaft contact fatigue life is unique to the engine and transmission application due to the specific loads, speeds, temperature, and lubrication associated with the designs, teardown analyses (TDAs) of engines and transmissions with onset of impending failures are necessary to achieve an accurate portrayal of how contact fatigue progresses on the bearings and gears, as well as facilitating an evaluation of vibratory CBM system diagnostic performance. Recommendations on component condition color codes based on a CBM score card may then be prepared to tie diagnostic performance to component condition as well as provide a prognostic estimate of time horizon to maintenance or RUL based on experience for each unique gear, bearing, and shaft application. This can lead to differences in acceptable damage (e.g. crack size or spall length and depth) between components from application to application. To ensure a consistent assignment of damage severity for each of the component conditions, commonly used references for grading component condition are employed. Some of these references are as follows:

ANSI/AGMA 1010-E95 - Appearance of Gear Teeth -Terminology of Wear and Failure

EPRI GS-7352 - Manual of Bearing Failures and Repair in Power Plant Equipment

Wilcoxon Research - Bearing Failure: Causes and Cures

Barden Precision Bearings - Bearing Failure: Causes and Cures

SKF - Bearing failures and their causes

FAG – Roller Bearing Damage Recognition of Damage and Bearing Inspection

After assessing the severity of the component's condition, engineers scoring the TDAs then consider the specific speeds, loads, temperatures, and lubrication around the component (along with L10 lives for bearings) to determine an estimated time horizon to maintenance or RUL. Currently the estimates for these RULs range from 10 to 100 operational hours, which is a detection improvement over current state-of-the-art chip detectors.

Until the CI maturity is verified and validated, the engineers' TDA score on the hardware condition may vary from the actual CI reading provided by the aircraft HUMS or Ground based station output.

While the vast majority of the vibratory CI thresholds for US Army rotorcraft are not mature, the incorporation of the sensors and processors on fielded aircraft allows a laboratory/research environment to exist on fleet aircraft performing their missions to facilitate data gathering and threshold maturation. The flight environment provides actual fleet data versus simulated data to be obtained and analyzed for further maturing the alert thresholds.

When components are removed from the aircraft due to legacy maintenance requirements, a confirmation of hardware condition may be made against the vibratory threshold alerts developed for CIs and HIs. When a significant number of hardware confirmations demonstrate acceptable accuracy, the CIs and HIs are adjusted to provide an alert based on data yielded from the fleet aircraft.

Since legacy maintenance practices remain in place for continued airworthiness of rotorcraft, the only concern for employing CBM using unvalidated CI/HI algorithms is that of removing components prematurely due to a potential false alert from the CBM system and incurring the additional maintenance costs. To reduce the number of components removed for CBM false alerts, designated platform working groups involving US Army rotorcraft platform managers, vibration analysts, maintainers, and hardware component experts jointly decide on whether a component should be removed for tear down analysis or wait for legacy inspections and/or chip detectors to provide an indication for removal.

This process for data gathering permits time to adjust maintenance alert levels built into the CI/HI algorithms. The vibratory CBM also increases the reliability of the overall aircraft system provided the sensor hardware reliability is not mission or flight critical and does not cause unscheduled maintenance impacting aircraft readiness. Note, also, the sensors and associated algorithms may be applied to focus on specific component failure modes versus all failure modes of complex components such as a transmissions or engines.

Once a CI alert threshold is verified and validated as mature via statistics, the engineers' scoring of hardware condition from TDAs should match that of the aircraft HUMS or ground based station display output.

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

D.5.1 Initial situation. The following section provides an example application of concepts defined in this Appendix pertaining to a fault in a gearbox.

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 D-3 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.

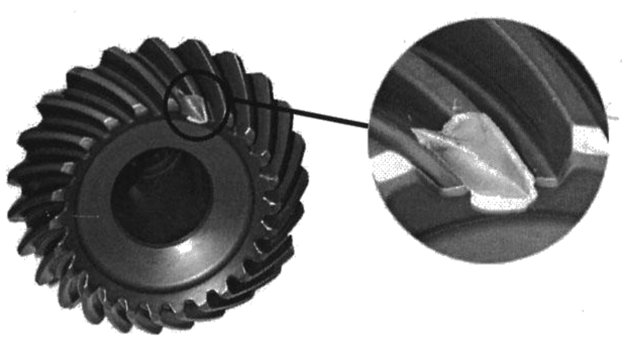

FIGURE D-3. Example of an input bevel gear fault

D.5.2 CI development process. Figures D-4, D-5, and D-6 overview the process and tools needed to develop CI/HIs

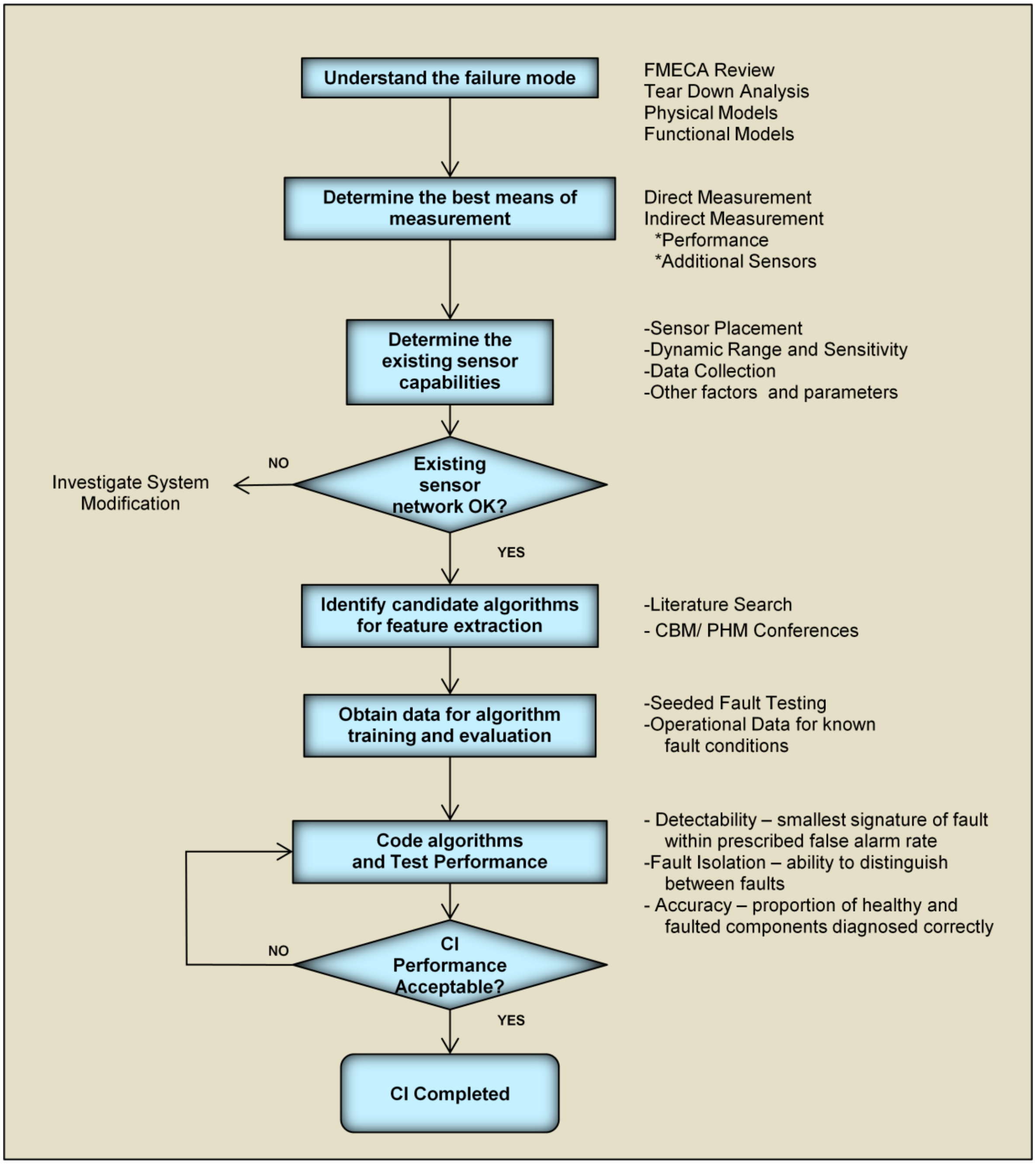

FIGURE D-4. CI development flow diagram

D.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 a safety message limiting flight or recalling specific parts.

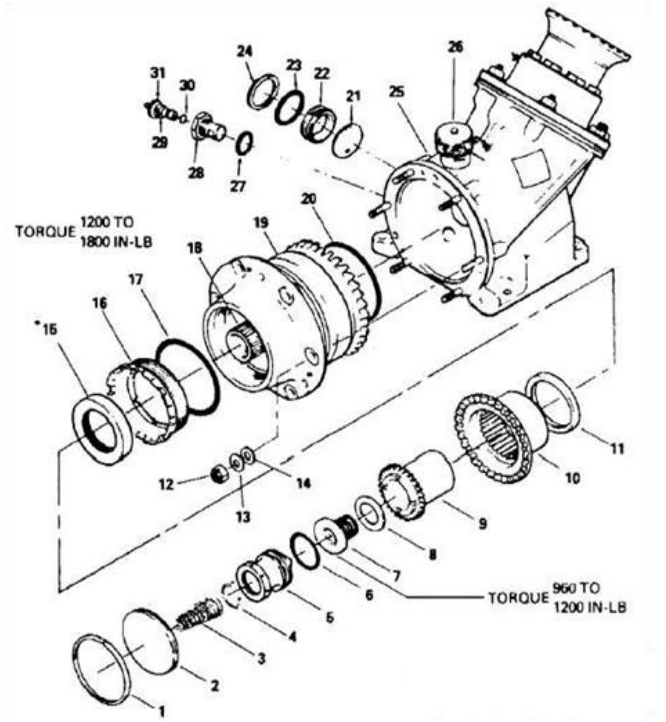

FIGURE D-5. An example of a typical schematic of intermediate gearbox used to understand physical parameters

Because the failure is related to material fatigue resulting in crack propagation, there are two major ways to detect the crack: 1) changes in the vibration sensed by an accelerometer; or, 2) 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 the 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.

D.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 frequency, 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.

D.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

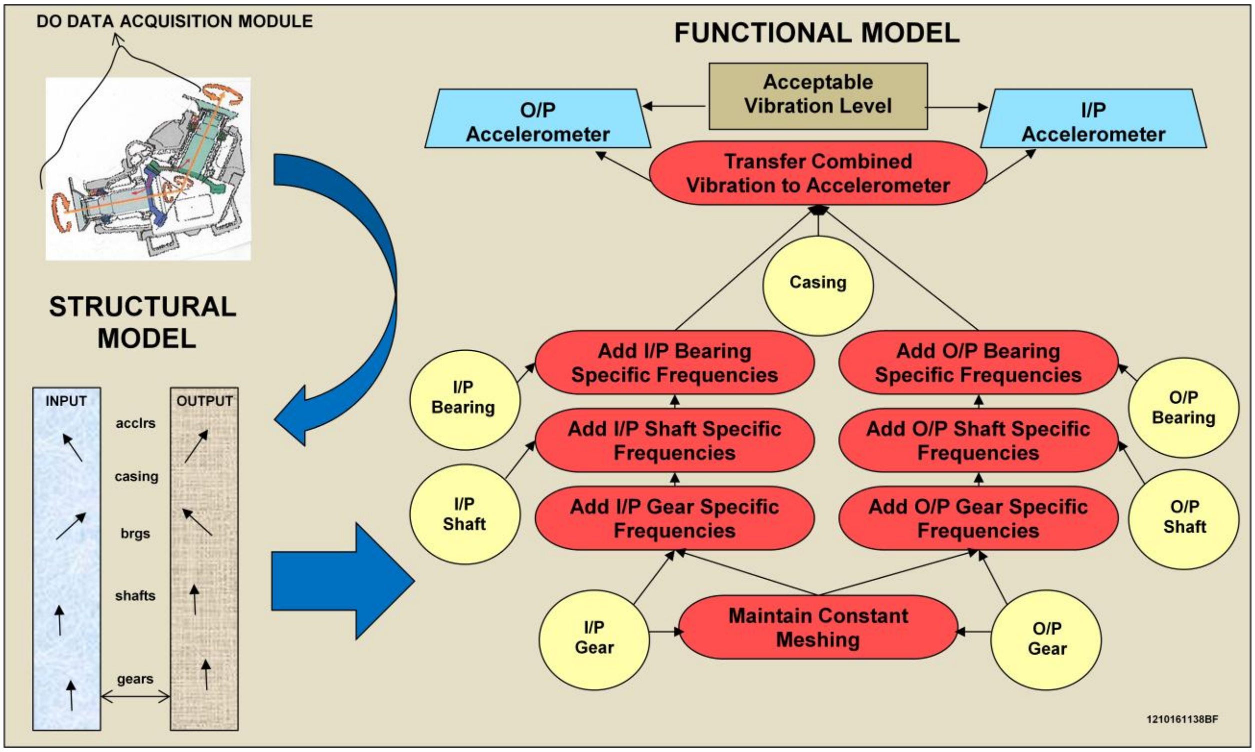

FIGURE D-6. An example of method of physical and functional modeling

D.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
b. Wavelet shrinking
c. Adaptive Thresholding

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 D-7 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.

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 E of ADS-79D lists a number of established CI algorithms. Engineering and scientific literature should also be searched for other promising feature extraction techniques.

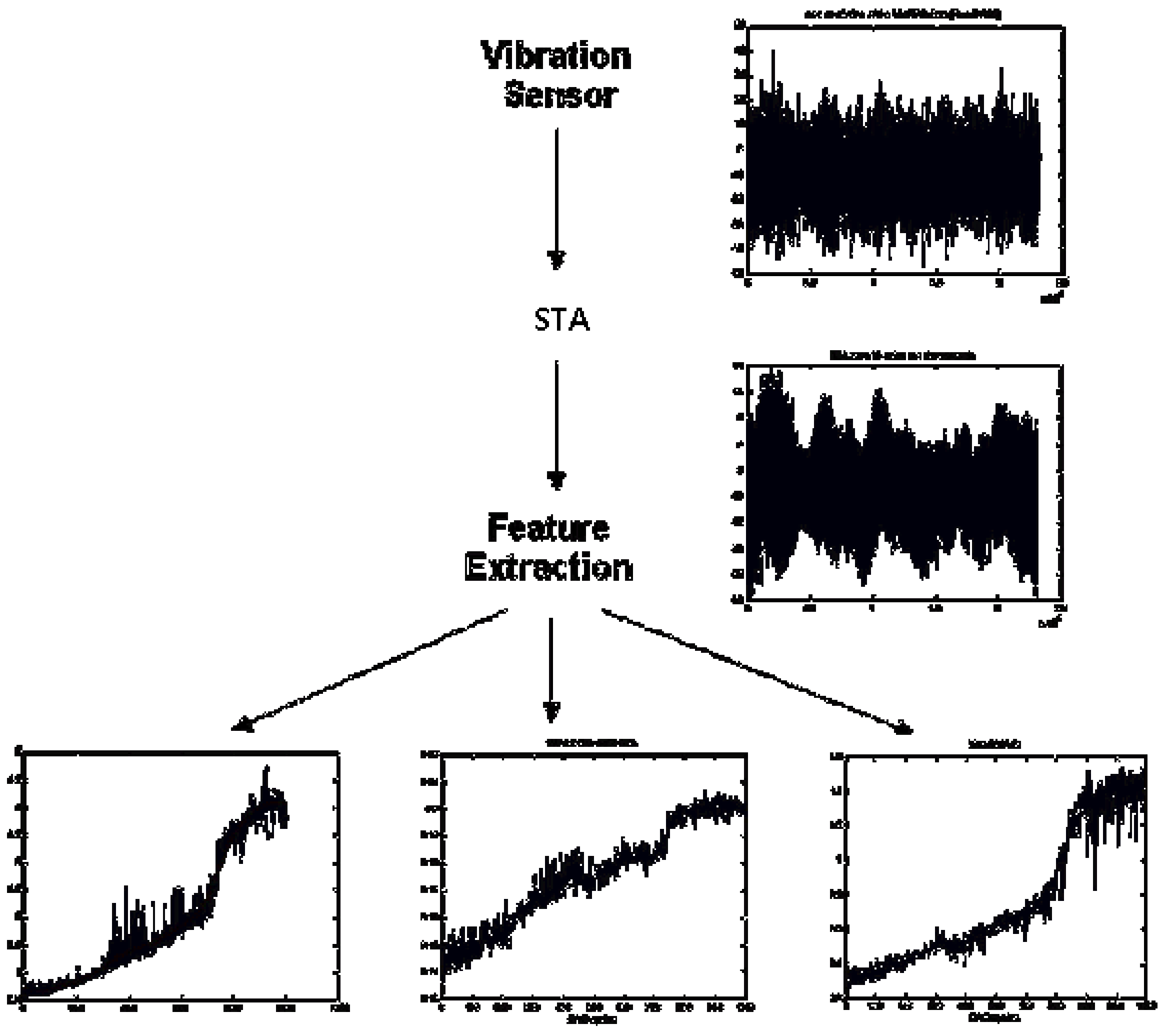

FIGURE D-7. Example of typical signal processing steps from data collection to CI comparison

D.5.2.5 Obtain data to train and 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-79D Appendix J) or 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).

Test rigs can be used to train the CI, and test articles should be chosen based solely on the expected failure modes. Seeding of faults is permissible but should concentrate on the actual failure mode regardless of the method by which the fault comes into existence. Thus, if the component is expected to always fail in a spalling failure mode, then any method can be used to create the spall. In situations where multiple methods can be used to seed a fault, the method that induces failure the fastest should be used. It is recommended that a minimum of 3 test articles be used per failure mode to train the CI.

Test rigs that are intended to measure prognostic accuracy and only enhance maintenance procedures can be tested in the same way as training diagnostic CIs. The minimum recommended number of test articles for a seeded fault prognostics test should be determined by identifying the number of expected failure mechanisms (methods to seed a fault) for the given failure mode. Thus the minimum number of recommended test articles is the number of failure mechanisms (which determine the critical failure mechanism) plus two additional tests of the critical failure mechanism.

D.5.2.6 Code the algorithms and test performance. After selecting a number of candidate algorithms for the CI, the algorithms are converted to software, typically through the use of engineering development software packages such as MatLab™. These programs are easily configured to read the data files obtained in Section 5.2.5 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 detectability 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 D-8 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.

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 D-9, 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).

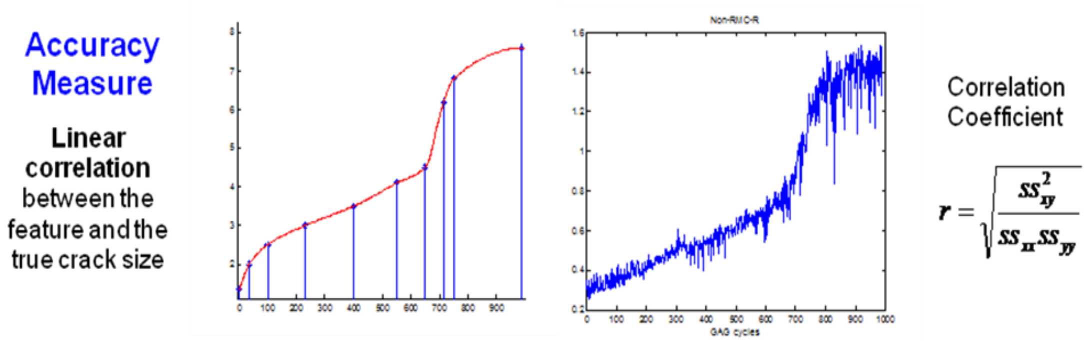

FIGURE D-8. An example correlation of fault dimension and CI value

In the bottom portion of Figure D-9, 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.

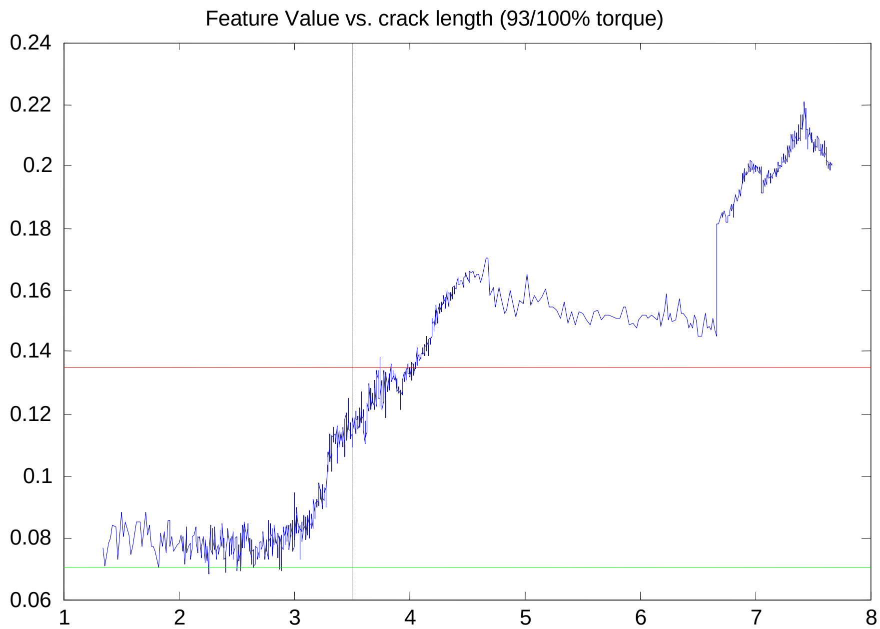

FIGURE D-9. Two examples of CI plots to compare detectability

For the purpose of this example, we assume that comparison of the CIs selected from Appendix E and the technical literature indicate that this CI has the best available performance in detectability, accuracy, and 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. Once the performance of the algorithms has been validated in this environment, they may proceed to implementation as directed by the aircraft program manager.

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

D.6.1 Initial situation. The following section provides an example of a new development aircraft which is an evolutionary design from a previous design. 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.

D.6.2 CI development process

D.6.2.1 Understand the failure mode. Reliability and Maintainability (R&M) 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.

D.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.

D.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.

D.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 E for examples of proven CIs for vibration based fault detection). Another approach is to use simulation and modeling. Figure D-10 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, detectability, separability, and identifiability

D.6.2.5 Obtain data to train and 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.

D.6.2.6 Code the algorithms and test performance. After selecting a number of candidate algorithms for the CI, the algorithms are converted to software, typically through the use of engineering development packages such as MatLab™ in the same manner as the legacy aircraft. These programs are easily configured to read the data files obtained in Section 5.2.5 and run through the algorithm calculations. The algorithms are subjected to the same analysis for accuracy, detectability, separability and identifiability.

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

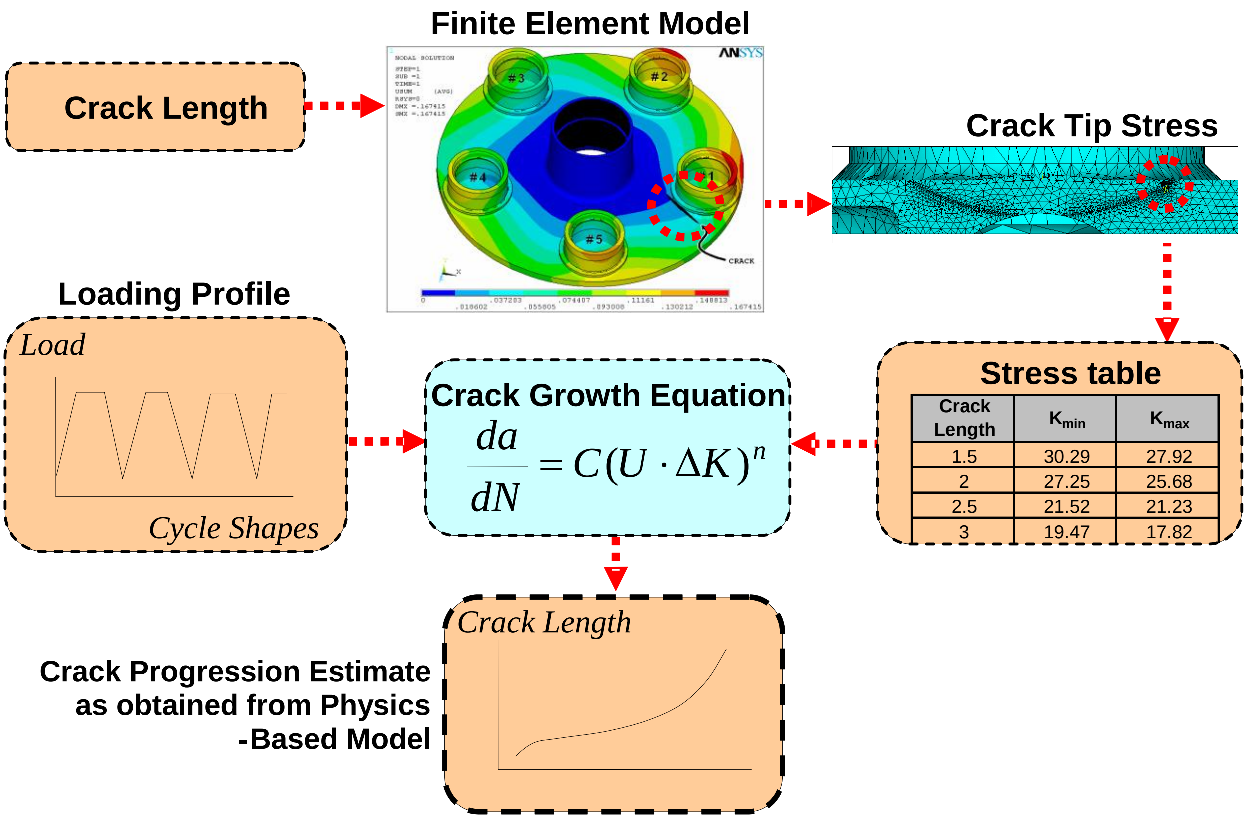

FIGURE D-10. An example of a framework for model based development of CIs

no different for CBM systems than for any Test and Evaluation (T&E) results of other major systems on the aircraft.

Section D.7 is a good example of an approach to the CI creation process for the Apache aft hanger bearings.

D.7 IN-SERVICE APPROACH EXAMPLE: 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.

D.7.1 Design Goals and Evaluation Criteria. The goal of CI development is to design one or more algorithms that produce scalar values that each strongly depend on some aspect of a component's condition. A CI's effectiveness is based on the strength of this dependence and is judged on the basis of its contribution to the ability of a health indicator (HI) to provide accurate recommendations concerning maintenance. The performance of a CI can only be evaluated in the context of its contribution to an HI.

The Modern Signal Processing Unit (MSPU) HI algorithm applies thresholds to each CI value so that each one is categorized with a status color. The HI algorithm then takes the highest status color of the most recent values for the CIs associated with a particular fault group and provides a maintenance recommendation based on this maximum fault group status. Every CI does not need to respond to every individual fault, and identification of the exact nature of the fault is not necessary, so long as the HI is able to reliably provide the correct maintenance recommendation.

For the AH-64 MSPU, two thresholds are commonly established for each CI. The lower threshold, the *caution* or *yellow* threshold, indicates that a component's functionality is reduced. Maintainers should maintain such a component as soon as practical. The higher threshold, the *exceedance* or *red* threshold, indicates that the component has a lack of functionality. Maintainers should immediately perform maintenance on such a component. The initial thresholds are set using engineering judgment and statistical analysis. As more data is collected and faulted components are found and confirmed through teardown analysis, the thresholds are revised to more accurately convey the condition of the component.

To determine the accuracy of an HI's maintenance recommendation and to evaluate a CI's contribution to it, one must first identify the correct maintenance recommendation for a number of 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. In seeded fault testing, a component with a known fault is placed on a test stand to determine how its CI values differ from those of healthy components. Often, at least one of two additional assumptions are made for the determination of HI and CI performance. The first is to assume that the vast majority of components are healthy. The second is to assume that components are healthy at the point they are installed. Neither assumption is completely accurate, but it is not feasible to tear down every component.

A good CI should provide distinct – more than 10% separation between healthy, degrading, and faulted component populations. This allows thresholds to be selected such that the known faulted components are above the healthy and degraded component thresholds and healthy components are below the degraded and faulted components.

D.7.2 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 an impulse response 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 is 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:

TABLE D-I. AH-64 aft hanger bearing properties

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

The rotational speed of the tail rotor drive shaft (ω_{shaft}) is 81.06 Hz on AH-64Ds (101%NR) and 80.25 Hz on AH-64A (100%NR). Table D-II shows the fault frequencies and harmonics for 101% NR.

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,

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

If the inner race is fixed,

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

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,

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

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.

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

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.

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

TABLE D-II. 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	1326.0
4	127.80	731.6	1150.0	1768.0

D.7.3 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 reject band centered at 685 Hz is used to exclude the gear mesh of the planetary gears in the main transmission. The frequencies that are captured by this bearing energy CI are highlighted in Table D-II.

D.7.4 MSPU High Frequency Energy CI Creation. Not all bearing faults produce detectable or separable vibrations at the fundamental bearing fault frequencies or their first few harmonics so additional CIs are needed. The simplest approach that is often effective is to select a higher frequency energy band and calculate the energy in that band. The determination of the band to use for such a CI is usually experimentally determined through seeded fault testing or from data associated with a component that has been torn down and found to be faulted. The band should maximize the separability between the known faulted components and the known/assumed healthy components. However, the band should not be so specific to the individual faulted components that it is really identifying unique features of those components rather than features that are likely to occur in other components with similar faults.

Based on the known faulted case of a corroded aft hanger bearing removed from AH-64D 01-05270 in October 2004, a CI was developed that calculates the energy in the band from 12.5 to 17.5 kHz.

D.7.5 Effectiveness. The HIs for AH-64 hanger bearings apply limits of 7 and 14 to the Bearing Energy CI and 20 and 40 to the High Frequency Energy CI. This results in an overall confirmed true positive (TP) rate of 100% and an overall confirmed true negative (TN) rate of 86.7%. Out of a total of five TDAs, the Bearing Energy CI contributes to four TPs and one false positive (FP), and the High Frequency Energy contributes to two TPs and one FP.

D.7.6 Normal and Faulted CI levels. The purpose of every CI is to distinguish between faulted and healthy 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 placed on a test stand to determine how its CI values differ from healthy components. Since it is impractical to tear down every component, the going assumption is that the vast majority of components are healthy. 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. 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 D-11 shows a portion of a spectrum from a faulted AH-64D aft hanger bearing and an average of spectra 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 spectra correspond to the fundamental BSF of 182.9 Hz and its harmonics. The average spectrum was calculated using 10 spectra (or the maximum number available) from each monitored tail number.

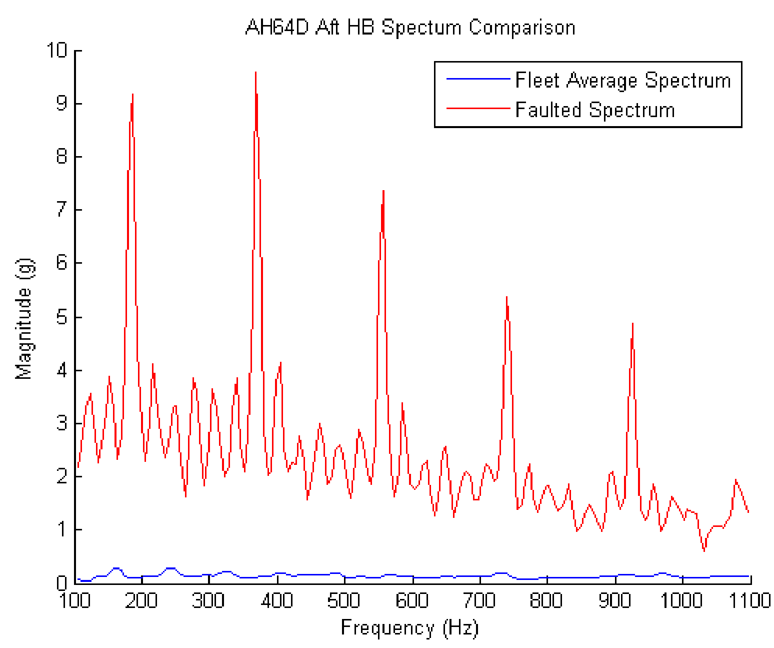

FIGURE D-11. Comparison of AH-64D aft hanger bearing faulted spectra

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 D-12).

FIGURE D-12. Damaged ball from 01-05270 aft hanger bearing

Figure D-13 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.

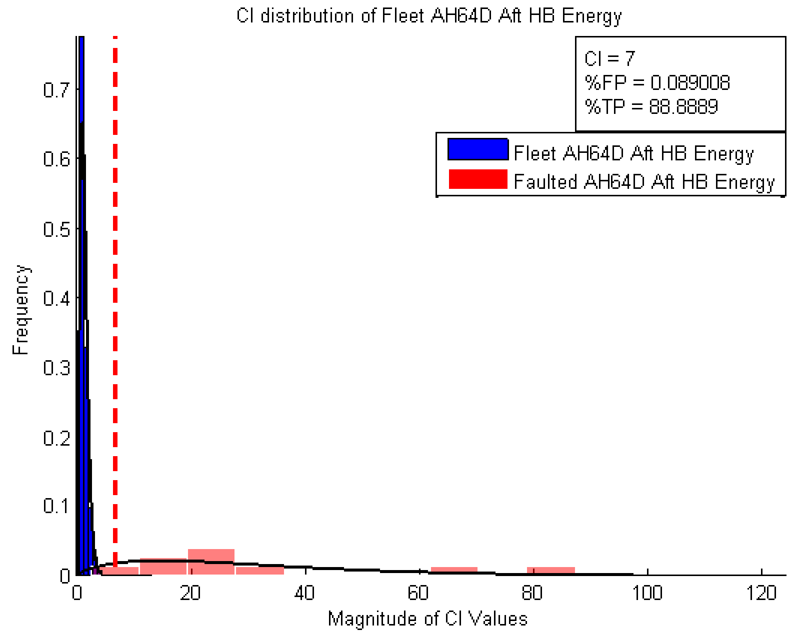

FIGURE D-13. AH-64D aft hanger bearing energy

Figures D-14, D-15, and D-16 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.

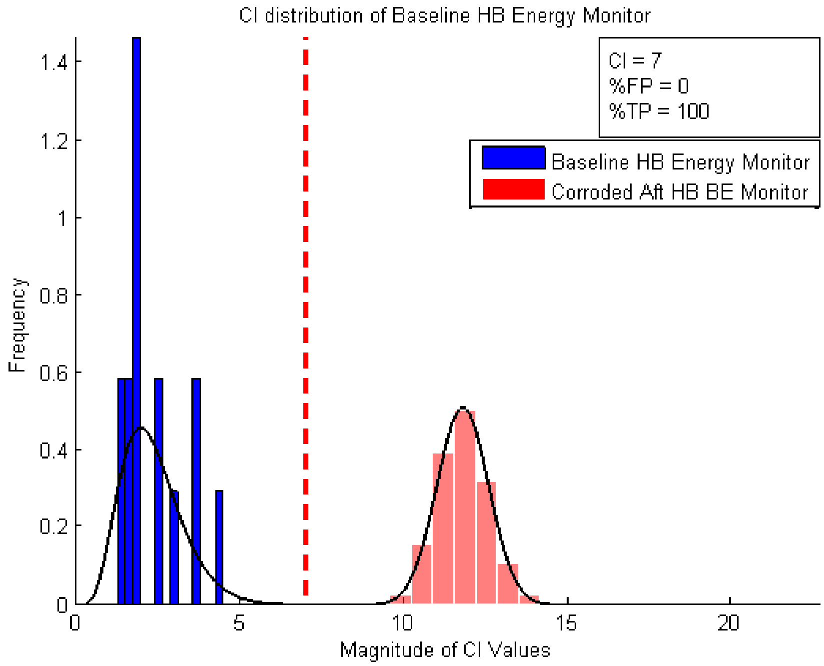

FIGURE D-14. Aft hanger bearing energy CI (saltwater corrosion fault)

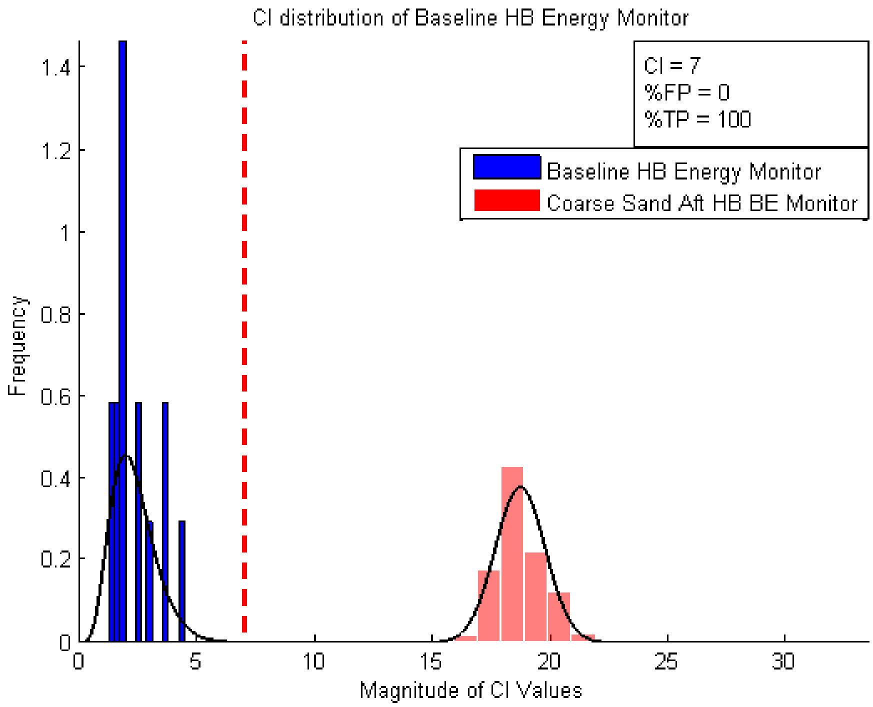

FIGURE D-15. Aft hanger bearing energy CI (coarse sand fault)

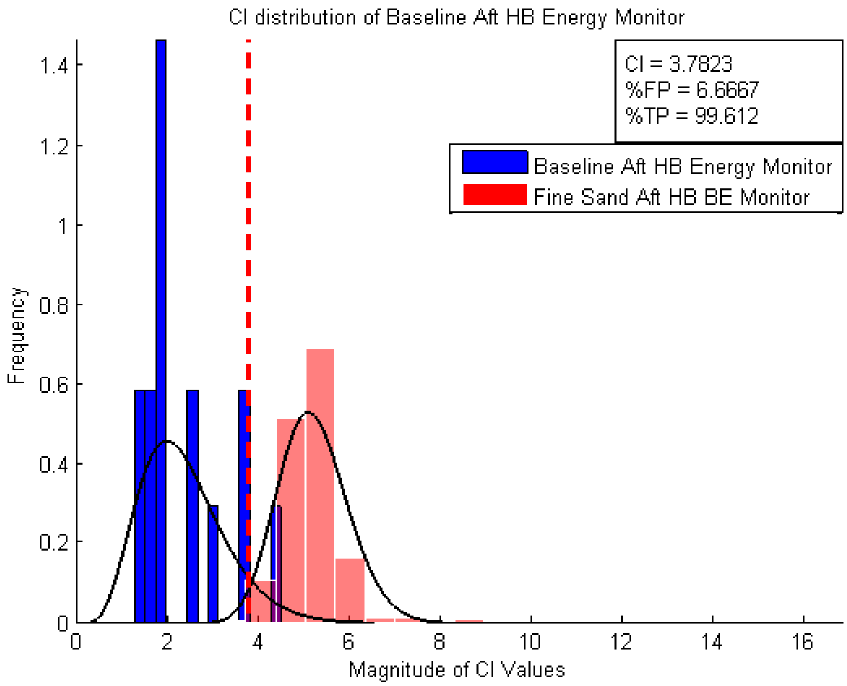

FIGURE D-16. Aft hanger bearing energy CI (fine sand fault)

D.7.7 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 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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APPENDIX E

VIBRATION BASED DIAGNOSTICS

E.1 SCOPE

This 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. Rotor smoothing and balancing procedures are discussed in Appendix F.

Vibration measurements are collected from sensors such as accelerometers at periodic intervals under specific aircraft operating conditions. This accounts for 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.

E.2 REFERENCES AND APPLICABLE DOCUMENTS

FEDERAL AVIATION ADMINISTRATION	
FAA AC 29-2C, Chg 3	Certification of Transport Category Rotorcraft. 30-September 2008.
FAA AC 27-1B	Certification of Normal Category Rotorcraft, 30 September 2008.

(Copies of these documents are available online at http://www.faa.gov/regulations_policies/)

SOCIETY OF AUTOMOTIVE ENGINEERS (SAE) INTERNATIONAL	
SAE Aerospace, Aerospace Recommended Practice ARP5783	Health and Usage Monitoring Metrics Monitoring the Monitor. 19 Feb 2008.

(Copies of this document are available from http://www.sae.org/standards/ or SAE World Headquarters, 400 Commonwealth Drive, Warrendale, PA 15096-0001 USA. Phone (US) 1-877-606-7323

OTHER	
Bracewell, R.M.	The Fourier Transform and its Applications. McGraw-Hill, 1965.
CAP 753	"Helicopter Vibration Health Monitoring" UK Civil Aviation Authority, Safety Regulation Group. August 2012. http://www.caa.co.uk
deSilva, Clarence	Control Sensors and Actuators, Prentice Hall, NJ, 1989.
Keller, J.A., R. Branhof, D. Dunaway, and P. Grabill.	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.
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. September 1985. http://oai.dtic.mil/oai/oai?verb=get Record&metadataPrefix=html&identifier=ADA173851
Ogata, K.	Discrete-Time Control Systems. Prentice Hall: Englewood Cliffs, NJ, 1987.
Roemer, M., J. Dzakowic, R. Orsagh, C. Byington, and G. Vachtsevanos.	"Validation and Verification of Prognostic and Health Management Technologies." IEEEAC paper #1344. 27 October 2004. http://standards.ieee.org/findstds/standard/1344-1995.html http://standards.ieee.org/findstds/standard/C37.118-2005.html
Zakrajsek, J.; P. Dempsey, E. Huff, H. Decker, M. Augustin, R. Safa-Bakhsh, A. Duke, and P. Grabill.	Rotorcraft Health Management Issues and Challenges. NASA/TM-2006-214022. February 2006. http://ntrs.nasa.gov/archive/nasa/casi.ntrs.nasa.gov/20060008910_2006008033.pdf

(Copies of these documents are available from sources as noted.)

E.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.

E.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).

E.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 response (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. [19]

E.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.

E.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 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.

E.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.

E.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.

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 E-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.

[19] deSilva, Clarence, *Control Sensors and Actuators*, Prentice Hall, NJ, 1989, pp. 51-53.

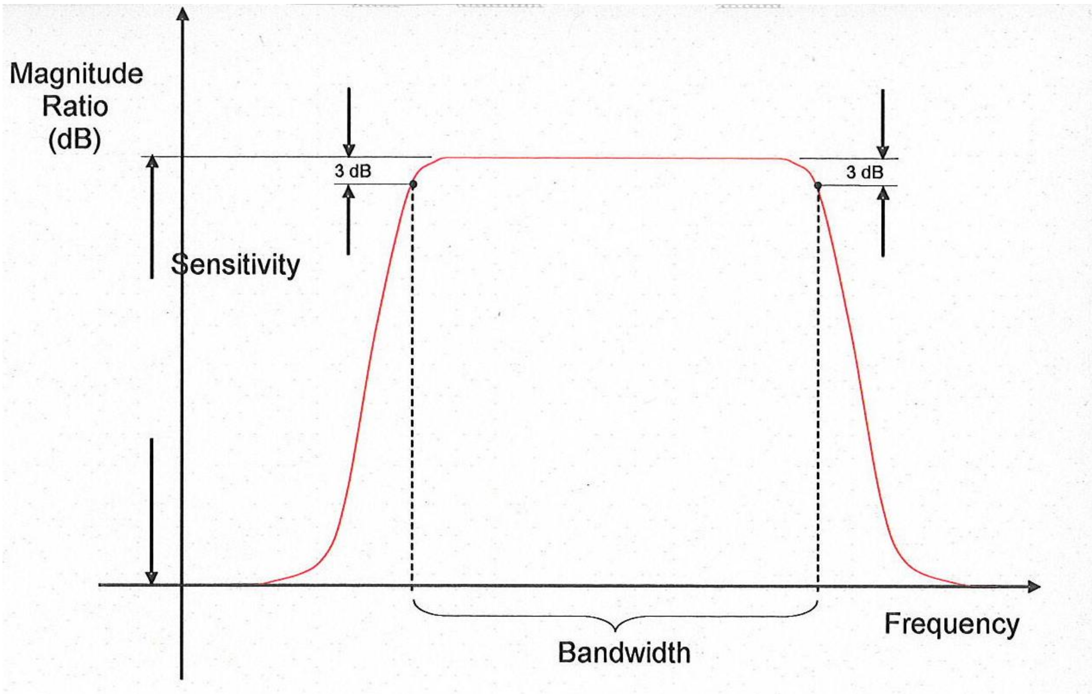

FIGURE E-1. Sensor response characteristics

E.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 (during developmental testing) to characterize the natural structural response at the mounting location. Mounting locations having resonant frequencies near defect frequencies of interest and the use of brackets should be avoided, especially when defect frequencies are narrowband and vary with rotor speed.

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

E.3.1.8 Sensor Reliability. The long term reliability of the sensor is important and information regarding its Mean Time Between Failure (MTBF) should be included in the system documentation.

E.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.

E.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. [20]

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

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 positive rate, detectability, and accuracy.

E.3.2.2 Data acquisition interval. 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.[21] 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.

E.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.

E.3.2.4 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.

E.3.2.5 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_1) 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.[22]

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 minimum required to reconstruct the continuous signal from the sampled data. In practice, the sampling frequency is frequently chosen to be 10 ω_1 to 20 ω_1.

E.3.2.6 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

[21] 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.

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

imply a rectangular window which can induce high levels of unwanted signal leakage, a redirection of the data into other spectral lines.

The rectangular window can be used in situations where the Fourier Assumptions are not violated by the signal being measured (e.g. Exactly 360° of a shaft signal average).

E.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.

E.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 which preclude execution of minimum system requirements or flight critical functions.

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.

E.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 designed CI behavior 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 frequently 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.

E.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.

E.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 (Table E-I). 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:

TABLE E-I. Shaft condition indicators

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

E.4.2 Shaft balance and rotor smoothing. Shaft balance and rotor smoothing diagnostics are basic CBM functions. Detailed discussion regarding rotor smoothing is in Appendix F.

E.4.2.1 Shaft balance 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 to reduce vibrations below determined thresholds.

E.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 (Table E-II). 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.[23,24] The following CIs are for bearings:

TABLE E-II. Bearing condition indicators

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

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

[24] 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.

E.4.4 Gear condition indicators. The following CIs (Table E-III) are laboratory proven on gear test stands operated by various commercial and Government organizations.[25]

TABLE E-III. Gear condition indicators

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

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

APPENDIX F

ROTOR TRACK AND BALANCE

F.1 SCOPE

The purpose of this Appendix is to provide methodology and guidance for the use of on-board information from the DSC to aid in the application of rotor smoothing processes.

F.2 REFERENCES AND APPLICABLE DOCUMENTS

F.3 DEFINITIONS

F.4 INTRODUCTION

The primary purpose of rotor smoothing is to reduce crew fatigue and wear and tear on the airframe and subcomponents. The vibration of interest is the rotor once per revolution (1P) vibration, which is caused by dissimilarities in the rotor blades such as subtle differences in airfoil contour, span moment, blade twist, stiffness distribution, and chord balance. Aircraft are equipped with pitch change links, trim tabs, blade wedges, balance weights, and blade sweep devices to reduce these 1P vibrations.

F.5 GENERAL GUIDANCE

In order to perform rotor smoothing, the aircraft is instrumented with devices to measure the blade height, vibration, and rotor position over multiple rotations of each shaft or rotor system of interest. Blade height is typically measured by blade trackers placed to view the rotor blades in the advancing direction, measuring the distance of each passing blade from the tracker itself. Vibration is typically measured by accelerometers or velocimeters placed in the cabin or near each shaft of the rotor system of interest, measuring the amplitude of the vibration signal. Rotor position is typically measured by a tachometer placed on each shaft or rotor of interest, measuring the phase of the vibration signal relative to a known position. This known position is typically referred to as the master position.

F.6 TECHNICAL GUIDANCE

The tachometer signal is critical in that it is used to process the time history of both the vibration and blade height measurements into an actionable set of information. Using the master position, individual rotations of the rotor or shaft of interest are identified in the tachometer signal. The timing of these individual rotations is then used to average together the multiple rotations of the vibration signal and blade track measurement. This averaging process reduces the contribution of non synchronous noise sources, such as turbulence, changes in control position or drive train vibration. The result is an averaged blade track height for each individual blade and a vibration signal synchronous with the rotor or shaft of interest. The blade track height is typically expressed in terms of the total track split, which is simply the maximum separation between blades, and referenced to the rotor blade hub. The vibration is typically expressed in terms of the 1P vibration magnitude in units of inches per second and the vibration phase angle relative to the master position. Higher harmonics of the rotor can be calculated from the averaged signal but are not typically considered for rotor smoothing operations.

The vibrations are typically measured in and out of plane of the rotor. In-plane vibrations are primarily caused by a difference of rotor blade mass, but other factors include span moments, aerodynamic drag, and induced lift. The adjustment for lateral corrections is made by adding or subtracting weight from the rotor blade or hub assembly with a secondary effect of adjustments to pitch change links. Out-of-plane vibrations are primarily caused by unequal lift of the rotor blades and corrections result in adjustments primarily to trim tab angles with a secondary effect of adjustments to pitch change links. Blade track height corrections are most strongly affected by adjustments to pitch change links.

The process of rotor smoothing consists of three steps; blade tracking, rotor balancing, and final rotor smoothing. The goal of initial blade tracking is to obtain a small blade track split, which is most important when one or more blades are newly installed on the aircraft. Blade tracking can be performed on the ground or in hover, and at one or more rotor speeds from idle to full speed. It ensures that vibration levels will not be wholly unreasonable due to large blade track differences prior to proceeding to the next step. Once acceptable conditions have been obtained in the blade tracking step, the aircraft is ready for rotor balancing. The goal of rotor balancing is to reduce any residual lateral vibrations which were not corrected in the static balancing process or are a result of minor differences in aerodynamic drag. Rotor balancing can also be performed on the ground or in hover, and at one or more rotor speeds from idle to full speed. Once acceptable conditions have been obtained in the rotor balancing step, the aircraft is ready for final rotor smoothing. The goal of final rotor smoothing is to obtain the lowest possible vibration levels while maintaining a reasonable blade track split. Vibration levels are considered in both in-plane and out-of-plane directions and across multiple aircraft operating conditions from on the ground, to in hover, to various forward flight speeds. Depending upon the primary mission of the aircraft and the opinion of the user community, vibration direction (in-plane versus out-of-plane) and operating state (hover versus forward flight) can be weighted differently to achieve the most acceptable vibration levels.

Many rotor smoothing algorithms exist. The algorithms use as input the measured vibrations and the blade track, and output a set of recommended rotor adjustments to minimize both. Examples of these algorithms are:

a. Neural Networks

b. Simultaneous Linear Equations

c. Statistical Methods

The algorithms also take into consideration practical constraints on the recommended adjustments such as:

a. Upper Limits to adjustments based upon physical constraints (i.e. maximum change in pitch link length due to number of available threads on the pitch link barrel)

b. Lower limits to adjustments based upon physical constraints (i.e. minimum change in blade weight due to the weight of an individual washer)

c. Quantizing adjustments based upon practical measurement levels (i.e. bending a trim tab to increments of ½ of a degree)

Smoothing of the main rotor system is complex due to the multiple physical differences in rotor blades which result in differences in vibration and blade track height, and is important in reducing both in and out-of-plane vibrations levels experienced by the aircrew. Smoothing of the tail rotor system is usually simplified in that only the in-plane vibration is considered; measurements of the out-of-plane vibration or tail rotor blade track are usually not acquired. Thus, tail rotor smoothing is usually referred to as tail rotor balancing. Likewise, smoothing of any driveshaft considers only in-plane vibration and is therefore referred to as shaft balancing.

The quality of the indications from the installed rotor smoothing hardware is important for good recommendations to be made. Diagnostic algorithms should be employed on board that indicate to the maintainer the quality of the measured tachometer, track, and vibration data. This should also be taken into account when making adjustment recommendations to the user, where low quality data should be given less emphasis than high quality data. During dedicated rotor smoothing flights, data quality should be made available to the pilot so that low data quality acquisitions can be re-flown prior to aircraft shut down.

Algorithms that learn from rotor adjustments made in the field should be scrutinized carefully by the Project Manager and the Airworthiness Authority. Mistakes made in the rotor smoothing process and learned by the system incorrectly will result in faulty information and thus learning algorithms must be employed carefully.

For rotor harmonics that are associated with the blade pass frequency, legacy aircraft typically use passive vibration control measures to improve crew comfort. These vibration absorbers are usually of the tuned mass/spring style applied in many high vibration environments to transportation and civil structures. DSCs can be used to ensure the proper operation of these devices or can be used to tune these devices by generating recommendations for maintainers.

New production and future aircraft may be equipped with active vibration control systems with force generators and airframe sensors. The force generators are thus used in place of the passive vibration absorbers to reduce the blade pass frequency vibration. Benefits could be achieved by linking these control systems with the DSC to measure and report the effectiveness of the control system.

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APPENDIX G

TURBOSHAFT ENGINE AND AUXILIARY POWER UNIT (APU) CONDITION BASED MAINTENANCE (CBM)

G.1 PURPOSE AND SCOPE

The purpose of this Appendix is to provide methodology and guidance to transition U.S. Army maintenance of gas turboshaft engines and auxiliary power units to condition based maintenance. This Appendix covers the use of sensors, engine usage monitoring, diagnostic and prognostic algorithms, performance trending, power assurance checks, oil and fuel monitoring, and methodology verification & validation. Further, it recommends the minimum technical requirements for a turboshaft engine health monitoring systems for condition based maintenance. Condition based health monitoring on turbofan, turbo prop, rotary, diesel, electric, and other type aircraft engines are not specifically addressed in this appendix but may be added at a later date depending on the need.

G.2 REFERENCES AND APPLICABLE DOCUMENTS

G.2.1 Standards.

ASTM INTERNATIONAL (AMERICAN SOCIETY FOR TESTING AND MATERIALS)	
ASTM D2276	Standard Test Method for Particulate Contaminant in Aviation Fuel by Line Sampling.

(Copies of these documents are available online at http://www.astm.org or from the ASTM International, 100 Barr Harbor Drive, P.O. Box C700, West Conshohocken, PA 19428-2959.)

MILITARY STANDARDS	
MIL-STD-882E	DOD Standard Practice for System Safety, 11 May 2012.

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

Radio Technical Commission for Aeronautics (RTCA)	
RTCA DO-178B	Software Considerations in Airborne Systems and Equipment Certification.

(Copies of this document are available from http://www.rtca.org/ or RTCA, Inc., 1150 18th Street, NW Suite 910, Washington, DC 20036, Tel: 202-833-9339, Fax: 202-833-9434 info@rtca.org

North Atlantic Treaty Organization Research and Technology Organization (RTO)	
RTO TR 28	Recommended Practices for Monitoring Gas Turbine Engine Life Consumption, North Atlantic Treaty Organization, Research and Technology Organization, April 2000.

(Copies of this document are available from http://ftp.rta.nato.int/public//PubFulltext/RTO/TR/RTO-TR-028///tr-028-$$toc.pdf or BP 25, 7 Rue Ancelle, F-92201 Neuilly-Sur-Seine Cedex, France)

SOCIETY OF AUTOMOTIVE ENGINEERS (SAE) INTERNATIONAL	
SAE ARP 1587B	Aircraft Gas Turbine Engine Health Management System Guide. 21 May 2007.
SAE AIR 1872A	Guide to Life Usage Monitoring and Parts Management for Aircraft Gas Turbine Engines. 29 Sept 2011.
SAE AIR 1873	Guide to Limited Engine Monitoring Systems for Aircraft Gas Turbine Engines. 5 May 1988.
SAE ARP 4754	Certification Considerations for Highly-Integrated or Complex Aircraft Systems. 21 Dec 2010.
SAE ARP 4761	Guidelines and Methods for Conducting the Safety Assessment Process on Civil Airborne Systems and Equipment. 1 Dec 1996.
SAE Aerospace Information Report AIR5113.	Legal Issues Associated with the Use of Probabilistic Design Methods. 7 June 2002.

(Copies of this document are available from http://www.sae.org/standards/ or SAE World Headquarters, 400 Commonwealth Drive, Warrendale, PA 15096-0001 USA. Phone (US) 1-877-606-7323)

TECHNICAL MANUALS	
TM 55-2835-205-23	Aviation Unit and Intermediate Maintenance, Gas Turbine Engine (Auxiliary Power Unit – APU) Model T-62T-2B, Headquarters, Dept. of the Army, March 14, 1983.

(Copies of this document are available online at http://www.armyproperty.com/tm/TB%2043-0211 or 505 E. Huron Street, Suite 202; Ann Arbor, MI 48104 2011 Crystal Drive, Suite 400; Arlington, VA 22202 DUNS Number: 829504880 / CAGE Code: 5BMR7 (703) 269-0013 / (734) 585-5061)

G.2.2 Papers

OTHER	
Gorinevsky, D., K. Dittmar, M. Dinkar, N. Emmanuel.	Model –Based Diagnostics for an Aircraft Auxiliary Power Unit, Presented at the IEEE Conference on Control Applications, Glasgow, Scotland, Sept. 18-20, 2002. http://ieeexplore.ieee.org/
Litt, J.; D. Simon, S. Garg; T-H Guo, C. Mercer, A. Behbahani, A. Bajwa, and D. Jensen.	A Survey of Intelligent Control and Health Management Technologies for Aircraft Propulsion Systems, NASA TM-2005-213622, May 2005 http://ntrs.nasa.gov/archive/nasa/casi.ntrs.nasa.gov 20050175887_2005173655.pdf
Shetty, P., D. Mylaraswamy, T. Ekambaram,	"A Hybrid Prognostic Model Formulation and Health Estimation of Auxiliary Power Units," ASME Journal of Engineering for Gas Turbines and Power, Vol. 130, March 2008
Stramiello, A., J. Moffatt, G. Kacprzynski, and J. Hoffman.	Aviation Turbine Engine Diagnostic System (ATEDS) for the CH-47 Helicopter, Presented at American Helicopter Society International - AHS International Condition Based Maintenance Specialists Meeting 2008, p 200-211, 2008 http://toc.proceedings.com/02629webtoc.pdf

Tumer, I. and A. Bajwa.	A Survey of Aircraft Engine Health Monitoring Systems, AIAA Paper No. AIAA-99-2528, presented at the 35th AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit, Los Angeles, CA, June 20-24, 1999 http://web.engr.oregonstate.edu/~itumer/publications/files/survey_final.pdf
Volponi, A. and B. Wood.	Engine Health Management for Aircraft Propulsion Systems, Pratt and Whitney, June 2006Can't find this doc – but found System Health Management: With Aerospace Applications by Volponi and Wood.

(Copies of these documents are available from sources as noted.)

G.3 DEFINITIONS

Engine Health Monitoring System (EHMS). A system for monitoring engine usage and behavior, capable of detecting and isolating deterioration and faults, predicting the remaining useful life or time until failure, and indicating needed maintenance actions.

Engine Mission Profile. A time-based description of engine operating conditions experienced in the course of a nominal mission.

Engine Monitoring System (EMS). A system and process for measuring, recording, processing, and analyzing engine parameters to assess the state of the engine.

Full Authority Digital Engine Control (FADEC). An engine control system wherein the control algorithms are implemented on a digital computer. The FADEC controls all aspects of engine performance and operation.

Health Monitoring. The technique of monitoring the output of a single and/or multiple condition indicators during operating conditions used to diagnose faulty states and predict future degradation of the equipment.

High Cycle Fatigue (HCF). Material damage caused by high cycle ($>10^5$ oscillations), low amplitude loading (e.g. vibration).

Life Usage Indicator. A damage accrual calculation based on a life usage algorithm used to estimate in real time the life consumed on a life-limited part.

Low Cycle Fatigue (LCF). Material damage caused by low cycle ($<10^5$ oscillations), large amplitude loading (e.g. engine start up-shutdown cycles).

Module Performance Analysis (MPA). A method of monitoring gas path performance parameters to infer the level of deterioration in the various engine modules and identifying the faulty components for maintenance actions.

Performance Trending. A technique of measuring engine parameters at a stable operating condition over a period of many flights and plotting the data as a function of time.

Power Assurance Analysis. A predictive analysis to determine whether an engine will be able to provide the required power within in its operating limits based on current engine performance data.

Reduced Order Algorithm (ROA). ROAs are simplified models of damage accumulation based on the more comprehensive analyses used for life calculations. These models are developed to run in real time to provide life consumption and remaining life predictions either onboard the aircraft or at the ground station after the flight.

Safe Life. The number of hours or cycles a component is expected to operate without failure under a nominal mission profile or set of mission profiles.

Usage Monitoring. A tracking method utilized for determining the life consumed or the life remaining on a life-limited part in a gas turbine engine.

G.4 DESCRIPTIONS

G.4.1 Turboshaft Engine. A turboshaft engine is a class of gas turbine engine designed to produce shaft power rather than thrust power. Thrust power is typical in turbojet and turbofan engines. The majority of turboshaft engines are used in helicopters, but they are also found in tanks, marine applications, and power generation. The GE T700 typifies a turboshaft engine and is shown in Figure G-1. This engine is used in both the H-60 and the H-64 among other aircraft. A cross section of this engine, shown in Figure G-2, illustrates the major components of a turboshaft engine. A turboshaft engine typically consists of five basic sections; a diffuser or intake, a compressor, a combustor, a turbine(s), and a nozzle or exhaust. In a turboshaft engine, the intake and exhaust are relatively simple components and require little in the way of servicing or monitoring.

The basic function of the compressor is to increase the pressure of the incoming air. This is accomplished through a combination of rotating airfoils (rotors) and non rotating airfoils (stators). The work required to operate the compressor is provided by the turbine section which extracts energy from the fluid again using a combination of rotors and stators. The T700 engine axial and centrifugal compressors generate a 21:1 pressure ratio. The compressor section delivers air to the combustor section as well as bleed air for starting, operability, and pneumatic and heating systems. As shown in Figure G-2, the gas generator turbine is upstream of the power turbine on the T700. On one shaft, the gas generator turbine extracts work from the hot expanding gases to drive the compressor section. On another shaft, the power turbine is used to generate shaft output which drives the main transmission. Depending on the design, the engine accessories may be driven either by the gas generator (T700) or by the power section. As these working sections contain components rotating at high speeds, they are typically the engine components requiring the most monitoring for both operations and maintenance.

A combustor is used with fuel injector nozzles to mix the fuel and the air. The combustor section is where the energy stored in the fuel is extracted through burning (combustion). Higher temperatures in the combustion chamber result in higher energy capable of being extracted, but they also result in lower engine life, thus the combustor is a key component to the engine life. For example, the T700 engine typically has a 5,000 hr service life but this is strongly driven by combustor temperature. Typical gas generator turbine inlet temperatures are on the order of 1297°C (2367°F) during maximum continuous operation and increases in these temperatures at higher power settings reduce the life of the turbine blades.

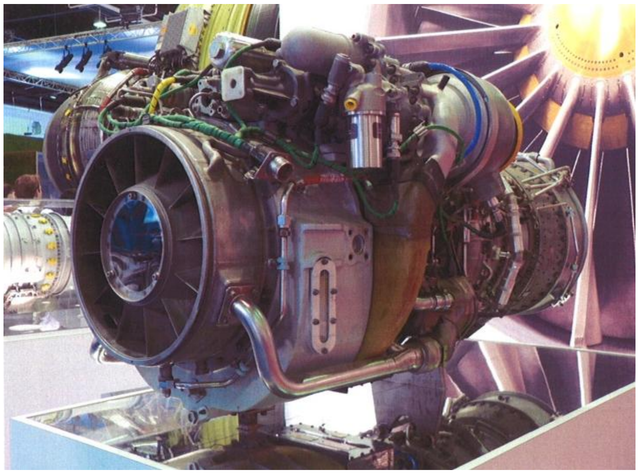

FIGURE G-1. GE T700 turboshaft engine

G.4.2 Auxiliary Power Unit (APU). An auxiliary power unit is a small turboshaft engine (See Figure G-3) designed to supply pneumatic / hydraulic / electrical power to the aircraft and to start the aircraft main engines. A typical gas turbine APU for US Army aircraft is comprised of three main sections: Power Turbine section, Reduction Gearbox Assembly, and Controls/Accessories.

The APU power turbine section is divided into the compressor and turbine assemblies.

a. The compressor provides a source of compressed air for combustion and, depending on the application, bleed air for general aircraft use. Air enters the compressor inlet, is compressed in a centrifugal impeller, the air is then discharged through a diffuser into the turbine plenum.

b. The turbine or "hot end" assembly consists of a turbine plenum, combustion chamber assembly, and turbine wheel assembly. The plenum serves as a receiver for compressor discharge air and as an enclosure for the combustion chamber. Compressed air is received by the turbine plenum and

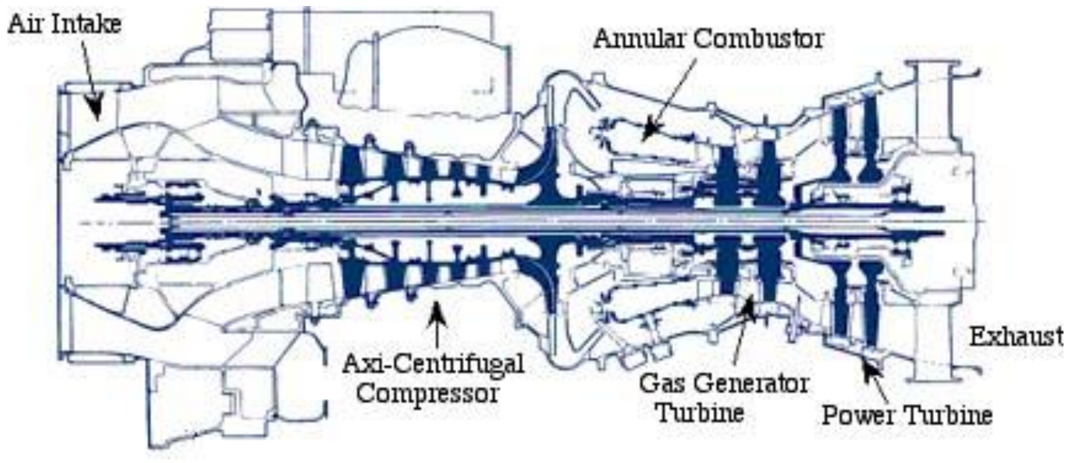

FIGURE G-2. Cross section of T700

directed through the combustion chamber, where fuel is introduced and burned. The hot gasses then flow into a radial fixed-area nozzle and a radial inflow turbine wheel. The turbine wheel drives the compressor and gearbox. After passing through the turbine wheel, the gas discharges axially through a short diffusing tail pipe section.

The reduction gearbox assembly, mounted to the output shaft, houses the reduction gear train required for use in driving APU accessories and customer-furnished equipment (i.e., APU starter, electrical generator, pneumatic pump, hydraulic pump). Additionally, the gearbox serves as an oil sump for the APU self-contained lubrication system.

Controls and accessories include those for APU operation (i.e., Electrical Sequencing Unit (ESU), fuel system, lubrication system, and ignition system). More advanced APUs include a fully-automatic electronic control system that properly sequences control of fuel and ignition during starting and operation. On ESU equipped APUs, APU speed is regulated by the ESU that directs delivery of the correct amount of fuel regardless of ambient conditions and load requirements. Overspeed protection is provided by electronic overspeed shutdown logic that is automatically actuated within safe limits at a predetermined speed.

G.5 INTRODUCTION

Turboshaft engine monitoring systems have evolved significantly over the past four decades. Simple engine monitoring systems (EMSs) that reported a few engine parameters such as oil pressure and exhaust gas temperature to the pilot have now become sophisticated engine health monitoring systems (EHMS) that record engine usage, diagnose faults, and predict the time to component failure (see Table G-I). These systems may reduce maintenance costs and improve safety when part of an engine CBM program. In the "on condition" maintenance approach, engine service is typically performed based on indications from the monitoring system rather than at a predetermined time between overhaul (TBO) interval. As long as the system reports that the engine is in good health, the engine can remain in operation until it reaches its performance limits or component life limits.[26]

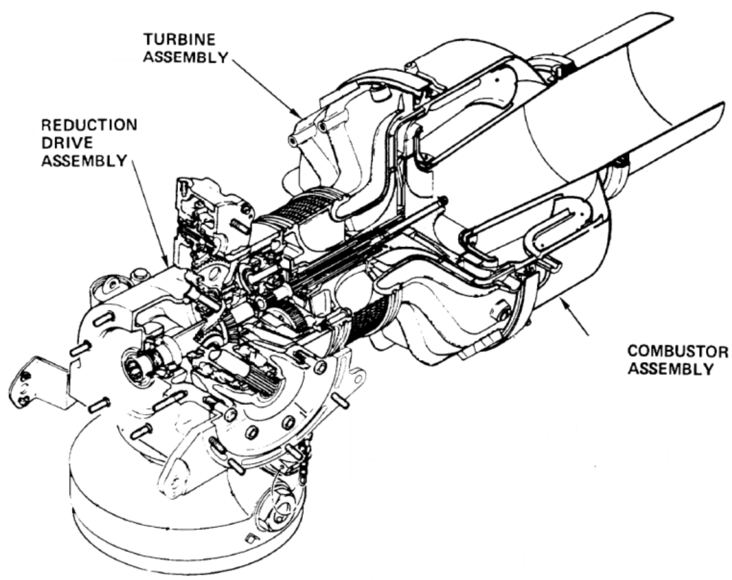

FIGURE G-3. Auxiliary power unit (reproduced from TM 55-2835-205-23[27])

[26] SAE AIR 1873 "Guide to Limited Engine Monitoring Systems for Aircraft Gas Turbine Engines", pg. 4.

[27] TM 55-2835-205-23 Technical Manual, Aviation Unit and Intermediate Maintenance, Gas Turbine Engine (Auxiliary Power Unit – APU) Model T-62T-2B, Headquarters, Dept. of the Army, March 14, 1983.

TABLE G-I EHMS Main Functions
usage monitoring
exceedance recording
remaining useful life estimation
performance trending
power assurance checks
fault detection and diagnosis
time to failure prognosis

This appendix will further define these functions and the minimal requirements for an EHMS on turboshaft engines and APUs.

G.6 GENERAL GUIDANCE

Turbine engine monitoring technology has progressed significantly over the past four decades as illustrated in Figure G-4. Engines first started incorporating exceedance monitoring of parameters such as exhaust gas, over-temperatures, and turbine over-speeds to prevent accelerated accumulation of damage and to serve as guides for maintenance action. Because the exceedances were normally displayed on cockpit gauges, this put the burden on the pilot to note and record the incidents. To reduce pilot workload and improve reporting, systems were developed to automatically record the exceedances. From there, systems progressed to collecting data in windows of time around the exceedances and then to algorithm/signal processing to look for faults (mostly in vibration and temperature). At about the same time, engine manufacturers began to develop relationships of time, temperatures, and cycling to calculate “life usage indicators” (LUIs) to replace the use of simple operating hours for life calculations. These limits (either hours or LUIs) were justified by analysis, testing, and evaluation of teardown results. Debris analysis in engine oil, to detect unexpected wear in internal components, began in the 1970s with off-aircraft spectral analysis. In the late 1990s and early 2000s the systems became laser powered to measure the size and distribution of particles in the oil. From that finding, particle distribution and failure analysis may be conducted.

The methodologies used for turbine engine health management can be grouped in two broad, complementary areas: life usage monitoring for life-limited components and condition monitoring for “on condition” maintenance. In general, life usage monitoring is concerned with tracking the changes in the engine’s operating conditions during the mission because these cyclical changes induce stress and thermal fatigue in engine components. With condition monitoring, the emphasis is on checking the engine’s behavior during operating conditions and comparing it to a baseline. Engine behavior that differs from the baseline can be indicative of normal degradation or an anomaly that is a precursor to failure. The following subsections provide a general discussion of these two areas as well as lubrication, fuel, and APU monitoring.

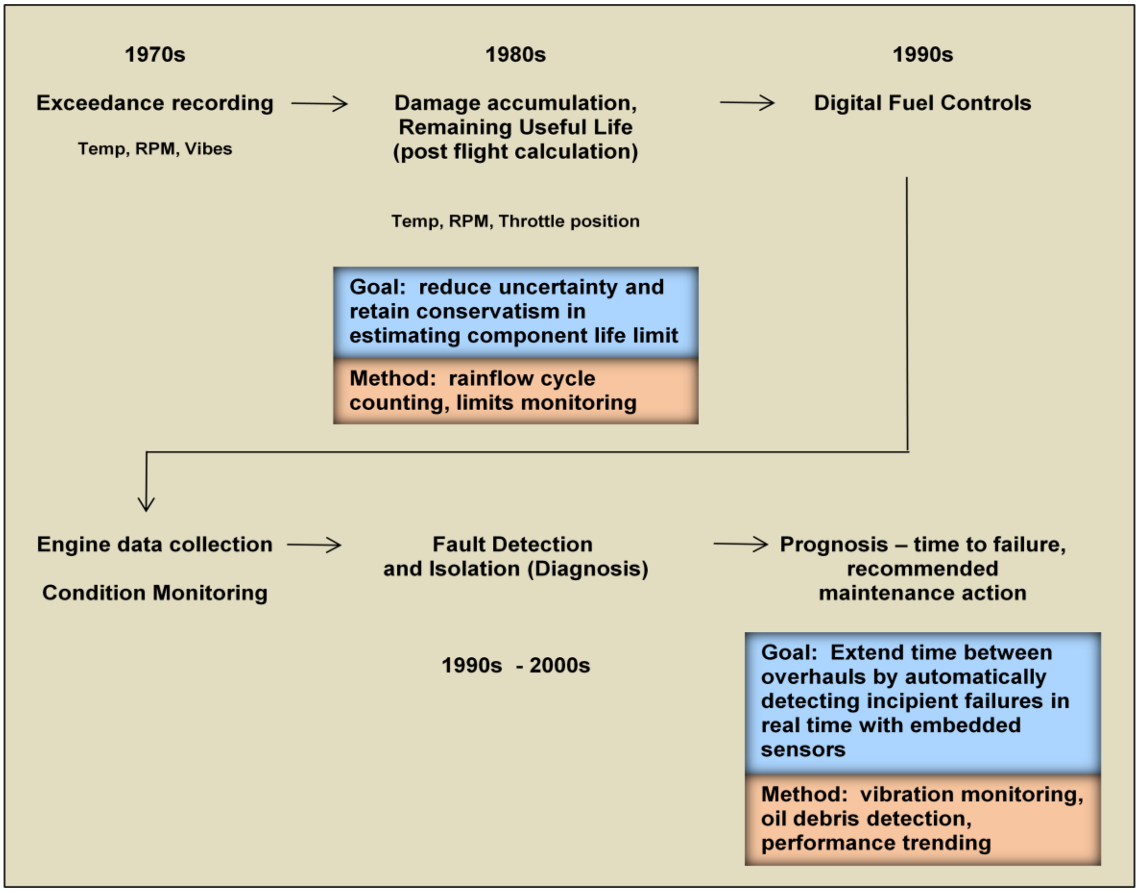

FIGURE G-4. Progression of turbine engine monitoring technology

G.6.1 Life Usage Monitoring. An engine part is life-limited if it is likely to fail through extended usage under normal design operating conditions.[28] The part's life is determined by evaluating the damage accumulated under various operating conditions. The engine's mission profile determines the frequency and duration of the various operating conditions. The traditional approach to calculating a part's life usage is to assume a standard mission profile for the engine application and sum the damage that occurs over time in normal operation. The overhaul interval is then set to ensure that the part is removed before its life has been used up. For safety purposes, some conservatism is used in setting the time between overhaul (TBO) intervals. This statistics-based approach may result in some parts being removed because they are damaged, while others, with life remaining, are removed because of the probability that they could fail.

The traditional approach may be quite suitable for commercial airline applications where mission profiles are well defined. For military applications, however, mission profiles can vary widely. If the normal mission profile for the engine is less stressful than the assumed standard profile, engine parts will be removed while they still have some remaining useful life. By monitoring the engine's actual usage, it is possible to better determine the accumulated damage and calculate the remaining useful life. This ensures life-limited components are not removed prematurely.

Another function, related to usage monitoring, is exceedance recording. As long as the engine is operated within its normal envelop, the damage accumulation can be based on the standard profile.

[28] SAE AIR 1872A Guide to Life Usage Monitoring and Parts Management for Aircraft Gas Turbine Engines, pg. 7

When the engine exceeds its normal operating limits, damage accumulates much more rapidly. Moreover, a rotor speed or temperature limit exceedance may require that the engine be removed and inspected.

The first step in the Life Usage Monitoring processes is predicting the life of a critical part. Under normal design operating conditions, failure in turbine engine life-limited parts is caused by a combination of low cycle fatigue (LCF), high cycle fatigue (HCF), thermal fatigue, and creep. Premature failure due to external factors such as foreign object damage, corrosion, material defects, etc. can be addressed through condition monitoring techniques described in section G.7.4.

LCF results from the large stresses induced by repeated cycling of the engine between low and high power operating points. The primary engine cycle is typically startup-flight-shutdown. Smaller stress cycles within this primary cycle also contribute to LCF. For example, throttle movements at higher engine speeds cause damage through centrifugal loads excursions on rotating components. In typical turboshaft engines, LCF is the predominant cause of failure.

HCF is caused by lower stresses than those associated with LCF, but at much higher frequencies. Typical causes of HCF are vibration and flutter.[29] HCF can be avoided under normal operating conditions by proper design and selection of materials. However, because military aircraft may operate under conditions conducive to foreign object damage (FOD) and inlet distortions, HCF can occur from notched blades caused by FOD and inlet distortions caused by weapons firings or rapid maneuvers.

Thermal fatigue results from stresses induced by temperature gradients across components. Stress levels and frequencies are similar to those associated with LCF.

Creep is the dimensional change in metal parts resulting from sustained loads at high temperatures. Creep may be avoided or controlled through proper design and selection of materials.

Determining the life of a part is predicated largely on predicting the effects of LCF and thermal fatigue. This requires conducting stress and heat transfer analyses using well-defined material properties. The expected stresses and environmental conditions for normal engine operation must also be used in these analyses. Bench, rig, and spin pit tests can then be used to validate and improve the initial analyses. Finally, developmental flight testing of the engine is used to verify the environmental conditions and stresses assumed in the analyses.

The safe life of a life-limited part is defined as the estimated amount of time before the first measurable crack appears or, in other words, there is low probability the material strength will degrade below its design ultimate value due to fatigue cracking. This estimation is based on the stresses derived from the stress analysis and the material properties (the empirical stress/cycle (S/N) data). The loads used in the stress analysis are developed based on an assumed mission profile or mix of mission profiles. Because of the uncertainty in material properties / stresses / mission profile / desired maintenance intervals, some conservatism is used to set service life limit lower than the safe-life.

With modern engine monitoring systems it is possible to determine usage much more accurately and reduce the uncertainty between the estimated life consumed (based on hour/cycles with a nominal mission profile) and the actual life consumed. The key is determining the actual number of cycles that the components experience during the mission. This can be done by a number of methods, with the

[29] SAE AIR 1872A Guide to Life Usage Monitoring and Parts Management for Aircraft Gas Turbine Engines, pg. 7

Rainflow cycle-counting method being the most widely accepted and successful method.[30] Using the actual cycle count history, the remaining life of the part can be estimated much more accurately than if estimating remaining life by assuming the same usage spectrum for a fleet of aircraft. Because complete life analyses require extensive use of complex, comprehensive models and data bases, reduced order models are used to calculate the life usage and remaining life in "real time", either on the aircraft or at a ground station.

G.6.2 Condition Monitoring. The focus of condition monitoring is to look for engine behaviors that are indicative of degradation or an incipient fault that could lead to engine failure or performance changes. The techniques that are employed include performance trending, fault detection and isolation (diagnosis), and prognosis. Engines monitored with these techniques are not removed at predetermined TBO intervals, but are instead allowed to operate until performance or life limits are reached.[31]

Trending is a technique of comparing the engine's operating characteristics over time to a baseline set of characteristics. Measurements are usually corrected to a set of standard or reference conditions before being compared to the baseline. Over time, trend data can reveal the rate of performance degradation in the engine. Gradual degradation represents the normal wear and tear on the engine. A rapid change in the rate of degradation can be indicative of a component fault or failure.

Fault detection methods are based on monitoring engine parameters such as temperatures/pressures along the gas path, vibration, fluids for oil debris, and fuel contamination. Throttle commands and actuator commands are also monitored. The signals are analyzed to search for anomalous behavior. If such behavior is detected, diagnostic algorithms are used to isolate the fault to a specific component. Much of the basis for the diagnostic algorithms can come from the engine's FMECA. The diagnostic methods that may be used include artificial neural networks (ANN), Bayesian Belief Networks, Genetic Algorithms, Fuzzy Logic, and Case Based Reasoning, among others.

The goal of prognosis is to determine where a fault is leading to a required maintenance action and in what time frame.[32] The time frame for further degradation is used to determine when action should be taken. Can the aircraft perform its next mission and should the component be removed before the engine's next scheduled overhaul? Before the development of automated analyses, the prognoses for failures were built into the inspection schedule. For example, the time between borescope inspections would be based on the time for an undetectable crack in a fan blade to grow to the point of failure. Accurate prognostic predictions can reduce the reliance on regular manual checks / inspections and increase the time between scheduled inspections.

G.6.3 Lubrication Condition, Debris, and Filter Monitoring. Because of the high rotational speeds in typical turboshaft engines, lubrication is essential to operation. The high thermal stresses in a gas turbine engine pose challenges to maintaining effective lubrication and condition during extended operation. The operating conditions inside the engine as well as environmental factors (dust, humidity) can introduce debris into the lube oil. If the debris is large enough, it can cause premature wear in the high tolerance bearings and other surfaces of the engine, causing failure. Normal wear of bearing material, shafts, and gears within the engine can also introduce very small particulates in the oil that can change its quality, as well as serving as early indication of failure in the parts from which the material came. As a result, turboshaft EMS/EMHS systems should also monitor the condition of the oil filter to

[30] SAE AIR 1872A Guide to Life Usage Monitoring and Parts Management for Aircraft Gas Turbine Engines, pg. 14

[31] SAE AIR 1873 Guide to Engine Monitoring Systems for Aircraft Gas Turbine Engines, pg. 4

[32] SAE ARP 1587B, Aircraft Gas Turbine Engine Health Management System Guide, pg. 9

assess the level of filter blockage and prognosticate the need for filter replacement / additional mechanical system maintenance. For guidance on lubrication condition, debris, and filter monitoring refer to Appendix M.

G.6.4 Oil Level Monitoring. Turboshaft EMS/EMHS systems should automatically monitor oil levels and prognosticate the need for oil servicing when limits on consumption rates are exceeded. The monitoring system should be capable of providing accurate readings during normal vehicle operation so as not to provide maneuver driven or inaccurate oil level/consumption indications.

G.6.5 Oil Pressure Monitoring. Turboshaft EMS/EMHS systems should monitor oil pressure and indicate / prognosticate exceedance trends of maximum/minimum limits during both steady state and transient operations.

G.6.6 Fuel Contamination Monitoring. Turboshaft engines have complex and sensitive fuel control systems and a series of high pressure fuel lines and nozzles that are sensitive to fuel contamination. EMS and EMHS should incorporate means to identify the faults caused by fuel contamination (typically shown by erratic fuel pressure, erratic fuel flow rate, or measured turbine temperatures). Direct tests for fuel contamination through in-line sensors similar to oil sensors are potentially viable, but such sensors require validation to standards such as ASTM D2276. Fuel contamination can arise from the presence of water, particulates, or microbiological organisms. Current practice is to subject fuel to extensive filtration and storage processes that eliminate the presence of contaminants. As the technology for sensors advances, the use of sensors to detect fuel contamination at the aircraft may be effective as a last line of defense. FMECA and maintenance history analysis should be used to assess the cost and benefit for using fuel contamination sensors as part of a CBM program for engines.

G.6.7 Fuel Filter Monitoring. The condition of the fuel filter should be monitored on turboshaft engines to assess level of filter blockage, fuel flow, and fuel actuator capability to prognosticate the need for filter replacement / fuel system maintenance.

G.6.8 Fuel Pressure Monitoring. Fuel Pressure should be monitored on turboshaft engines to indicate / prognosticate trends towards exceedance of maximum/minimum limits during both steady state and transient operations.

G.6.9 Auxiliary Power Unit Monitoring. Diagnostic algorithms for APUs have different requirements compared to bigger turbine engines. Especially in older APUs, the instrumentation selected is typically based on the control system requirements and not for health monitoring purposes[33]. Thus, the health monitoring and diagnostic algorithms have only limited data with which to work. Typically monitored parameters include speed, exhaust gas temperature, oil temperature, and discrete aircraft commands such as APU start/stop and main engine start (MES).

Performance trending can be used to assess the health of an APU under consistent and repeatable conditions, such as MES. The monitored parameters can be compared to baseline values generated by performance models of the MES condition. The results of the comparison are then trended over time to show the degradation in performance. Sudden changes in the slope of the performance trends are indicative of faults or failures. Efforts are currently underway to develop diagnostic and prognostic

[33] Gorinevsky, D., K. Dittmar, M. Dinkar, N. Emmanuel, "Model –Based Diagnostics for an Aircraft Auxiliary Power Unit," Presented at IEEE Conference on Control Applications, Glasgow, Scotland, Sept. 18-20, 2002, pg. 2.

capabilities using more accurate APU models, particularly models of the APU start condition, combined with fault model knowledge.[34]

G.7 SPECIFIC GUIDANCE

The following subsections provide specific guidance for turbine engine monitoring systems for use in a condition based maintenance program. See Tables G-II through G-VIII.

G.7.1 Data Collection. Usage monitoring systems and condition monitoring systems both rely extensively on sensors and processors to monitor, record, and process engine parameter data. Usage monitoring systems should capture segments of raw data and employ signal conditioning or time synchronizing routines where appropriate. Segments of raw data preceding anomalies or actual faults detected should be captured and available for downloading to enhance algorithm development and validation. The following guidance on data collection is relevant to both types of systems.

G.7.1.1 Sensors. Parameter measurements are defined by engine manufacturers based on the algorithms used. Typical parameters for usage monitoring can be categorized in four groups[35] (Note: Specific guidance on oil quality and debris sensors is provided in Appendix M.)

Table G-II Engine parameters

Spool speeds, shaft speeds	Engine intake temperature
Compressor inlet temperature	Stator outlet temperature
Oil Pressure	Exhaust gas temperature
Interturbine temperature	Turbine Exit temperature
Engine intake pressure	Compressor exit pressure
Oil temperature	Fuel flow
Torque	Throttle command

Table G-III Aircraft parameters

Outside air temperature	Indicated airspeed
Pressure altitude	G load, normal acceleration
Weight on wheels indicator	

Table G-IV Discretes and events

Date and time	Starts	Temperature events
Speed events	Cycle counts	Stall events

Table G-V Configuration data

Aircraft type, aircraft variant	Engine type, engine variant
Monitoring system hardware and software version numbers	Aircraft and engine serial numbers
Engine Inlet variant (e.g. FOD Screen/Barrier Filter/Particle Separator	Exhaust variant (e.g. Standard Tailpipe/IR Suppressor)

[34] Shetty, P., D. Mylaraswamy, T. Ekambaram, "A Hybrid Prognostic Model Formulation and Health Estimation of Auxiliary Power Units," ASME Journal of Engineering for Gas Turbines and Power, Vol. 130, March 2008.

[35] RTO TR 28, "Recommended Practices for Monitoring Gas Turbine Engine Life Consumption", North Atlantic Treaty Organization, Research and Technology Organization, April 2000, pg. 8-7.

In trend analyses, parameters are monitored and examined for shifts over time or in comparison with reference levels. Both accuracy and repeatability are important in selecting the parameters to be measured. Greater measurement repeatability is desired to separate performance shifts from the data scatter. Parameters typically measured in a condition monitoring system include[36]:

Table G-VI Parameter representing thrust or power setting
Low pressure rotor speed
Engine pressure ratio
Engine pressure ratio

Table G-VII Engine trending parameters
Exhaust Gas Temperature (EGT)
Power Turbine Inlet Temperature (PTIT)
Interturbine Temperature (ITT)
Mass fuel flow
Rotor speeds

Table G-VIII Accessory load parameters
Bleed status
Power extraction
Anti-icing condition

Table G-IX Mechanical parameters	
Oil quality	Fuel Filter Health
Oil Level/Consumption	Fuel Pressure
Oil Filter Health	Vibration
Oil Pressure	Control positions
Fuel contamination	

The general guidance on sensor requirements provided in Appendix E – Vibration Based Diagnostics is equally applicable to sensors used in engine monitoring systems. This guidance covers sensor characteristics including: sensitivity, dynamic range, linearity, drift, and bandwidth. Additional requirements for engine monitoring system sensors are provided below.

To the extent possible, engine monitoring systems should use signals already available for other purposes such as engine control, cockpit control and display, or crash data recording. Further, the sensors should be calibrated in the range where data acquisition is most likely to be required.

Modern turbine engines make extensive use of full authority digital electronic control systems (FADEC) to obtain optimal engine performance. These systems include a number of sensors that are robust, well-placed, and monitored for proper functionality with built-in tests. Many of these engine control signals are categorized as “flight critical”. As such, due consideration should be given to sharing these signals for CBM purposes. Many parameters can be shared safely with EMS/EMHS devices over the aircraft data bus. However, where data latencies impact the ability of CBM algorithms to function properly, it may be necessary to tie into the analog sensor outputs directly. This is sometimes the case

[36] SAE AIR 1873 Guide to Engine Monitoring Systems for Aircraft Gas Turbine Engines, pg. 5.

with engine speed sensor signals. Great care should be taken to prevent these signals from being corrupted by the EMS/EMHS devices.

In cases where it is not feasible to directly monitor a required parameter, it may be easier to derive the parameter from other measured data using a suitable model algorithm. This approach would not be suitable, however, if the existing instrumentation does not cover the operating envelope of the monitoring system with sufficient accuracy. Typical requirements for instrumentation accuracy and resolution are provided in Table G-IX.

For fluids monitoring, multiple COTS sensors are available for monitoring oil, lubrication, hydraulic, and fuel systems. These systems have been lumped together as the sensors used are nearly identical.

TABLE G-X Typical sensor requirements[37]

Signal	Accuracy	Resolution
Spool speed	0.1%	0.05%
Temperature	1.8 °F	0.9 °F
Engine intake	7.2 °F	3.6 °F
Compressor exit	7.2 °F	3.6 °F
Stator outlet	7.2 °F	3.6 °F
Exhaust gas	3.6 °F	1.8 °F
Turbine blade	7.2 °F	3.6 °F
Engine intake pressure	0.290 psi	0.145 psi
Compressor exit pressure	1.45 psi	0.435 psi
Indicated airspeed	2 kts	1 kts
Pressure altitude	100 ft	50 ft
G-load	0.01 g	0.005 g

Temperature sensors for oil are pervasive, but a necessary system for monitoring the performance of engines or APUs. A rapid rise in oil temperature is usually an indicator of a component failure and typically leads to an emergency shutdown. Day to day monitoring which indicates a shift in oil temperature for a steady state operating condition may indicate impending trouble and thus is an important part of a CBM system.

Equally as pervasive are oil pressure sensors. Oil pressure sensors may be used as an indicator of a loss in oil pressure which can indicate a hose failure or some other source of leakage or blockage. when used as part of a CBM system, the pressure sensor can also provide day-to-day indications of changes in steady state oil pressure. Based on the system and types of changes observed, the pressure fluctuations can indicate impending pump failure or existing partial blockage in the system.

Oil quality condition and debris sensors are also available.

For specific descriptions and guidance on aircraft engine oil condition, oil debris, and oil filter sensors, refer to Appendix M of this document.

[37] RTO TR 28, "Recommended Practices for Monitoring Gas Turbine Engine Life Consumption", North Atlantic Treaty Organization, Research and Technology Organization, April 2000, pg. 8-8.

G.7.1.2 Data Acquisition. The sampling rate for sensors (Figure G-5) should be high enough to detect all modulations of the signal and to prevent aliasing. The sampling rate should be high enough to capture higher order harmonics (typically up to 4th order), which for turboshaft engines could mean sampling rates above 20-30KHz, depending on the fault characteristic of interest and specific geometry and rotation speeds. The sampling window (Figure G-5) should be on the order of one second at quasi-steady state conditions (e.g., in a hover or at level cruise). The sampling interval (Figure G-5), or time between samples, is dependent on the type of fault/failure mode of interest. If the sampling and subsequent signal processing detects an anomaly or a possible fault (with low/moderate confidence), the sampling interval should be adjusted to capture more data and improve the fault detection confidence.

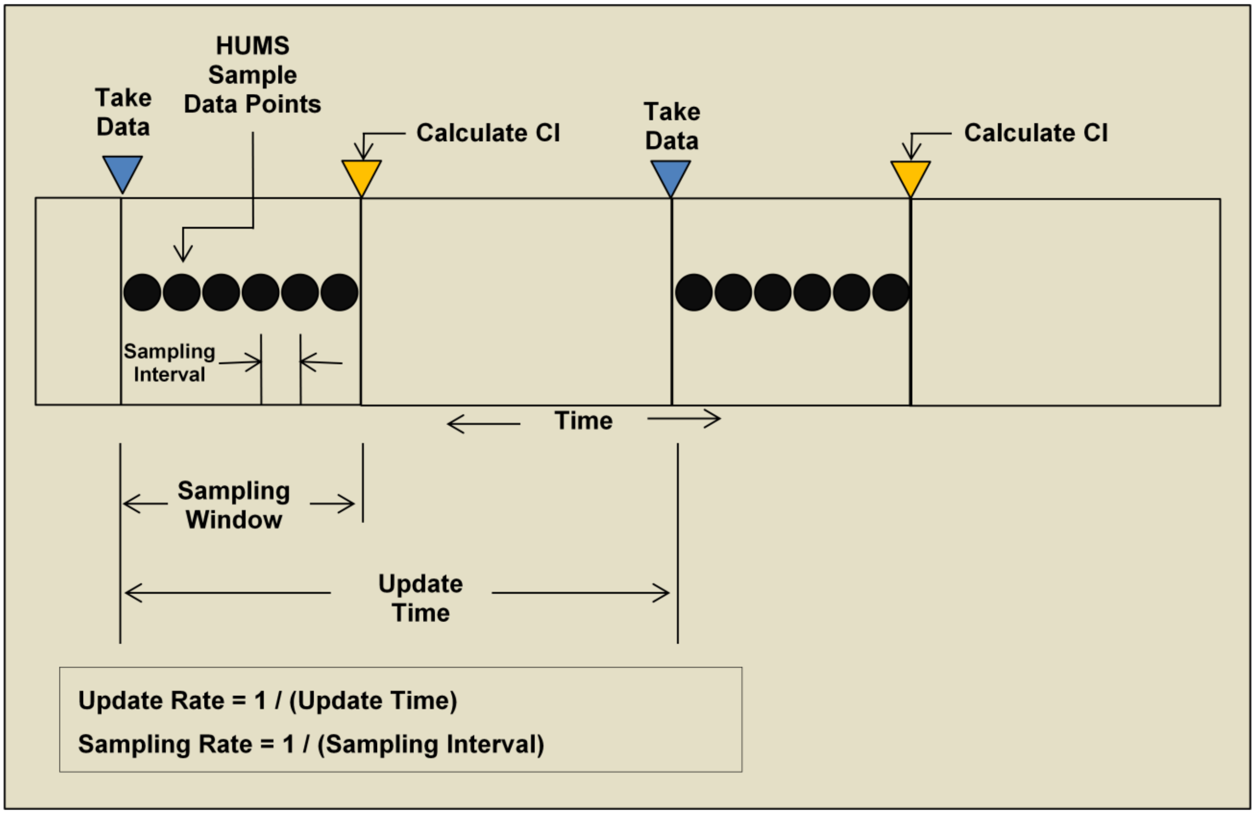

FIGURE G-5. HUMS time scale

Update Time intervals associated with data collection impact the on board software setup significantly. If the fault(s) to be monitored is expected to occur over a significant time horizon (i.e. 50 or more flight hours), the Update Time interval should be sized accordingly, along with the amount of data stored by the Digital Source Collector (DSC). Likewise, if the fault(s) is expected to occur in a short time (i.e. 1 or 2 flight hours) the Update Time Interval should be shortened appropriately and the data stored by the DSC (per acquisition) should be minimized.

How often a measurement is taken is referred to as the update rate or the iteration rate. This rate is usually determined by the component being monitored. Typical turboshaft EMS update rates for vibration are provided in Table G-X. As noted in Appendix E.2.3.4, the sampling frequency of the sensor must be high enough to capture the highest frequency of interest in the component without aliasing. Also, the duration of the measurement should be long enough to capture the lowest frequency of interest.

TABLE G-XI. Typical EMS sensor update rates [38]	
Vibration Monitored Component	Update Rate
Compressor disks	1 to 2 Hz
Turbine disks	1 to 4 Hz
Shafts	1 to 2 Hz
Turbine blades	2 to 8 Hz

The data acquisition system should have a built-in test capability to check the validity of the incoming sensor data. The system should notify maintenance personnel of any data that is suspected of being faulty. The integrity of an engine monitoring system should be ensured by having the ability to detect and flag faulty sensor data. Typical fault detection techniques for sensors include, but are not limited to:

a. Out of range – reading is outside the range physically possible for the parameter being measured
b. Rate of change – rate of change in the sensor reading exceeds what is physically possible for the parameter
c. Parameter interrelationships – cross checking between redundant sensors or between dissimilar sensors using simple models of the parameter relationships

If faulty data are detected, a process should be in place to restore the missing information or to account for the lost data. Linear interpolation between good values is one approach. Another would be to use substitute calculations which may be based on a model using other signals.

Wherever processing is carried out, whether onboard or ground-based, the onboard unit should be able to record one or more flights of raw data, depending on scheduled maintenance intervals, for independent processing. This will provide a routine system check and may be used to investigate suspicious results. Again, segments of raw data that precede anomalies or actual faults detected should be captured and available for downloading to enhance algorithm development and validation.

G.7.1.3 Data Transfer. Data transfer should be quick, easy to operate, robust, and user friendly. When damage data or flight data is transferred from the onboard system to the ground system, the system should unambiguously identify the monitored components with the associated data. The engine configuration should be included with the data base of the reduced engine model. Data transfer should be carried out with a high level of automation. Manual support or even manual transcription should be kept to a minimum. Provisions should be made to ensure data integrity.

The downloading device should be able to download and manage data from several aircraft, preferably a whole operational unit, before the operator has to return to the ground station. Data from different engines should be clearly separated and uniquely tagged to avoid misinterpretation and faulty results. In addition, data should be uniquely tagged with an engine run number / time / date.

[38] RTO TR 28, "Recommended Practices for Monitoring Gas Turbine Engine Life Consumption", North Atlantic Treaty Organization, Research and Technology Organization, April 2000, pg. 8-8.

G.7.1.4 Data Management. Engine and APU data should be carefully managed by onboard and ground systems to ensure integrity throughout its lifecycle from collection to destruction. Detailed guidance specific to proper data handling procedures is provided in Appendix L –Data Integrity.

As described in Appendix L, the degree to which data should be managed to ensure integrity is ultimately determined by the criticality of the maintenance decision derived from the data. As a result, engine and APU maintenance decisions which could result in catastrophic, hazardous/severe, or major safety issues carry a higher burden for managing the associated data against corruption / loss.

The Engine and APU CBM design should also specify procedures for recovery in the event of data loss or corruption. Again, criticality should serve as guidance for the measures taken to re-establish engine and APU maintenance in the event that data is compromised. Given the cost and effort to acquire engine and APU data, the design should anticipate long-term archival of captured data. This broader approach to retaining maintenance data will allow for later data mining to uncover long-term trends in engine and APU reliability, availability, and performance that can be used for future improvements to CBM algorithms.

G.7.2 Exceedance Recording. Any significant exceedances should be recorded when each event occurs and flagged to the ground crew. Exceedances may be used to trigger a usage assessment.

G.7.2.1 Hot Starts. In conformance with the engine design definition for out of limit conditions on start, the duration and maximum temperature reached during an improper ('hot') start should be recorded to the nearest second.

G.7.2.2 Over Torque. In conformance with the engine design definition, conditions which represent out of limit conditions for torque applied by the engine ('over-torque') should be recorded to the nearest second and should include the maximum value sensed by the torque sensor during the period of exceedance.

G.7.2.3 Overspeed. In conformance with the engine design definition, conditions which represent out of limit conditions for the compressor and power turbine speeds (typically known as Ng or Np) should be recorded to the nearest second and should include the maximum value sensed by the sensor during the period of exceedance.

G.7.2.4 Over-temperature. In conformance with the engine design definition, conditions which represent out of limit readings for temperature as measured by the installed temperature sensors should be recorded to the nearest second and should include the temperature reached at each second during the period of exceedance.

G.7.2.5 Vibration. In conformance with the engine design definition, conditions which represent out of limit readings for vibration should be recorded to the nearest second and should include the value sensed by the accelerometer during the period of exceedance. EMS/EHMS should monitor the engine vibration and dynamic response at all engine speeds and powers including steady state and transient operation throughout the environmental conditions and operating envelope of the engine. The monitoring system should indicate / prognosticate trends towards exceedance of engine vibration limits at each sensor location for the compressor, turbine, and gearbox sections. The monitoring system should automatically correlate vibrations with component degradation / damage. The monitoring system should detect unusual vibration changes while distinguishing between airframe, drivetrain, and engine induced vibrations to prevent false indications.

G.7.3 Life Usage Monitoring. A complete usage monitoring system contains the following functions:[39]

a. Data recording – acquisition of the necessary data for calculation of the component usage values (see Section G.7.1)

b. Damage calculation – calculating the life used by each component or critical area

c. Life management – organizing the life usage information to support decisions on aircraft deployment, component retirement, engine removals, and engine and spares management

The physical implementation of an engine monitoring system is usually accomplished with several basic elements:

a. The airborne system hardware and software (hardware includes any additional sensors that are not part of the engine control system)

b. Equipment for data transfer to the ground station

c. Ground station hardware and software

Usage monitoring systems rely heavily on counting the cycles and the magnitudes of the stress variations incurred through changes in the engine's operating conditions. Therefore, the EMS/EHMS should automatically capture and record, on an engine-by-engine basis, steady state and transient operating data and events indicative of structural life usage. Guidance for monitoring the primary parameters of a usage monitoring system is as follows:

G.7.3.1 Operating Hours. Engine operating hours should be recorded in a manner consistent with the OEM design specification and definition used to acquire and test the engine. Typically, operating hours begin with the engine at stabilized conditions immediately after engine start and end when the engine RPM reaches a defined value at shutdown. Operating time should be recorded to the nearest whole minute except where otherwise defined.

G.7.3.2 Start/shut Down Cycles. Start and shutdown should be recorded consistent with the specification and design definition for the engine and recorded to the nearest whole minute unless otherwise defined.

G.7.3.3 Operating Speed/Duration. Engine operating speed should be monitored and recorded at a frequency of at least 1Hz during the period of operation (between start and shut down) as defined by the engine design specification.

G.7.3.4 ITT / PTIT / EGT. ITT, PTIT, and EGT are typical parameters collected to monitor the temperature which is experienced by the turbine blades. These parameters are normally associated with turbine blade life calculations. These temperatures should be measured at an update rate of at least 1Hz during operation.

[39] RTO TR 28, "Recommended Practices for Monitoring Gas Turbine Engine Life Consumption", North Atlantic Treaty Organization, Research and Technology Organization, April 2000, pg. 7-3.

G.7.3.5 Torque. Torque is a measure of the mechanical power output from the engine and should be measured as defined in the engine design specification for the duration of engine operation at an update rate of at least 1Hz.

Ideally, each life-limited part should be monitored separately by the EMS/EHMS. A widely used method is to monitor only one or two parts on each rotating shaft and to derive the life usage on the unmonitored parts by applying relational ‘read-across’ factors. This method is practical only when the life usage relationship between parts is linear and is usually more feasible when thermal stresses are absent or negligible.

G.7.3.6 Lifing Algorithms. Lifing algorithms are used to compute in real time the damage accrued and the remaining useful life of components. These algorithms are developed from the more detailed analyses used during the design process to calculate the component lives. They are frequently referred to as reduced order models (ROM) or Life Usage Indicators. If more than one critical failure can be present, enough features should be monitored to ensure the integrity of the modeling process. The engine life model should take into account the engine operating conditions and the mechanical properties of the materials and components. The EMS/EHMS, therefore, should track the operating conditions and provide data to assess engine status to the module level while providing sufficient information to determine required maintenance actions.

The EMS/EHMS should have extensive built-in test (BIT) capability to ensure that internal computational routines are operating within their defined limits. In the event of an EMS/EHMS failure, the affected module or card should be identified and the effect of the failure on recorded data should be considered. In addition, any cycles incurred but not recorded should be input manually either in accordance with a simplified algorithm or a worst case scenario to ensure that components have not consumed more life than is recorded.

The procedures which control the manual activities and any techniques used to identify questionable data and to repair it or compensate for it should be subject to audit to provide assurance that they are adequate and satisfactorily maintained.[40] The audit should account for and include the following aspects:

a. Identification of personnel carrying out the download under various circumstances

b. Storage capacity of the airborne unit and transfer unit

c. The required frequency of downloads and the number of aircraft and flight which can be downloaded to the transfer unit without return to the ground station

d. The identification of and compensation for system failures and data loss

e. Program for regular critical reviews of the system outputs

f. Training program for operating personnel

[40] RTO TR 28, “Recommended Practices for Monitoring Gas Turbine Engine Life Consumption”, North Atlantic Treaty Organization, Research and Technology Organization, April 2000, pg. 7-13.

The consumed life should be available after each flight or at the end of the day's flight operations. The damage computation may be made available to the maintainer in real time using an onboard computer or very soon after the aircraft lands using a ground-based computer.

Prior to introduction to operational service, the life usage monitoring algorithms should be verified and validated. Verification ensures that the reduced order algorithms produce the same results that the full analyses models would produce. One method of verification is to compare the output of the reduced order model to the full analysis model used for life determination for a given set of flight profiles. The reduced model is considered equivalent to the full model when the mean values of the computed damage are the same and the maximum difference is below a defined limit.

To ensure the equivalence between the design model and the reduced-order models, all the verification procedures of the reduced-order damage model should be conducted for each material on a wide range of flight profiles. Typically there are differences between the design model and the reduced order model. First, the material data base incorporated into the reduced model is a simplified derivation of the design material data base. Second, the thermo-mechanical history of a real engine is much more erratic than the conventional flight profiles used in design.

The lifing model and the material data base should accommodate[41]:

a. Numerous small cycles of real flight

b. Very rapid loading which occurs during rotor acceleration and deceleration

c. Constant loading can lead to some concerns in real time creep and fatigue modeling

The verification of the accuracy of the structural model should be performed on a specified variety of flights to ensure the operational temperatures have been validated.

Algorithm validation requires a large database of service experience to establish accurate correlations. Initial information in support of this objective can be acquired through Accelerated Simulated Mission Endurance Tests (ASMET).

Information generated by the reduced order damage models should be kept during the whole fleet life of perhaps 30 years, considering the age of current aircraft fleets. If the reduced model is revised, then the relevant parts of the verification and validation process should be repeated. When a component is replaced during servicing by another with the same life characteristics, this need only be recorded in the life database, unless the monitoring system needs to be reinitialized for the life consumed. If a component is redesigned or a modification to other components alters the behavior of the engine, the rate of damage accumulation for the monitored parts may change. The reduced order algorithms may have to be altered, which will require repeating the verification and validation process. If the engine starts being used on a different flight profile than already experienced, the algorithms should be checked for accuracy using the new flight profile.

G.7.4 Condition Monitoring. A condition monitoring system consists of the following four functions[42]:

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[41] RTO TR 28, "Recommended Practices for Monitoring Gas Turbine Engine Life Consumption", North Atlantic Treaty Organization, Research and Technology Organization, April 2000, pg. 7-7.

[42] SAE AIR 1873 Guide to Engine Monitoring Systems for Aircraft Gas Turbine Engines, pg. 7.

a. Data acquisition

b. Data reduction

c. Data presentation

d. Interpretation of results

Condition monitoring relies on techniques such as trend analysis and anomaly detection. For trending analyses, parameters that have large errors inherent in their measurement systems should not be used as a basis for comparison with other more consistent parameters. Furthermore, parameter data should be acquired at stable conditions (steady state) which will allow shifts in parameter behavior to be more readily detected. Transients caused by changes in accessory settings or control settings can add noise or variation to the parameter trends making the detection of a true shift difficult. If data smoothing algorithms are used to improve the readability of trend data, they should not mask the presence of a true parameter shift.

G.7.4.1 Performance Monitoring. Aircraft engine reliability has steadily advanced through the history of Army aviation. Removing the engine for failure to perform in accordance with design limits, typically associated with torque, temperature, or vibration, is now more common than removal for catastrophic failure. Performance monitoring is, therefore, the first 'level' or type of condition monitoring. Engine Monitoring Systems should include the means to periodically assess engine parameters in comparison to those same parameters as measured upon engine installation in the airframe. This periodic assessment should be done in similar states of power and load for consistent comparison, and should be defined by the Using Service with input from the engine OEM. The comparison of these parameters through basic trend analysis, or other algorithms, should also be defined by the Using Service with input from the OEM. The OEM should determine what elements of performance monitoring should be included in flight and propose those elements to the Using Service aircraft program management office as part of the Condition Based Maintenance Plan for the engine. When parameters are selected for off-board monitoring, the means and validation of this off-board monitoring should also be included in the CBM Plan for the engine and validation of the monitoring should be part of the overall engine test and evaluation process.

G.7.4.1.1 Power Assurance Monitoring. Power Assurance Monitoring is a special case of performance monitoring. Historically, flight crews have conducted power assurance checks as part of pre-flight planning and mission preparation, as well as during post maintenance functional check flights. Power Assurance checks are performed to validate the expected level of torque supplied by the engine at given environmental conditions of density altitude and temperature under operational conditions. These checks are typically performed to ensure that the engines are capable of supplying sufficient torque to complete the anticipated mission. Engines equipped with EMS or EHMS capable of performing power assurance checks should automatically assess the engines for power assurance check on the first application of power to hover the aircraft, and should display a simple cockpit indication (e.g., 'green light' for acceptable power / percent power available) to the flight crew. Since this information to the pilots is critical safety information, Automatic Power Assurance Algorithms should be baselined and checked against calibrated engines. The following detailed topics should be considerations in implementing Power Assurance systems in EMS/EHMS:

G.7.4.1.2 Safety Assessment. A safety assessment should be conducted in accordance with (IAW) SAE ARP 4761 and initiate a Functional Hazard Assessment (FHA) to identify and classify failure condition(s) associated with the system functions and combinations of functions. SAE ARP

4761 further recommends a Fault Tree Analysis be performed which utilizes the failure conditions from the FHA to systematically determine all credible single faults and failure conditions. Once these conditions are identified, a hardware and software design assurance level (DAL) may be established. Further, the safety hazards should then be assessed IAW MIL-STD-882. Supporting analyses and tests should then be performed to validate the design meets the safety requirements. Documentation of the results should then be consolidated in a Safety Assessment Report.

G.7.4.1.3 Software. Software should be developed IAW RTCA DO-178B to the DAL determined by the Safety Assessment for Software utilized to perform power assurance checks in EMS/EHMS; especially when integrating with engine controls / cockpit flight management systems. Use of RTCA DO-178C and four notable associated documents, DO-330, 331, 332, and 333 is also recommended. A Safety Assessment Report should define the DAL and rationale.

Software audits should be performed at critical stages of the software development and verification process. These audits should be performed IAW the FAA Job Aid document: "Conducting Software Reviews Prior to Certification" and be conducted by the Using Service. However, at the very least, self-audits should be conducted by the EMS/SHMS developer to ensure all required RTCA DO-178B objectives, for the given DAL, have been met.

G.7.4.1.4 Accuracy and Repeatability. An EMS/EHMS power assurance check output should be repeatable, for any given pressure altitude, outside air temperature, power condition, and flight speed. The EMS/EHMS output should agree with the output obtained via any other valid set of conditions for the same airframe and engine installation. The power assurance check output should also be accurate when the aircraft data, as compared to test cell data (corrected for platform installation losses), are consistent within a predefined tolerance.

For off-platform accuracy testing, engine power margin, as determined by the EMS/EHMS, should agree with that determined in a test cell on the same engine to within $\pm$ 3 percent. Test data should be obtained on both new and deteriorated engines to demonstrate this capability. The power margin should be defined at maximum rated Turbine Gas Temperature (TGT) and a set of ambient conditions agreed upon between the Developer and the Government.

The accuracy should then be assessed on the operational platform using both new and deteriorated engines. The predicted Power Available to the first engine limiter should agree to that observed on-platform for any given pressure altitude, outside air temperature, power condition(s), and flight speed to within $\pm$ 4 percent. For helicopters, rotor droop should confirm regions of engine control limiting during testing.

Accuracy and repeatability of the EMS/EHMS power assurance check should also be tested on the platform using all engine inlet, exhaust, bleed configurations affecting engine performance (i.e. exhaust suppressors, inlet barrier filters, customer bleed variations). Configuration variations between aircraft models within the fleet should also be considered when implementing a power assurance system.

If the data output of EMS/EHMS power assurance checks is provided to ground based stations and these stations perform data manipulation to determine engine health, then accuracy and repeatability of the ground based station should be verified and validated as well.

G.7.4.1.5 Verification / Validation. EMS/EMHS system level requirements should be verified and validated IAW SAE ARP 4754, Sections 7 and 8. System integration testing should be performed in both laboratory and operational environments to verify and validate the functional performance of

EMS/EHMS power assurance checks. For EMS/EHMS power assurance checks employing an adaptive line to predict power available, flight test plans should demonstrate accuracy requirements using a minimum of 6 flight test engines calibrated in a test cell either before or after "sufficient" flight testing with the EMS/EHMS algorithms on the airframe. "Sufficient" is defined here as enough flight test data such that the engine Shaft Horse Power versus Turbine Gas Temperature characteristic has adapted to reflect an engine-specific profile. Two of the six flight test engines should be degraded at or below what is currently defined as field acceptable. The engine calibrations should consist of a minimum of 10 data points and should include all engine rating points to include Contingency rating and nominal Maximum (10 minute) limiting temperature rating if possible. The Government should approve the engine calibration plan prior to data collection.

G.7.4.2 Engine Stall Monitoring. The propulsion system should be able to identify engine stall events, categorize the severity of the stalls, and indicate appropriate maintenance actions.

G.7.4.3 Anomalies and Fault Detection. Fault Detection for CBM concerns the use of sensors and various signal processing algorithms to identify and isolate a fault in a part or subsystem of the engine. The number of faults detected through sensing is determined by FMECA, which is accomplished as part of engine acquisition. The FMECA defines the expected failure modes and faults that the engine may experience. The EMS and EHMS should be capable of detecting all flight critical faults through data collected by installed sensors. The EMS should also be able to detect at least 70% of all faults and failure modes, which result in the need for maintenance action, established by FMECA accepted by the Government Program Manager.

When the EMS/EMHS detects a fault and directs maintenance action to correct the fault, there should be no more than 10% false positive indications based on subsequent tear down analysis.

The sensors and data management hardware and software should be able to develop CIs and HIs that correspond to flight critical faults with sufficient time to prevent catastrophic failure and possible injury to the flight crew.

In some cases, such as LCF, detection of the faults is not possible in a meaningful time frame prior to catastrophic failure. In those cases, the FMECA should document those failure modes and indicate mitigation methods to preclude their occurrence with means other than direct monitoring.

When the sensors and data management equipment detects an abnormal, but unknown condition (an anomaly), it should record the raw signal in a sufficient interval surrounding the time of occurrence, as well as the computed CI during that interval for post flight processing and analysis.

Determining the sensor strategy, sensor placement, Condition Indicator (CI), and Health Indicator (HI) development for the engine is accomplished the same way as for other aircraft systems and components because of the central role the engine plays in aircraft performance. It is natural to allocate more sensors, computing resources and testing to ensure that engine fault detection is a robust part of the aircraft CBM system.

G.7.4.3.1 Engine Component Faults. EMS and EMHS should be able to detect all flight critical engine faults with 90% confidence.

G.7.4.3.2 Instrument Faults. EMS and EMHS should be able to detect and isolate sensor and instrument faults in their system and highlight suspect data caused by sensor or instrument error.

G.7.4.3.3 Diagnostics. The objective of a turboshaft engine diagnostic system is to identify faults before they lead to catastrophic failures and to aid maintenance personnel in taking corrective action. This prevents costly damage to engine components and can reduce costs due to premature or past-due maintenance.[43]

Diagnostic algorithms should isolate faults to specific components with high confidence. Without specific isolation of a fault through the calculation of an accurate CI and corresponding HI, mechanics may begin replacing suspect components until the fault clears (the engine returns to normal operational limits). This process unnecessarily increases maintenance costs and aircraft down time. Current generation turbine engines have built in tests that can provide the degree of fault isolation required if the algorithms are designed to use all the available evidence.

While condition monitoring systems can reduce maintenance costs and aircraft downtime, the systems themselves can be expensive. The more sophisticated the system is, the more it will tend to cost to develop and implement. This cost should be balanced against the gains made in maintenance and logistics costs. As a reference point, the key performance parameters for the CH-47 Aviation Turbine Engine Diagnostic System (ATEDS) in 2004 were[44]:

a. Detect and isolate engine faults from airframe faults and provide fault detection and isolation (threshold) 96%, (objective) 100%

b. Verify proper operation of engine control units, electronic sequencing units, Line replaceable unit/Line replaceable module (LRUs/LRMs) engine sensors, and engine wiring harness 96% (T), 100% (O)

c. Verify proper operation of engine-to-airframe interface (electrical) at 96% (T), 100% (O)

d. Mean time between Essential Function Failure should be at least 360 hours (T), 820 hours (O).

Appendix D contains guidance related to developing CIs and HIs for the CBM system. While EMS and EMHS have often been developed independently of aircraft or vehicle health monitoring systems (VHMS), the intent should be for the aircraft and propulsion systems to integrate and consolidate system resources whenever possible to save weight, cost and complexity.

G.7.4.4 Prognostics. In propulsion systems, there are typically two different categories of remaining useful life (RUL) which are based on either: 1) the existence of a confirmed fault or 2) the established life usage indicators for the rotating components which have a defined design life (turbine blades, disks, etc) based on exposure to thermal stresses or high centrifugal loading (fault mechanisms that are always present).

The estimates of RUL, which influence maintenance actions to remove either the engine or subcomponents, are based on the existence of faults detected by CIs and follow the process of this ADS and its Appendices D and E.

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[43] Litt, J.; D. Simon, S. Garg; T-H Guo, C. Mercer, A. Behbahani, A. Bajwa, and D. Jensen. “A Survey of Intelligent Control and Health Management Technologies for Aircraft Propulsion Systems,” NASA Tm-2005-213622, May 2005, pg. 10.

[44] Stramiello, A., J. Moffatt, G. Kacprzynski, and J. Hoffman. “Aviation Turbine Engine Diagnostic System (ATEDS) for the CH-47 Helicopter,” 2008.

For predicted RUL calculations based on life usage algorithms that measure fatigue or thermal damage, the engine manufacturer should develop models and establish life limits which follow established military and commercial standards as referenced in Appendix D. These life limits should be reserved for only the most critical and difficult to detect failure modes that are analyzed and investigated during engine qualification testing.

G.7.5 Auxiliary power units. Designs of APUs over time have become more efficient and reliable. The FMECA should establish the need for monitoring fault modes and establishing CIs and HIs for the APU. Trade studies in conjunction with the FMECA should establish the optimum design of sensors and data management, balancing the relative ease of maintenance and reliability of the specific APU and airframe with added cost of CBM data management.

The process for developing CI and HI for the APU is found in Appendix D and E. Integration of the sensors, signal processing and data management for the APU with the CBM system of the aircraft should be a high priority in the design of the APU monitoring system.

Specific objectives for fault detection, mean maintenance hours to repair, and MTBF are controlled by the cognizant government engineering office, based on recommendations from the aircraft manufacturer.

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APPENDIX H

EMBEDDED DIAGNOSTICS/PROGNOSTICS AND HEALTH MANAGEMENT (PHM) OF ELECTRONICS COMPONENTS

H.1 SCOPE

This Appendix addresses Embedded Diagnostics/Prognostics and Health Management for Electronics. This appendix addresses:

(1) Methodology and Implementation of Diagnostic of Electronics Components. It covers the use of Built-in Test (BIT)/Built-in Test Equipment (BITE), sensors, acquisition systems, Portable Maintenance Aids, Automatic Test Equipment, and signal processing algorithms to detect, identify, and characterize faults in aircraft electronics systems.

(2) Future placeholder for Methodology and Implementation of Prognostics for Electronics Components

(3) Methodology and implementation for optimizing environmental design specifications (ie. test profiles, thermal analyses, thermal zones) (this might actually need to go in a separate appendix other than electronics – but believe we need to capture this methodology somewhere).

H.2 REFERENCES AND APPLICABLE DOCUMENTS

ARMY REGULATIONS	
Army Regulation 70–1	Army Acquisition Policy, 22 July 2011.
DA PAM 750-43	Army Test Program Set Implementation Guide, 28 June 2006.

(Copies of these documents are available online at http://www.apd.army.mil/)

MILITARY STANDARDS	
MIL-HDBK-470	Designing And Developing Maintainable Products And Systems, Vol I, 4 August 1997.
MIL-HDBK-2165	Testability Handbook For Systems And Equipment, 31 July 1995.

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

TECHNICAL MANUALS	
TM 1-1520-248-T	Operational Checks And Maintenance Action Precise Symptoms (Maps) Diagrams Aviation Unit and Intermediate Maintenance Manual for Army OH-58D Helicopters
TM 1-1520-248-MTF	Maintenance Test Flight Army OH-58D Helicopter

(Copies of this document are available online at http://www.armyproperty.com/tm/TB%2043-0211 or 505 E. Huron Street, Suite 202; Ann Arbor, MI 48104 2011 Crystal Drive, Suite 400; Arlington, VA 22202 DUNS Number: 829504880 / CAGE Code: 5BMR7 (703) 269-0013 / (734) 585-5061)

BOOKS	
Vachtsevanos, George and Frank Lewis, Michael Roemer, Andrew Hess, Biqing Wu,	Intelligent Fault Diagnosis and Prognosis for Engineering Systems, Wiley, September 2006.
Pecht, Michael.	Prognostics and Health Management of Electronics, Wiley, September 2008.

(Copies of these documents are available from sources as noted.)

H.3 GENERAL GUIDANCE.

H.3.1 Background Maintenance processes are currently focused on diagnosing failure of electronic components. The Army is utilizes a two level maintenance process, field maintenance which consists of aviation unit maintenance (AVUM) and aviation intermediate maintenance (AVIM), and sustainment maintenance (Depot). Typically, unit level maintenance is performed while electronic systems are still installed on the aircraft via primary test equipment built into the electronic systems. Maintenance technicians use the built-in test (BIT) and built-in test equipment (BITE) to verify that systems are operating properly while they are in the helicopters. Pilots also use the equipment to verify system readiness before and during combat missions. The repair capability at the unit level is normally limited to minor troubleshooting, removal and replacement of parts and components, and daily servicing. The AVIM refers to maintenance that must be done in a repair shop. If a system defect is identified and cannot be repaired at the unit level, the faulty component is removed and sent to the intermediate level repair shop. Various terms are used to refer to an item that is removed and replaced and include Line Replaceable Unit (LRU) and Weapon Replaceable Assembly (WRA). The repair shop has more sophisticated test equipment that can diagnose faults at a more detailed component level. Intermediate maintenance provides backup support for the unit level maintenance as well as an expanded capability to perform diagnostic troubleshooting, tear-down analysis and repair, and limited rebuilding of components, to include engines. Doctrinally, repairs of aircraft and components completed by the intermediate maintenance unit are usually returned to the owner.

Faulty components that cannot be repaired at the intermediate level repair shop are shipped to a remotely located depot. The depot level refers to maintenance that is beyond the capability of the AVUM/AVIM and is performed at a central facility located farther away from tactical units.

During the conceptual and early design phase a failure rate prediction is a method that is applicable mostly, to estimate equipment and system failure rate. Following models for predicting the failure rate of items are given (in MIL-STD-217F):

- Failure rate prediction at reference conditions (parts count method)

- Failure rate prediction at operating conditions (parts stress method)

Failure rate data for electronic components refers commonly to the phase with constant failure rate (useful life period on the classical bathtub curve). It is recognized that the constant failure rate assumption for electronic is sometimes not justified. The design can be less than perfect or not every failure of every part will cause the equipment to fail (error correction circuitry, memory, redundant/fault-tolerant circuitry etc.) But such an assumption provides suitable values for comparative analysis.

H.3.2 Diagnostic framework for electrical/electronic systems. Figure 1 illustrates the typical configuration of an electrical subsystem as part of the overarching LRU.

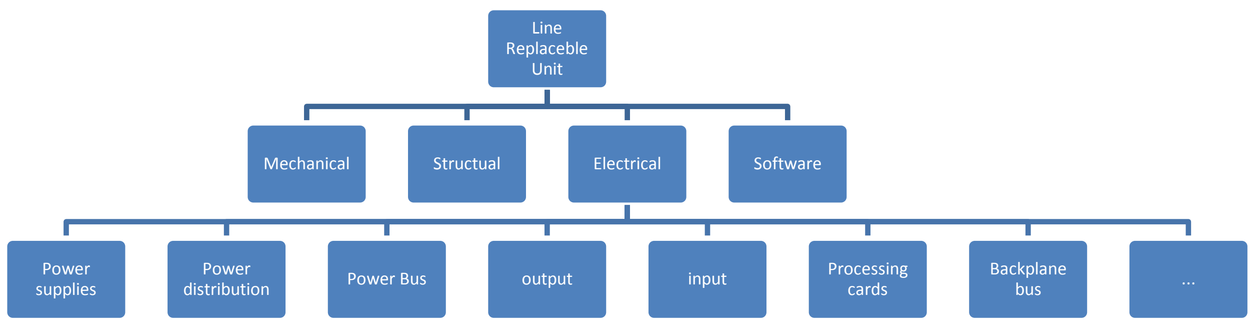

FIGURE H-1 LRU electrical subsystems

The elements of a typical electronic component, such as solder joints, wires, and printed circuit boards (PCBs), accumulate damage at rates depending on their environment and operational conditions. Therefore, the time between repair/removal is a function of environmental history, design tolerances, material properties, and operational conditions including software. Software controls how the embedded electronic system operates, speed, duty-cycle, etc and therefore effect to internal temperature of the system. Parameters that accelerate failure most significantly in electronics, and in turn should be monitored by the health monitoring system, are

- Temperature and humidity
- Temperature variation
- Vibrations
- Shock
- Power quality
- Extreme operating conditions

Electrical system health monitoring relies on thermal monitor, impedance measurement (contact and breakdown), and visual inspection. Many systems include built-in test, either integrated in the system or as an external device that performs tests of the system to ensure proper operation which can be used to identify system faults as they occur. This approach can result in a large number of false positives. These false positives often result in Can Not Duplicate (CND) failures within the electronic modules. High level of CND failures limits the ability to transition from diagnostics to prognostics for electronics. However, there are few systems that use distributed sensors to continuously monitor the sources of failure, such as vibration, heat, or power quality, and from those sensors detect the onset of incipient failures.

Electronic components have other attributes that influence the requirements for diagnosis, prognosis, and health management.

- Aside from the failure modes of the individual elements its failure has an effect on adjoining components that can generate additional failure modes and cause increases in data or power loads on individual elements.

- The path of propagation can be variable, causing the estimation of MTBF/MTTR to be more complex and probabilistic.

- A typical electronic system frequently is upgraded in capability or performance, making it difficult to manage the development of diagnostic and prognostic processes for these coexisting, multiply-variant configurations.

A possible framework to model system behavior and detect faults and predict the onset of incipient failures useful life in electronics may include the following tools:

- Embedded life modes. Failure models derived from reliability studies, OEM data, etc. Such models express in a probabilistic sense the remaining life estimate of electronic components.

- Operational Environmental information. Sensors can be distributed to measure temperature, humidity, and vibration. From test data, models can be developed that estimate the life impact to exposures of various levels of environmental stressors and estimate the useful life remaining.

- Operational data. The number of power-on cycles and run time can be an important factor in wiring and PCB failures. Tracking the operational data per serial number, as well as the power quality feeding the system during operation, can be used to update MTBF/MTTR.

H.3.3 Current Diagnostic Test Methods. There are generally two types of test: closed loop test and open loop test.

H.3.3.1 Closed loop tests Closed loop tests are tests that do not require operator interaction to complete successfully. A loopback path from the actuator (or data source) to a sensor (or data sink) which is used to verify correct operation. Because no human interaction is required in the test loop, the loop is said to be “closed.” Examples of a closed loop test include communications interface loopback tests, discrete and analog output loopback tests (where hardware support is available), and other special purpose electronic interfaces where diagnostic loopback paths have been designed into the hardware.

Closed loop tests can affect the operation of the system during their execution and may not be comprehensive. For example, a closed loop test may not verify end-stage hardware such as line drivers/receivers, lamps, and attached instrumentation.

H.3.3.2 Open Loop Tests Open loop tests are tests that require some kind of operator interaction to complete and determine whether the test succeeded or failed. With an open loop test, there is no loopback path from the source to the sink that does not include an operator. A human is part of the test loop, so the loop is said to be "open." Examples of open loop tests include lamp and gauge tests, switch tests, and other tests that usually involve operator controls and displays

There are generally two kind of testing environments, in-situ or depot environment. There resources available for each environment would greatly effects the diagnosis.

H.3.3.3 Testing in In-Situ Environments In situ tests are executed at the O-level, with the UUT installed in its normal operating environment. In situ test are often referred to as Built-in Tests (BITs) and can also be categorized by their execution time relative to system state. In situ test and BITs can be conducted offline or online. Offline tests are executed outside of normal system operation, often in a specialized test environment. Online tests are a collection of background diagnostics that can run in parallel with normal system operation without affecting it. Offline and online in situ test and BITs are described in more detail in the following sections.

H.3.3.3.1 In situ offline Tests In situ offline tests are characterized by when and how they are executed. Typically, a collection of tests is run when the system is powered-on and initiated, with additional tests being run on request. These tests are executed when the prime function of the system is non-operational, or "offline." The power-on and initiation tests directly support the Verification of Operational Readiness diagnostic, while operator-initiated tests more directly support the Fault Isolation, Repair of Repairable, and Other Maintenance Action diagnostic.

Offline in-situ tests can include power-on BITs, power-on self-tests, and initiated BITs, which are described in more detail in the next section.

H.3.3.3.1.1 Type of BIT: Power-on BITs, often referred to as PBIT (power-on BIT) or IBIT (initial BIT) tests, are tests that run automatically when a system is powered-up and initialized. They are typically closed loop and support the Verification of Operational Readiness mission. PBIT tests can have an additional sub-category of tests called power-on self tests (POSTs). POSTs are a subset of PBITs. POSTs are a set of low-level tests, usually developed by a commercial off-the-shelf (COTS) hardware vendor or specialized hardware developer. They are hosting in non-volatile memory and execute prior to system software boot. One of the jobs of PBIT is to collect the diagnostic information from the various POST tests.

Initiated BITs, (IBITs) also referred to CBITs (commanded BITs) are tests that run when an operator initiates them. IBITs support the Fault Isolation and Diagnosis and Repair of Repairable missions. Initiated BITs usually consist of a subset of PBIT tests augmented with additional diagnostics. The additional diagnostics can include open loop tests to verify controls and displays and tests to verify proper communication and interaction with other connected systems. Initiated BIT tests can run as a single iteration or as a repeating set of one or more tests. Repetitive execution of the tests is necessary to help detect and isolate intermittent failures.

H.3.3.3.1.2 Online Tests In situ online tests are run periodically or continuously in the background during normal system operation. Continuous BITs or periodic BITs are online tests that run in the background to support the Fault Detection and Characterization diagnostic mission. They are typically either closed loop tests or statistics collection activities that attempt to verify that data is flowing properly across the entire system. Examples of continuous BIT tests include the collection of network interface error statistics (such as checksum failures or parity errors) and other closed loop tests that can be run without affecting normal system operation.

The purpose of these tests is to support the Fault Detection and Characterization diagnostic and to provide sufficient Fault Isolation to support line maintenance operations. Line maintenance operations consist of component replacement and other adjustments made at the LRU level.

H.3.3.4 Testing in Depot Environments. Depot tests are run when the Unit Under Test (UUT) is installed outside of the normal operating environment. The goals of testing in a depot environment are to verify that new components function properly and to support the Repair of Repairable diagnostic.

Testing in a depot environment typically includes, but may not be limited to, all of the in situ tests run for an LRU. These tests are run to verify that an LRU has failed or has been repaired and to focus additional diagnostic and repair activities that are available only to the depot, vendor, or hardware developer.

Depot tests that are executed below the LRU level typically require the support of additional ATE. The development, standardization, configuration, and use of ATE is its own complex domain and is beyond the scope of this appendix.

H.3.4 Diagnostic Example The current process is lends itself to scenarios where the current testing methods result in Can Not Duplicate (CND) failure for the electronic module. The following scenario is typical in the diagnosing faults for electronic modules:

Test Incident #19: IBIT Fail (050010) "IR focus test"

Description of Problem: Unable to manually focus IR video.

Root Cause: Undetermined. After the first fail, the receiver was removed from the turret and, when bench test attempted to isolate to the failed item, the failure symptom went away. Efforts to reproduce were unsuccessful. Turret was reassembled and placed back in test. After ~150 hrs the failure returned. Again the open circuit was verified at bench test within the turret but once the turret was removed, the failure vanished again.

Current Status: TBD

Number of Occurrences: 2

It is possible the failure is related to operating environments such as high temperature, temperature cycling, humidity, and vibration. The environmental stressors would degrade the connections or cause solders joint cracks or corroded which would cause the intermittent failure.

One method to correct this CND failure is to capture the conditions of use and determine the accumulated damage. The sensor will be embedded to the LRU (Temperature, vibration and humidity sensors). The data would be transfer to base station where it would be simplified and analyzed. The following process outlines the steps to implement an electronics Health Monitoring System to alleviate the CND outcome.

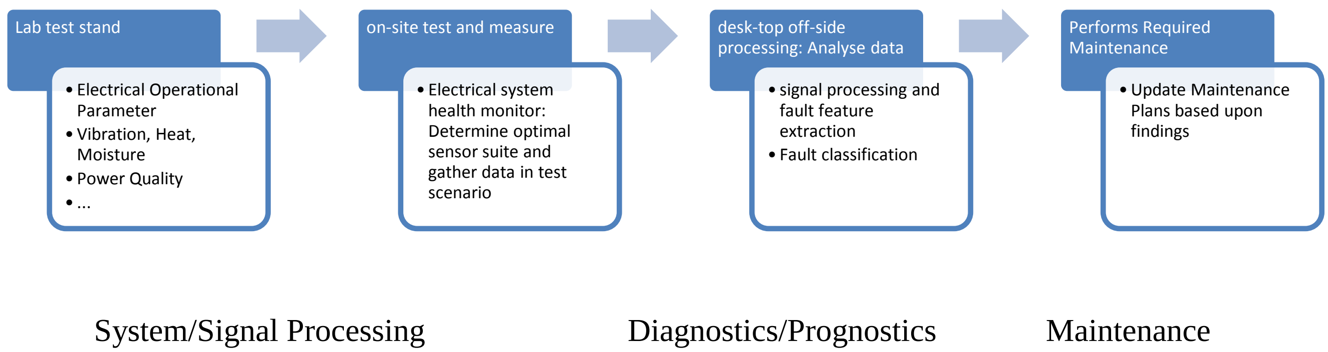

FIGURE H-2: Diagnostic using BIT tests and embedded sensor.

Steps to implement are:

1. Study the failure on lab test stand to establish baseline database. Applying environmental and usage parameters to the electronic module in a lab setting to determine which parameters drive component failure. (electrical failure plus temperature, vibration and humidity stress data)

2. Integrate the electrical system health monitoring: Based upon lab findings, develop sensor suite to capture data for analysis.

3. When failure detected, perform signal processing, fault feature extraction and fault classification.

4. Perform Maintenance: Based upon lab data, update maintenance procedures. Any system created to mitigate testing and inspections should be at least as reliable as the legacy test. The system should demonstrate a reliability level greater than the current testing process.

H.3.5 Environmental and Operational Monitoring

H.3.5.1 Baseline Usage Condition. Health monitoring and identifying a baseline usage condition to evaluate system health are fundamental for Diagnostics/prognostics. The challenge here is developing an efficient "training" program for the algorithms to define healthy conditions. Another challenge is identifying what usage and environmental conditions to consider for the baseline. For environmental monitoring, autonomous tags that utilize RFID and programmable sensor kits offer a noninvasive solution. These tag devices could host a range of environmental monitors for contamination, corrosion, electrical degradation, and so on. Further research is needed to develop tags for prognostics.

H.3.5.2 Operational Environmental Load and Usage Condition. Environmental and operational monitoring could also be considered in the development of environmentally tolerant electronics. Environmental and usage conditions obtained in field trials could be fed into design tools to simulate whether future devices and design can withstand these conditions. Simulation techniques, tools, and autonomous sensors are all areas of opportunity for research and development.

Currently the BIT is primarily used for diagnostic purposes only. However, there are opportunities to capture the CBIT data while in flight, interface with the embedded health monitoring system and provide data back to the maintainer and engineering community to develop prognostic algorithms. For example, assume a radio is being continuously monitored in flight. That CBIT data should be collected across the fleet for engineering analysis. Based upon the fleet data, components exhibiting a high intermittent failure rate in flight can be tracked. By capturing the data, engineering analysis could build CIs for the radio predicting failure based upon the CBIT failure trends in flight.

The collection of CBIT data will also allow for the correlation of the failure responses with in flight conditions and usage. By collecting the data through the HUMS system it should be possible to correlate usage data with the CBIT data. By trending the information, it may be possible to identify degrading parts as they fail.

APPENDIX I

SAMPLE SIZES FOR MAINTENANCE CREDITS USING VIBRATORY CBM ON PROPULSION SYSTEMS

I.1 SCOPE

This appendix provides guidance for methodologies, applications, and considerations of sample sizes and statistical processes in verifying and validating vibratory CBM algorithms prior to approval of US Army On-Condition maintenance as a replacement to legacy TBO maintenance. Examples are provided to facilitate an understanding to the guidance.

I.2 REFERENCES AND APPLICABLE DOCUMENTS

I.2.1 References

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Antolick, L., J. D. Branning, D. Wade, and P. Dempsey.	"Evaluation of Gear Condition Indicator Performance on Rotorcraft Fleet," Proceedings of the American Helicopter Society 66th Annual Forum, Phoenix, Arizona, May 11-13, 2010.
Dempsey, Paula, J., Jonathon A. Keller, Daniel R Wade.	Dempsey, Paula, J., Jonathon A. Keller, Daniel R Wade. *Signal Detection Theory Applied to Helicopter Transmission Diagnostic Thresholds*, NASA/TM-2008-215262; AMRDEC PAO Control Number FN 3597, July 2008.
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Rees, D. G.	"Foundations of Statistics," CRC Press, 1987.

(Copies of these documents are available from sources as noted.)

NATIONAL INSTITUTE OF STANDARDS AND TECHNOLOGY (NIST)	
NIST/SEMATECH e	Handbook of Statistical Methods, Engineering Statistics Handbook, 2011

(Copies of these documents are available online at http://www.nist.gov/itl/sed/gsg/e-handbook.cfm or NIST, 100 Bureau Drive, Stop 1070, Gaithersburg, MD 20899-1070)

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(Copies of this document are available online at http://www.rmc98.com/Comparison%20of%20Test%20Stand%20and%20Helicopter%20Oil%20Cooler%20Bearing%20Condition%20Indicators.pdf)

I.2.2 Applicable Documents. The documents listed below are useful in understanding the information provided by this Appendix.

I.2.2.1 Government Documents.

MILITARY STANDARDS	
MIL-HDBK-781	Reliability Test Methods, Plans, and Environments for Engineering, Development, Qualification, and Production, Handbook for. April 1996.

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

ASTM INTERNATIONAL (AMERICAN SOCIETY FOR TESTING AND MATERIALS)	
ASTM D664-11A	Standard Test Method for Acid Number of Petroleum Products by Potentiometric Titration.
ASTM E1049-85(2011)e1	Standard Practices for Cycle Counting in Fatigue Analysis.
ASTM E 122-09	Standard Practice for Calculating Sample Size to Estimate, with Specified Precision, the Average for a Characteristic of a Lot or Process, 1999.

(Copies of these documents are available online at http://www.astm.org or from the ASTM International, 100 Barr Harbor Drive, P.O. Box C700, West Conshohocken, PA 19428-2959.)

I.2.2.2 Non-Government Documents.

SOCIETY OF AUTOMOTIVE ENGINEERS (SAE) INTERNATIONAL	
AIR5113 SAE Aerospace Information Report.	Legal Issues Associated with the Use of Probabilistic Design Methods. 7 June 2002.

(Copies of this document are available from http://www.sae.org/standards/ or SAE World Headquarters, 400 Commonwealth Drive, Warrendale, PA 15096-0001 USA. Phone (US) 1-877-606-7323)

OTHER	
Abernethy, B. Robert	The New Weibull Handbook – Reliability & Statistical Analysis for Predicting Life, Safety, Survivability, Risk, Cost, and Warranty Claims, Fourth Edition, 2000.
Conover, W. J.	Practical Nonparametric Statistics. New York, NY: J. Wiley and Sons, 1999.
Montgomery, Douglas C.	Design and Analysis of Experiments. New York: Wiley, 1991.

(Copies of these documents are available from sources as noted.)

I.2.3 Notations

$\boldsymbol{\alpha}$ = type I error rate, reject null hypothesis when null hypothesis is true (false positive on a healthy component)

$\boldsymbol{\beta}$ = type II error rate, fail to reject null hypothesis when null hypothesis is false (false negative or missed alarm on an unhealthy component)

$\boldsymbol{\delta} = \bar{y}_1 - \bar{y}_2$ = difference in means

***E* =** maximum tolerant error of mean estimate

H_0 = the Null Hypothesis

H_1 = the Alternative Hypothesis

n = *sample* size

P = Probability

Q = probability of failure

R = probability of success

R_{U1} = Upper Confidence Limit or Maximum Designed-In Reliability of CI/HI

R_{L1} = Lower Confidence Limit or Minimum Demonstrated Reliability of CI/HI

r = number of failures in a trial

***S* =** standard deviation

s = the number of successes in a trial

Sp = pooled or average standard deviation

$\boldsymbol{\sigma}$ = standard deviation

σ^2 = variance

μ =*population* mean

$\mu_1 = damaged\ population$ mean

μ_2= *undamaged population* mean

$\bar{X}$ **=** estimated sample mean

$\bar{y}_i$ =*sample* mean of undamaged components

$\bar{y}_2$ =*sample* mean of damaged components

z = (standard normal) distribution

Z= confidence Interval

I.3 GENERAL GUIDANCE

I.3.1 Background**.** To verify and validate CIs and His, an appropriate sample size of faulted and unfaulted components needs to be determined. In theory, more samples would be better. But in reality, sample size is always a limiting factor for validation problems. Many factors impact the selection of appropriate sample size. When choosing a sample size, the following issues must be considered:

a. What population parameters we want to estimate;

b. Criticality due to lack of accurate information;

c. Cost of sampling (importance of information);

d. How much is already known (prior knowledge / experience);

e. Spread (variability) of the population;

f. Practicality: how hard is it to collect data;

g. How precise we want the final estimates to be.

Inevitably, there is a trade-off among sample size, cost and precision of the anticipated regression model. To develop a highly accurate predictive model, more samples are generally required which results in higher validation cost.

To select an appropriate sample size, a probabilistic statement about what is expected of the sample is needed. The targeted estimate from sampling must be determined first. Typically, several statistical characteristics are of interest, including mean value, standard deviation, population proportion, etc. Depending on an identified target estimate, different sampling requirements may be followed. For example, to establish a reasonable statistical estimate for average value, a small sample size (as low as 8 – 10) may be sufficient, while a larger sample size may be needed to ensure a good estimate of standard deviation or population proportion. The required precision for the estimate should also be defined. In general, the potential use of the estimate, in terms of criticality of application, affects the precision requirement which further impacts sample size selection. A probability statement connecting the desired precision of the estimate with the sample size is the essential step in sample size determination. The statement may contain unknown properties of the population such as the mean or variance. This is where prior information can help. The final sample size should be scrutinized for practicality. If it is unacceptable, the only way to reduce it is to accept less precision in the sample estimate.

I.3.2 Concept of Confidence Interval and Hypothesis Testing. In statistics, a confidence interval is a particular kind of interval estimate of a population parameter. Instead of estimating the parameter by a single value, an interval is likely to include the given parameter. Thus, confidence intervals are used to indicate the reliability of an estimate. The width of these confidence intervals is a measure of the overall quality of the estimated parameter or regressed model. A narrower confidence interval indicates a tighter statistical estimate of the parameter with less variability or standard deviation around its estimated value. Mathematically, a confidence interval can be defined as the boundary in which an experimental outcome is anticipated to stay for a given level of probability.

In many engineering applications, it is often required that we make a decision whether to accept or reject a statistical statement about a statistical parameter of interest. The statement is often referred to as the hypothesis and the associated decision making procedure about the hypothesis is called hypothesis testing. This is one of the most useful aspects of statistical inference, since many types of decision-making problems, tests, or experiments in the engineering world can be formulated as hypothesis-testing problems. Furthermore, there is a very close connection between hypothesis testing and confidence intervals.

A statistical hypothesis is a statement about the parameters of one or more populations. For any given statement about parameter μ, there are two hypotheses: H_0 and H_1, such as:

$$H_0{:}\,\mu = \mu_0;\ or\ H_1{:}\,\mu \neq \mu_0$$

where H_0 is called the Null Hypothesis and H_1 is referred to as the Alternative Hypothesis. From a probabilistic standard point of view, the chance that μ equals exactly μ_0 is zero. Therefore, the Null Hypothesis is practically associated with a region, within which μ is close enough to μ_0 so the null hypothesis will be accepted. By convention, this region is usually called the acceptance region. In general, the confidence interval is used to determine the boundary for the acceptance region.

Selection of a Null Hypothesis varies from one application to another. For the case associated with structural damage detection, the Null Hypothesis usually states that damage exists. For mechanical diagnostics, the Null Hypothesis typically states that no defect or unacceptable fault exists.

I.3.3 Type I and Type II Error**.** Due to the inherent variability associated with any experimental study, the outcome of an observable event exhibits some randomness. As a result, the aforementioned decision procedure may lead to two wrong conclusions: Type I error and Type II error.

As listed in Table I-I, Type I error is related to cases where Non Destructive Equipment (NDE) or a CBM monitoring CI/HI indicates detection when no fault actually exists (false positive on a healthy component). While, Type II error is associated with the situations in which NDE or a CBM monitoring CI/HI misses detection when a fault exists (false negative or missed alarm on an unhealthy component).

TABLE I-I. Decision in hypothesis test

Decision	H_0 is True	H_0 is False
Reject H_0	Type I Error	No Error
Fail to Reject H_0	No error	Type II Error

I.3.4 Confidence Interval on Sample Mean. Suppose that a set of *n* random samples are drawn from a given distribution of CI/HI readings with unknown mean μ and known variance σ^2. According to the Central Limit Theorem, the estimated sample mean $\bar{X}$ is normally distributed with a mean μ and variance σ^2, if the sample size of CI/HI readings is sufficiently large. We may standardize by subtracting the mean and dividing by the standard deviation, which results in the variable:

$$Z = \frac{\bar{X} - \mu}{\sigma/\sqrt{n}}$$

Now Z has a standard normal distribution. Given a significance level of α, confidence interval of Z can be expressed as:

$$P\left(-z_{\alpha/2} \le \frac{\bar{X} - \mu}{\sigma/\sqrt{n}} \le z_{\alpha/2}\right) = 1 - \alpha$$

So, the confidence interval of an estimated mean can be calculated by:

$$\bar{X} - z_{\alpha/2}\,\sigma/\sqrt{n} \le \mu \le \bar{X} + z_{\alpha/2}\,\sigma/\sqrt{n}$$

where $z_{\alpha/2}$ is the upper 100α/2 percentage point of the standard normal distribution.

Once the significance level is defined and maximum tolerant error of mean estimate, $E = |\mu - \bar{X}|$ is specified, the sample size can be determined by:

$$n = \left(\frac{z_{\alpha/2}\sigma}{E}\right)^2$$

In regression analysis, Student's t-distribution (or simply the t-distribution) is of primary interest. The t-distribution arises in the problem of estimating the mean of a normally distributed population when the sample size is small (e.g. less than 30). It is also the basis for establishing confidence intervals of the regression model.

Again, let us assume that a set of *n* random samples drawing from a normal distribution. In this case, both mean μ and variance σ^2 are unknown. The random variable:

$$t = \frac{\bar{X} - \mu}{S/\sqrt{n}}$$

has a t - distribution with *n* - 1 degrees of freedom. $\bar{X}$ and S are the estimated mean and variance obtained from the sample data. In general, the general appearance of the t - distribution is similar to the z (standard normal) distribution. Both distributions are symmetric and unimodal, and the maximum ordinate value is reached at their mean value. However, the t - distribution has heavier tails than the normal; that is, it has more probability in the tails than the normal distribution. As the number of degrees of freedom approaches infinity, the limiting form of the t distribution is the standard normal distribution.

Similar to the *z*-distribution, confidence interval of t can be determined given a significance level of α:

$$P\left(-t_{\alpha/2,\ n-1} \leq \frac{\bar{X}-\mu}{S/\sqrt{n}} \leq t_{\alpha/2,\ n-1}\right) = 1-\alpha$$

So, the confidence interval of estimated mean can be calculated by:

$$\bar{X} - t_{\alpha/2,\ n-1}\, S/\sqrt{n} \leq \mu \leq \bar{X} + t_{\alpha/2,\ n-1}\, S/\sqrt{n}$$

where $t_{\alpha/2,\ n-1}$ is the upper 100α/2 percentage point of the t – distribution with (n – 1) degrees of freedom.

Once the significance level is defined and maximum tolerant error of mean estimate, $E = |\mu - \bar{X}|$ is specified, the sample size can be determined by:

$$n = \left(\frac{t_{\alpha/2,\ n-1} S}{E}\right)^2$$

It should also be noted that the value of t – distribution at given α level stabilizes once the number of degrees of freedom reaches 4, as depicted in Figure I-1, below.

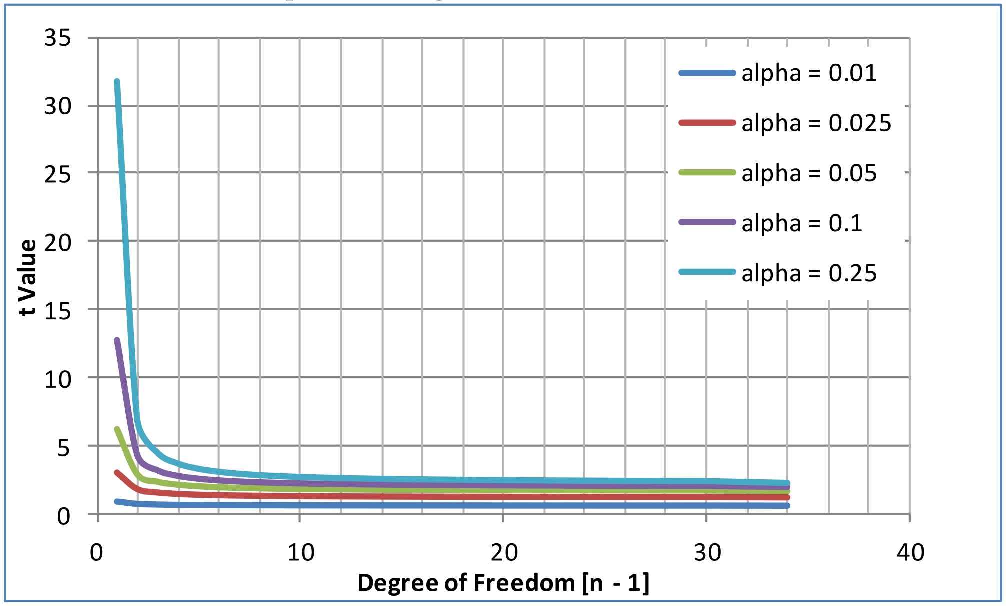

FIGURE I-1: Information gain as function of sample size

Initially, t-distribution applies when the population standard deviation is unknown and has to be estimated from the data. Quite often, however, some practical problems will treat the population standard deviation as if it were known. These problems are generally of three kinds: (1) those in which the sample size is so large that one may treat a data-based estimate of the variance as if it were certain; (2) those that illustrate mathematical reasoning, in which the problem of estimating the standard deviation is temporarily ignored because that is not the point that the author or instructor is then explaining, and (3) those that a great understanding or prior knowledge exist about the variance of the anticipated population.

Given known standard deviation and significance level, the sample size can be easily determined using the aforementioned equation. If the error tolerance is reasonable, the required sample size can be optimized.

I.3.5 Test on Population Proportion. It is often necessary to construct confidence intervals on a population proportion. For example, suppose that a random sample of size n has been taken from a large population and that ($X \leq n$) observations in this sample belong to a class of interest. Then $\hat{p} = X/n$ is a point estimator of the proportion of the population p that belongs to this class.

Binomial or Bernoulli trials are often used in Population Proportion test. In Bernoulli trials, it assumes that (1) each trial results in either a success or a failure, (2) the probability of success does not change from trial to trial, and (3) the outcome of one trial does not affect the outcome of any other trial. If the aforementioned conditions hold, the probability of exactly s successes out of n Bernoulli trials can be calculated via Binomial distribution,

$$P(s) = \binom{n}{s} R^s Q^{n-s} = \binom{n}{s} R^s (1-R)^{n-s}$$

where, n is the total number of Bernoulli trials, s is the number of successes trials, R is the probability of success in each trial for the CI/HI to correctly categorize a faulted/unfaulted component, and Q is the probability of failure for the CI/HI to correctly categorize a faulted/un-faulted component in each trial. Accordingly, the cumulative Binomial distribution function is:

$$P(s \leq S_1) = \sum_{k=0}^{S_1} \binom{n}{k} R^k (1-R)^{n-k}$$

which gives the probability of up to and including S_1 successes in n trials, when each trial has a probability of succeeding of R.

Given the number of failures, r, or the number of successes, s, in n Bernoulli trials, the average value of reliability can be estimated, from:

$$\bar{R} = \frac{s}{n}; \quad \bar{Q} = \frac{r}{n}$$

The confidence limits on R can also be determined, upon a given significance level.

The lower one-sided confidence limit on the reliability, R_{L1} may be obtained from:

$$\sum_{k=0}^{r} \binom{n}{k} {R_{L1}}^{n-k} (1-R_{L1})^k = \alpha = 1 - CL$$

Similarly, the upper one-sided confidence limit on the reliability, R_{U1} may be obtained from:

$$\sum_{k=r+1}^{n} \binom{n}{k} {R_{U1}}^{n-k} (1-R_{U1})^k = \alpha = 1 - CL$$

I.3.6 Reliability Demonstration Test. Reliability demonstration test is often employed to illustrate that a component or system meets the design requirement and possesses a desirable level of reliability. Several methodologies are available for such purpose. Among them, accept-reject testing has been widely used. In accept-reject testing, the null assumption is that reliability to be demonstrated meets the requirement. In addition, two types of risks are considered and previously discussed: Type I Error α, such that:

$$P(Reject|R \geq R_{U1}) \leq \alpha$$

and Type II error β-such that:

$$P(Accept|R \leq R_{L1}) \leq \beta$$

where α and β are usually chosen to be 1%, 5%, 10%, or 15%, depending on the maximum risk tolerated. The significance level α and β may be chosen to be equal or different.

The accept-reject test consists of running *n* single sample CI/HI readings from separate component tests at specified conditions and mission duration and accepting the reading if *r* or fewer sample readings out of *n* fail during the tests. The validation process is rejected if more than *r* samples fail during the test. Obviously, outcome of the accept-reject tests depends on the designed-in reliability of the validation process, the total sample size and maximum allowable number of failures, and the underlying risk tolerated. The sample size can be determined by:

$$P(Accept|R \leq R_{L1}) = \sum_{k=0}^{r} \binom{n}{k} {R_{L1}}^{n-k}(1 - R_{L1})^k = \beta$$

once the reliability goal, R_{L1}, and allowable number of single sample CI/HI readings from separate component tests that may fail for an accept decision, *R*, are determined and confidence level (1-β) is specified. Known total sample size and the number of allowed number of failures, the reliability of the validation process corresponding to a risk of α can be estimated by:

$$P(Reject|R \geq R_{U1}) = \sum_{k=r+1}^{n} \binom{n}{k} {R_{U1}}^{n-k}(1 - R_{U1})^k = \alpha$$

The accept-reject test consists of following steps:

a. Determining a reliability goal R_{L1} as appropriate for the product or process;

b. Selecting a Confidence Level appropriate for the accuracy required from the results of the accept-reject testing;

c. Selecting the test duration and conditions to achieve desirable reliability level for demonstration;

d. Calculating the CI/HI Sample Size for the testing;

e. Performing the testing with calculated sample sizes to validate the required Reliability with given number of allowable failure.

If additional failures are observed within the testing duration, the sample size needs to be recalculated or confidence level needs to be re-determined.

Table I-II lists the sample size requirement given reliability goal and number of allowable failures at various confidence levels using Eq (1) [45]. It can be observed that the required sample size increases with higher reliability goal or confidence level. The sample requirement for the case of zero failures varies from 5 (75% of reliability with 75% confidence) to 459 (99% of reliability with 99% confidence). In addition, the sample size increases significantly with the increased number of allowed failures during the test. For example, twenty-two single sample CI/HI readings from separate components are needed to demonstrate a reliability goal of 90% with 90% of confidence if no failure to correctly categorize faulted and un-faulted components is allowed. For the same target (90% of reliability with 90% confidence) the number of total number of single sample CI/HI readings from separate components increases to 38 if one failure is allowed during the test. The sample size increases further to 91 if 5 failures are allowed.

Due to the high cost associated with reliability demonstration of complex systems, it is highly desirable to reduce the required number of samples. One way of achieving the goal is for the system to include redundancy (such as on-board monitoring with multiple sensors). The risk of sensor failure can be mitigated through data fusion among the sensors and continuous/frequent monitoring, accordingly, may reduce the reliability goal. If there is a significant amount of time between early detection of CI/HI and final failure, the requirement of reliability of detection can be reduced. In addition, the existing knowledge of physics of failure, legacy fielded data, or sensor performance data obtained from developmental stage can be used to reduce the sample size through Bayesian Inference.

The reliability literature has several conflicting "rules of thumb" for sample size. One is that it is usually sufficient to test between 5 and 20 independent times; another is that you generally have to test at least 30 independent times. Military Handbook 781A[46] says that for reliability acceptance, one should test at least three independent times per lot and preferably 10%, up to a limit of 20, tests per lot. Because of such conflicting information, it is understandable that selection of a sample size requirement may be confusing. But the reason for lack of consistency in the rules of thumb is they are based on different assumptions about what constitutes acceptable confidence and reliability levels. Once these are established (a choice made with a view to defending oneself should a decision come under close public scrutiny), there is no longer any question what the sample size should be.

Additional sample size information may also be found in ASTM E 122-09[47] which provides numerous equations and examples for sampling processes and material lots.

I.4 DETAILED GUIDANCE

The following subsections provide detailed guidance for determining sample sizes to verify and validate CIs/HIs for use in condition based monitoring systems intended to replace or modify legacy maintenance inspections or TBOs.

I.4.1 Sample Size Method 1 To achieve a sufficient sample size for validation of CBM algorithms, one approach is to apply reliability criteria to the CI/HI. Sample size can be determined using binomial pass/fail criteria for reliability testing.

[45] Abernethy, B., Robert, “The New Weibull Handbook – Reliability & Statistical Analysis for Predicting Life, Safety, Survivability, Risk, Cost, and Warranty Claims,” Fourth Edition, 2000.
[46] MIL-HDBK-781, Department of Defense Handbook, “Reliability Test Methods, Plans, and Environments for Engineering, Development, Qualification, and Production,” April 1996.
[47] ASTM E 122-09, American Society for Testing and Materials, “Standard Practice for Calculating Sample Size to Estimate, with Specified Precision, the Average for a Characteristic of a Lot or Process,”, 1999.

This approach involves setting thresholds for faulted components (requiring mandatory maintenance); and un-faulted components (to include components with tolerable defects, for which maintenance is optional). Since it is vibratory CBM being discussed herein for transmissions and engines, there are also numerous faults to consider with different thresholds (e.g. bearing spall, chipped/scuffed gear teeth, and bent shafts). For each threshold, a binomial distribution must be established which means a separate set of component samples must be established for each verification/validation effort on each threshold. Also, each test for faulted and un-faulted components is assumed to result in a success or failure and the outcome on one test does not affect the outcome on another test. Finally, the configuration of the monitoring hardware (i.e. processor, wiring, sensors, sensor location, amplifiers) is assumed to be identical for each test.

An advantage to this approach is that a sample size can be established in advance, for any desired number of allowed failures, without detailed knowledge of how the CI/HI data is distributed.

A disadvantage to this approach is that the latest vibratory condition monitors installed on legacy US Army rotorcraft have not evolved to be the optimal monitor to capture all component failure modes on complex systems such as transmissions and engines. Further, replacing TBOs on legacy systems with On Condition monitoring may not be a valid option to consider unless "No Build Windows" are considered as viable. No Build windows would be implemented to avoid issues when fatigue critical components on a transmission or engine are involved. A No Build window refers to a depot process of not rebuilding an engine or gearbox incorporating a fatigue life limited component that is within a specified proximity range of the published retirement life.

Another disadvantage perceived to this approach is the relatively high cost and lengthiest time among the alternatives discussed within this Appendix. The costs and time are associated with the requisite sample size to achieve 90% confidence and 90% probability of detection (reliability). These percentage levels are used by the Aviation Engineering Directorate to assess risk. Further substantiation for the use of high confidence and reliability levels in aviation may be found in various statistical books, websites, and documents[48,49,50,51]. Per the reliability methodology, sample size can be calculated by using the following equations[52]:

$$\boldsymbol{n_1} = \ln(1\text{-confidence}) / \ln(\text{reliability}) = \text{faulted} \qquad \text{(Equation 1)}$$

$$\boldsymbol{n_2} = \ln(1\text{-confidence}) / \ln(\text{reliability}) = \text{un-faulted} \qquad \text{(Equation 2)}$$

48 Rees, DG, "Foundations of Statistics," CRC Press, 1987.
49 NIST/SEMATECH e, National Institute of Standards and Technology, "Handbook of Statistical Methods," Engineering Statistics Handbook, 2011.
50 Olson, E. and T. Olson, "Real Life Math: Statistics," Walch Publishing, 2000.
51 AIR 5113 Aerospace Information Report, Society of Aerospace Engineers (SAE), "Legal Issues Associated with the Use of Probabilistic Design Methods," 2002-06.
52 Abernethy, B. Robert, "The New Weibull Handbook – Reliability & Statistical Analysis for Predicting Life, Safety, Survivability, Risk, Cost, and Warranty Claims," Fourth Edition, 2000.

TABLE I-II. Sample size requirement for reliability demonstration via accept-reject tests

		Minimum Reliability Goal to be Demonstrated																	
Confidence Level		$R_{L1} = 0.75$			$R_{L1} = 0.80$			$R_{L1} = 0.85$			$R_{L1} = 0.90$			$R_{L1} = 0.95$			$R_{L1} = 0.99$		
$1-\beta$	$1-\alpha$	N	r	R_{U1}	N	r	R_{U1}	N	r	R_{U1}	N	r	R_{U1}	N	r	R_{U1}	N	r	R_{U1}
75%	75%	5	0	0.9441															
		10	1	0.9036															
		15	2	0.8837															
		20	3	0.8716															
		24	4	0.8576															
		29	5	0.8524															
80%	80%				8	0	0.9725	10	0	0.9779	16	0	0.9862	32	0	0.9931			
					14	1	0.9407	19	1	0.9564	29	1	0.9715	59	1	0.9861			
					21	2	0.9261	28	2	0.9447	42	2	0.9632	85	2	0.9819			
					27	3	0.9138	36	3	0.9355	54	3	0.9572	110	3	0.9791			
					33	4	0.9051	44	4	0.9291	66	4	0.9529	134	4	0.9769			
					39	5	0.8984	52	5	0.9241	78	5	0.9496	157	5	0.9751			
85%	85%				9	0	0.9821	12	0	0.9865	19	0	0.9915	37	0	0.9956			
					16	1	0.9569	22	1	0.9687	33	1	0.9792	67	1	0.9898			
					23	2	0.9413	31	2	0.9566	46	2	0.9709	94	2	0.9858			
					29	3	0.9285	39	3	0.9471	59	3	0.9651	119	3	0.9828			
					35	4	0.9191	47	4	0.9399	72	4	0.9611	144	4	0.9806			
					41	5	0.9116	55	5	0.9344	84	5	0.9673	169	5	0.9769			
90%	90%				11	0	0.9905	15	0	0.9931	22	0	0.9952	45	0	0.9977			
					18	1	0.9701	25	1	0.9785	38	1	0.9858	77	1	0.9931			
					25	2	0.9551	34	2	0.9671	52	2	0.9786	105	2	0.9895			
					32	3	0.9444	43	3	0.9588	65	3	0.9729	132	3	0.9867			
					38	4	0.9346	52	4	0.9525	78	4	0.9685	158	4	0.9845			
					45	5	0.9284	60	5	0.9466	91	5	0.9651	184	5	0.9828			
95%	95%				14	0	0.9963	19	0	0.9973	29	0	0.9982	59	0	0.9991			
					22	1	0.9836	30	1	0.9881	46	1	0.9922	93	1	0.9962			
					30	2	0.9722	40	2	0.9792	61	2	0.9865	124	2	0.9934			
					37	3	0.9622	50	3	0.9722	76	3	0.9818	153	3	0.9911			
					44	4	0.9541	59	4	0.9661	89	4	0.9776	181	4	0.9891			
					50	5	0.9464	68	5	0.9609	103	5	0.9743	208	5	0.9874			
99%	99%																459	0	0.999978
																	662	1	0.999775
																	838	2	0.9962
																		3	
																		4	
																		5	

n = Total Number of Single Sample CI/HI Readings on Separate Components
r = Number of Failed Sample CI/HI Readings
R_{U1} = Maximum Designed-In Reliability of the CI/HI
R_{L1} = Minimum Demonstrated Reliability of the CI/HI

Reliability is the desired probability that the system will correctly detect a fault (that is, probability of detection). Confidence is the desired probability that this reliability will be obtained. When using 90/90 confidence and reliability levels in sample size determination, Eq (1) and (2) each yield 22 samples for zero occurrences of incorrect classification of faulted and un-faulted items. The ramifications of this equation are that if it is required to have a vibratory CI/HI to demonstrate 90% reliability at a 90% confidence level, the fault must be correctly identified on 22 components with corresponding Tear Down Analyses (TDAs). In addition, 22 components without faults must maintain a false positive rate of less than 10%. These detections should be validated either on a validated test stand or on the actual aircraft. The CI/HI must demonstrate a 90% fault detection rate and a 90% correct classification of no fault conditions based on a threshold. If justification is provided for reduced reliability/confidence for specific component/fault conditions, the number of samples may be reduced. For example, if the confidence and reliability can be reduced to 80%, only 8 samples are required for faulted and un-faulted conditions using equations one and two.

Other methods may be investigated to determine their benefit in determining an appropriate sample size for extending or replacing legacy maintenance with vibratory CBM while still demonstrating high confidence and high reliability. If one is pursuing alternative approaches, it is important to step back and review how the CI/HI reliability and performance is determined prior to identifying it as a candidate for CBM.

I.4.2 Sample Size Method 2 The following example demonstrates one approach to assessing the ability of a CI to detect a component fault. This should be the first step in identifying the ability of a CI/HI to respond to a fault and replace a time based maintenance interval. The approach will be demonstrated by applying the process to one CI used to detect one type of fault on one specific component. It should be noted that the component used for this analyses is not a candidate for on condition maintenance and is discussed for demonstration purposes. The focus is on one component, one type of fault and one CI. The component used for this example is the input pinion in the nose gearbox of the AH64 helicopter. CI data before and after replacement of the nose gearbox (NGB) of eleven AH-64D helicopters with pitted pinion teeth will be analyzed.[53] During tear down analyses of the NGB, pitting damage was observed and documented on several of the pinion teeth. The CI for the pinion, referred to as the Sideband Index (SI) for the input gear, was recorded in the on-board health monitoring system when damage occurred and after replacement and will be used for this analysis. The Sideband Index (SI) is a measure of local gear faults. This CI is defined as the average sideband order of the fundamental gear meshing frequency. An increase in the magnitude of the sidebands of the fundamental gear meshing frequency indicates pinion tooth damage. A minimum sample size is required to answer the question - How many faulted components correctly detected by the CBM system are required for validation?

The approach applies a statistical analysis to a problem hypothesis statement. The general guidance of Section I.3 can be used as a reference on some of the statistical methods used in this approach. First, a hypothesis and test statistic must be defined to determine sample size. A hypothesis test is used to answer a question about the dataset. The question to be answered is "*Does the CI, input gear sideband index, respond to the failure mode, pitting on several pinion teeth?*" The *hypothesis* is a quantitative statement that states something about the *population* is true. *Samples*, sub-sets of the *population*, are typically used to evaluate the *hypothesis*. For this application, differences between two *populations* will be investigated. The *hypothesis* will be defined to determine if the CI for the

[53] Antolick, L., J. D. Branning, D. Wade, and P. Dempsey. "Evaluation of Gear Condition Indicator Performance on Rotorcraft Fleet," Proceedings of the American Helicopter Society 66th Annual Forum, Phoenix, Arizona, May 11-13, 2010.

"damaged" component (*pitting damage on two or more pinion teeth)* are *significantly different* than the CI values of an "undamaged" *(no pinion teeth damage)* component.

A test statistic is used to make a decision about the sample data set. The test statistic selected is dependent on the hypothesis and statistical characteristics of the data. The t test is the test statistic selected for this analysis. It will be used to compare the CI mean values from the damaged and undamaged gears to decide if they are statistically different. The test statistic is used to determine if there is enough evidence to reject or fail to reject the *null hypothesis* (H_o). The alternative is referred to as H_1. The test statistic selected for this analysis will determine if the CI values from the damaged component are significantly different from the component with no damage:

H_o: $\mu_1 = \mu_2$ (CI values for the damaged and undamaged gear respond the same)
H_1: $\mu_1 \neq \mu_2$ (CI values for the damaged and undamaged gear respond differently)

Where:
μ_1=*population* mean of undamaged gear teeth
μ_2=*population* mean of gear teeth with pitting damage on two or more pinion teeth

If we reject the null hypothesis (H_o) the means are not equal and CI values differ significantly. These differences will enable differentiation of a damaged and undamaged gear based on the response of the CI to the damaged and undamaged gear states.

Since the test statistic selected is dependent on statistical characteristics of the data, the CI data and a histogram of the eleven tails in the time interval prior to removal and after replacement are plotted in Figures I-2 through I-12. The time that the gear damage occurred within the replacement interval was unknown. This means that the gears could have been undamaged during collection of the CI data within the replacement interval. For this reason, a threshold of 2 was defined to indicate the start of damage to a gear tooth. Once the CI was found to be equal to or greater than 2, from that time forward the data was used for the damage state of the gear. Figure I-13 is a plot of the CI and histogram for all eleven helicopters with this method applied to the damage dataset.

Review of Figures I-2 through I-13 found the CI, Sideband index increased when the gear was in the damaged state in the time period prior to replacement. Several observations can be made after reviewing Figures I-2 through I-13. In Figure I-7 the CI did not appear to respond to the damaged gear as well as CI values measured on the other helicopters. This could have been due to the type or level of damage that occurred on these pinion teeth was less than the damage observed on the other pinion teeth prior to NGB replacement.

The tear down analyses of these eleven helicopters are currently under review and a damage level factor will be defined prior to the next revision of this handbook. Another observation is that in Figure I-12, only two data points were available after replacement. Due to this limited data, the tail plotted in Figure I-12 was not used for further analysis bringing our dataset down to 10 helicopters. As additional data points are collected for these helicopters they will be added to future analyses.

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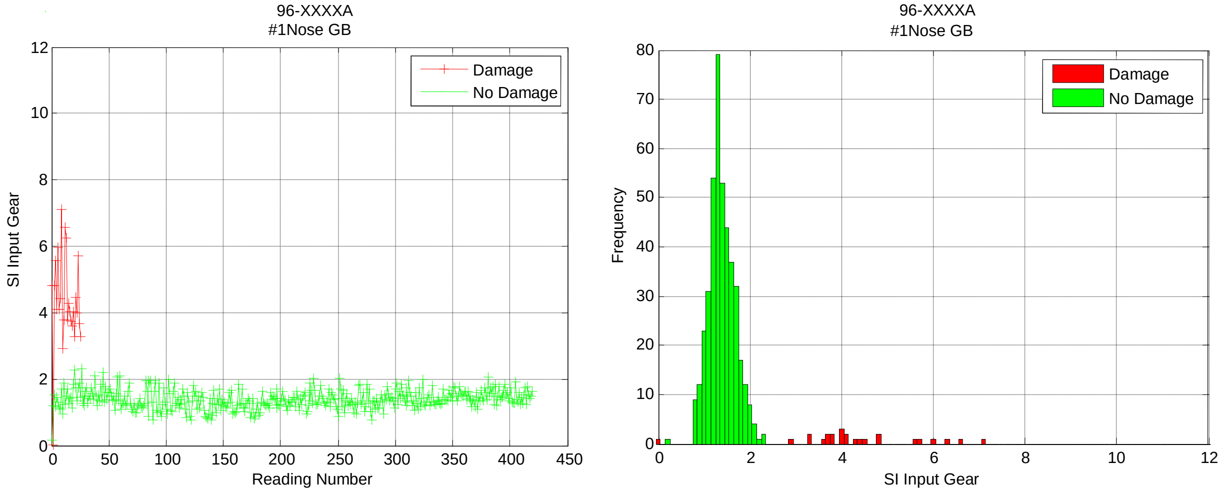

FIGURE I-2: CI data before and after gear replacement

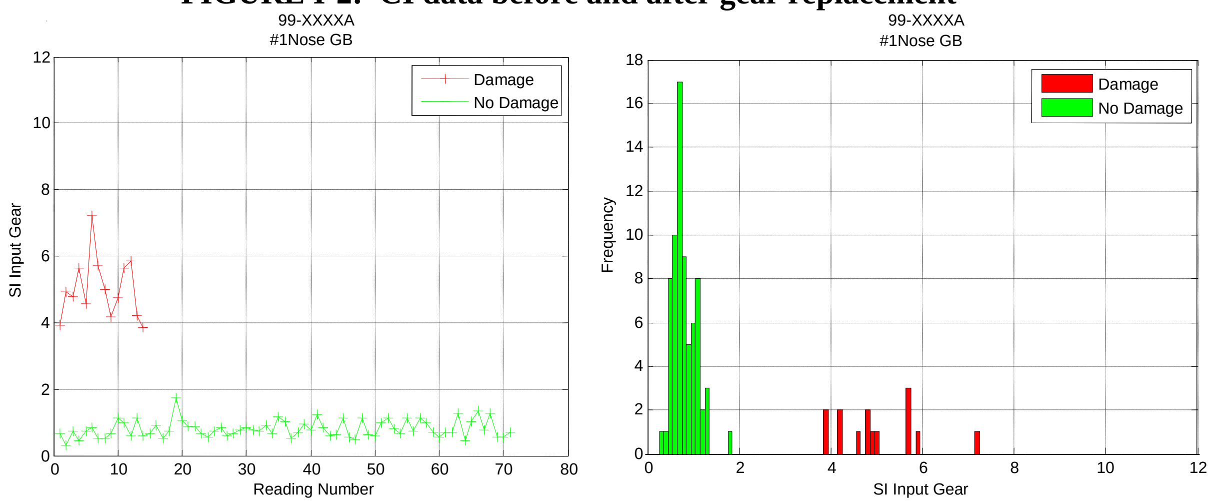

FIGURE I-3: CI data before and after gear replacement

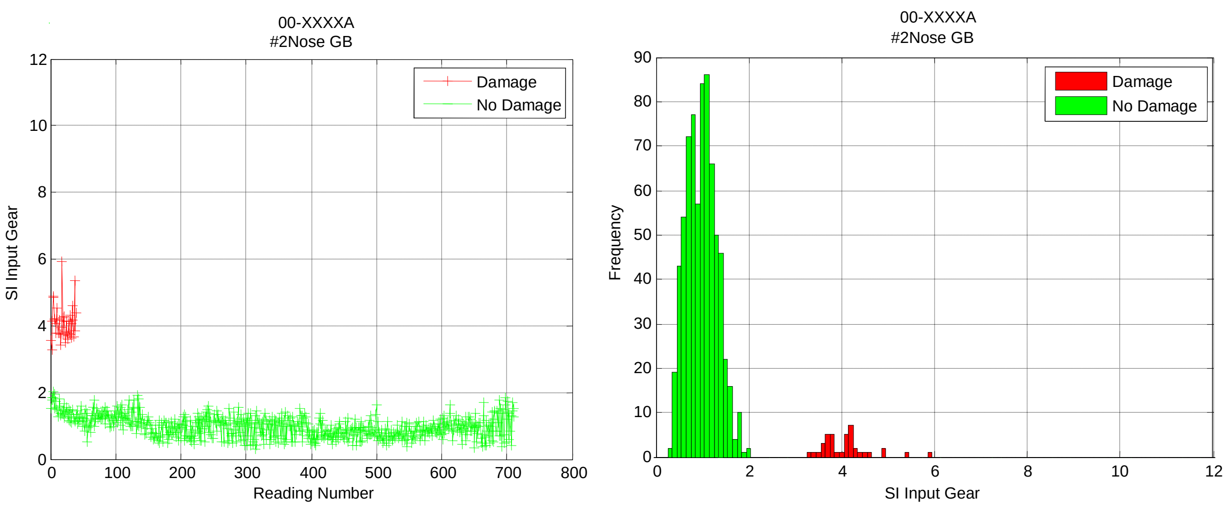

FIGURE I-4: CI data before and after gear replacement

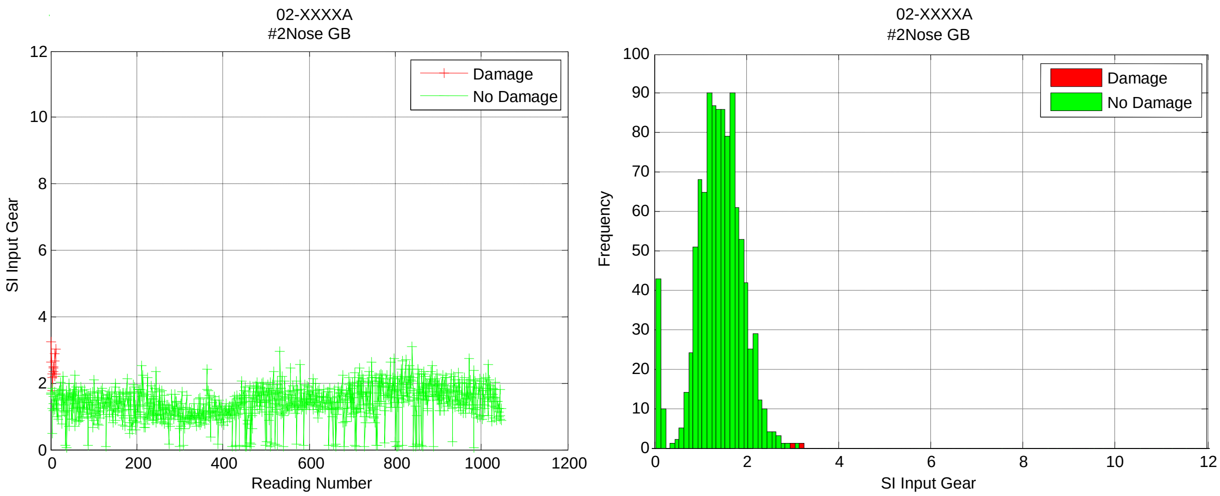

FIGURE I-5: CI data before and after gear replacement

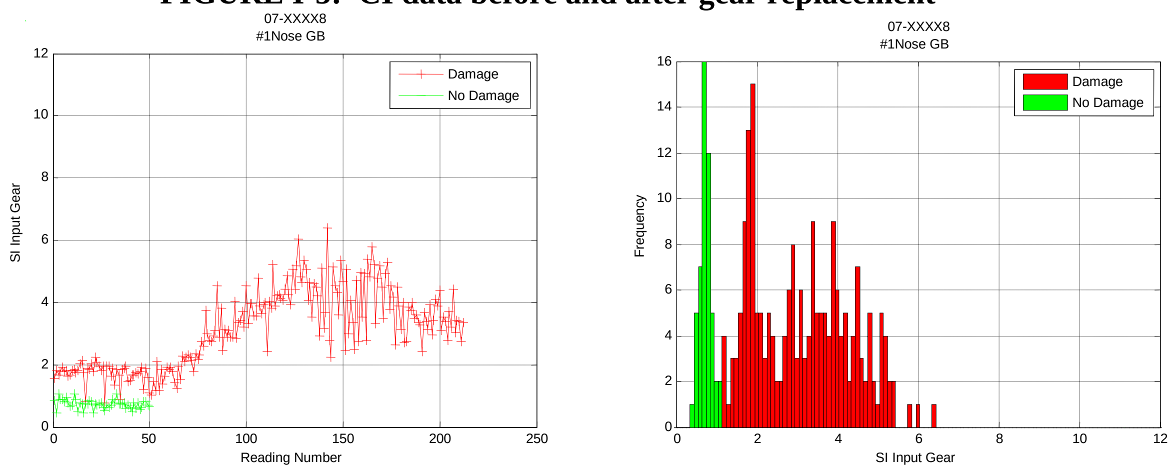

FIGURE I-6: CI data before and after gear replacement

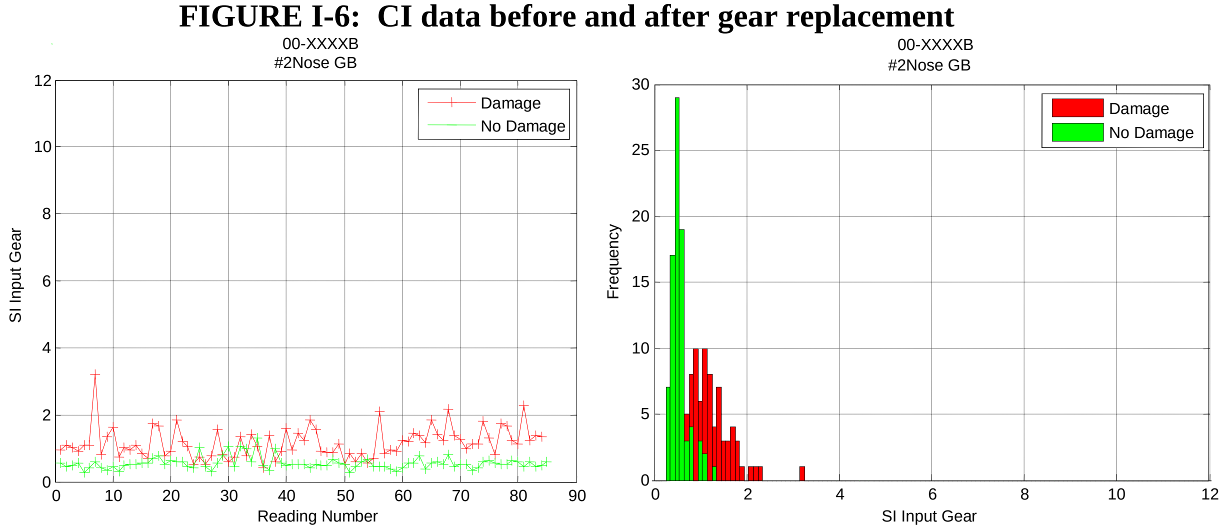

FIGURE I-7: CI data before and after gear replacement

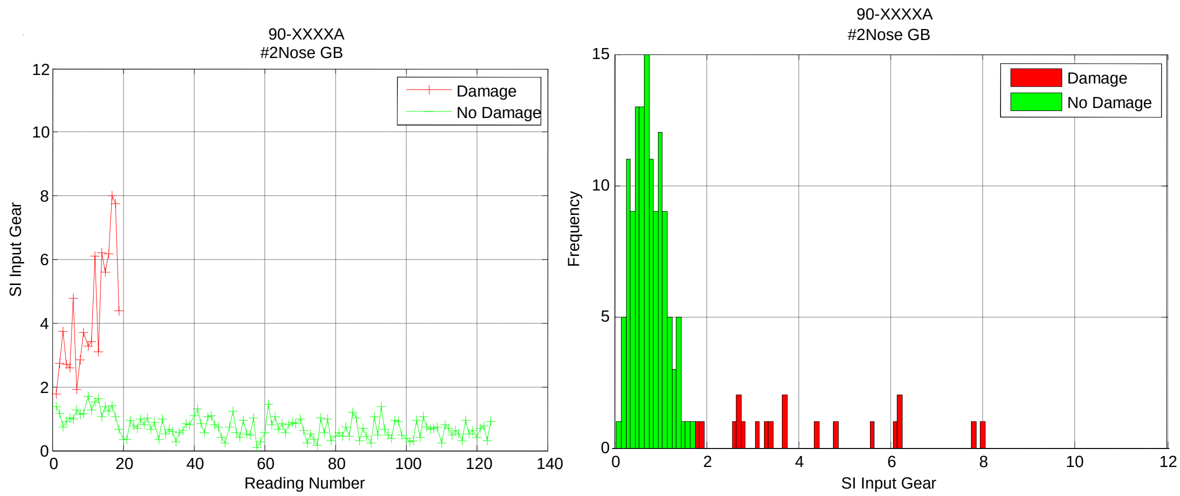

FIGURE I-8: CI data before and after gear replacement

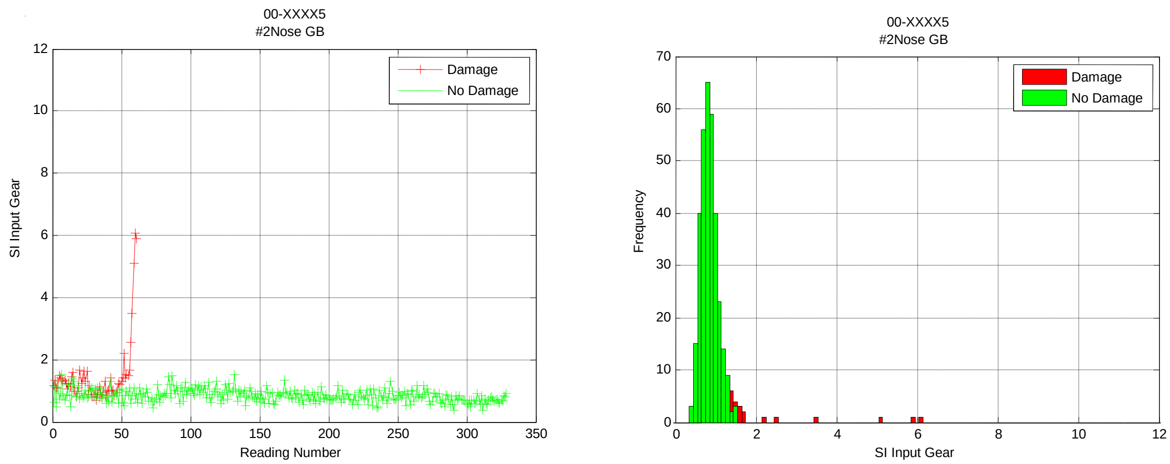

FIGURE I-9: CI data before and after gear replacement

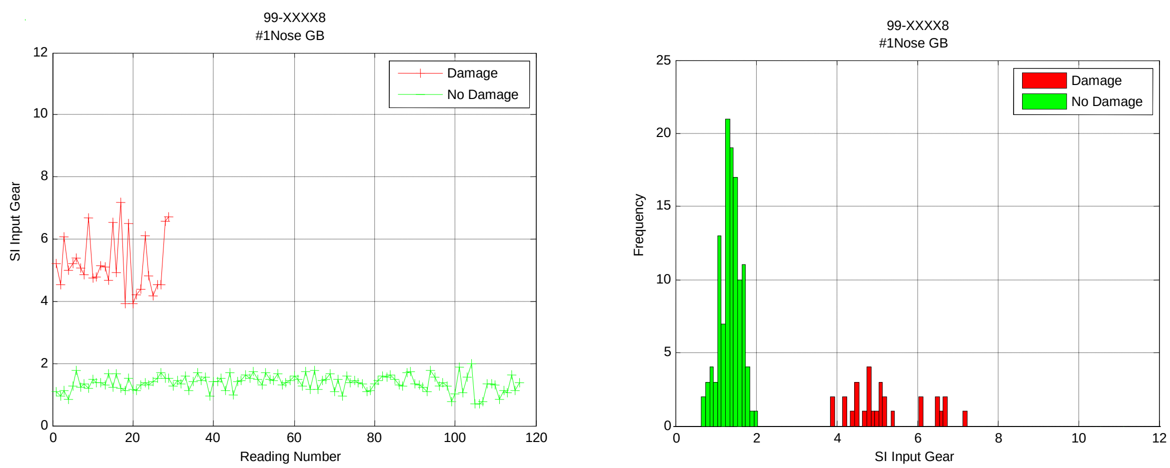

FIGURE I-10: CI data before and after gear replacement

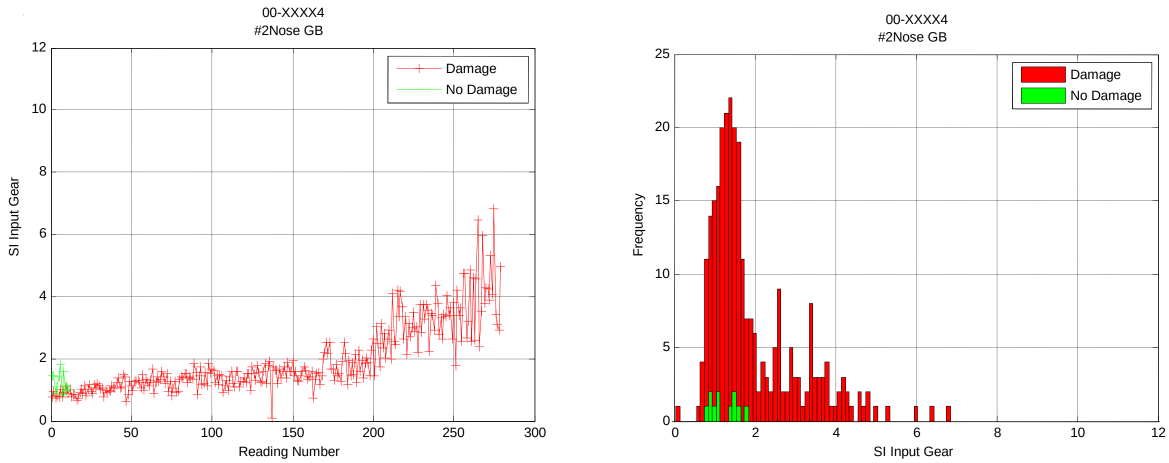

FIGURE I-11: CI data before and after gear replacement

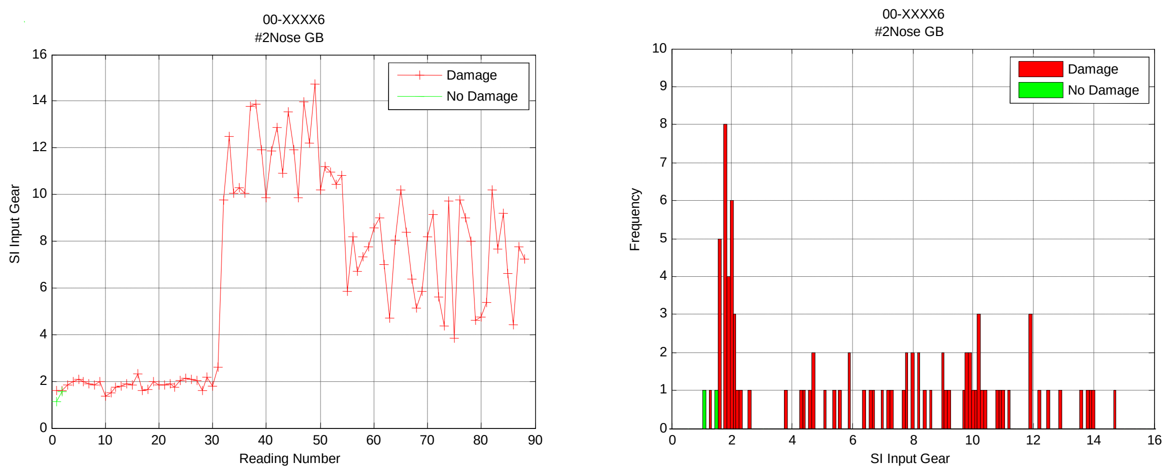

FIGURE I-12: CI data before and after gear replacement

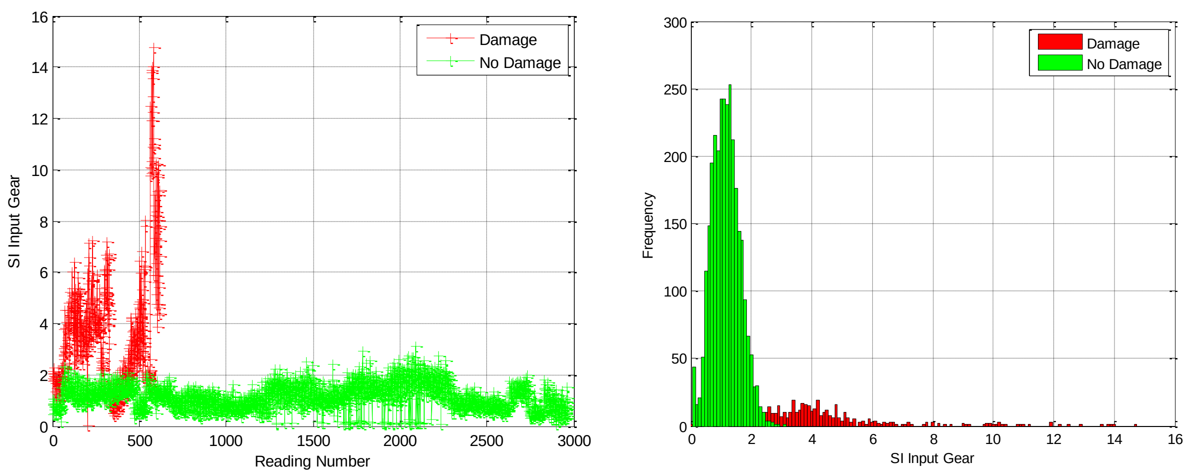

FIGURE I-13: CI data before and after gear replacement of all 11 helicopters

A two-sample t test is commonly used to test for the equality of means from two distributions. However, this test assumes that both distributions are normal (also, it usually assumes that both distributions have equal variances, in which case the means are equal if both distributions are subsets of the same normally distributed population).[54] Based on the histogram date, the CI distribution does not appear to follow a normal distribution. The Lilliefors normality test was applied to all the CI data plotted in Figure I-13, before removal and after replacement, and found that neither distribution passed the normality test.[55]

For this reason, the non-parametric Wilcoxon Rank Sum test was used, which tests the null hypothesis that the two distributions are identical against the alternative hypothesis that the two distributions differ only with respect to the median.[56] Per this test, the null hypothesis was rejected at a significance level of .05. CI values for the damage and no damage data sets differ significantly. The significance level of .05 or Type I error indicates at 5% chance of rejecting the null hypothesis when it is true. Type I and Type II error rates are described in more details in Table I-III. For this example, Figure I-14 provides a block diagram of the analysis steps required prior to defining a sample size.

TABLE I-III: State of system—health of component

		***H_o* is True** $CI_{damaged} = CI_{undamaged}$ No Damage	***H_o* is False** $CI_{damaged} \neq CI_{undamaged}$ Damage
Decision	**Reject H_o** Indicate Damage	False Positive **(FP)** (α = Type I Error = significance = p-value) False Alarms	True Positive **(TP)** (1-β or power) Hits
	Fail to reject H_o Indicate No Damage	True Negative **(TN)** (1-α) Correct Indication	False Negative **(FN)** (β = Type II Error) Missed Hits

Typical statistical analyses for determining sample size are based on the assumption that data follows a normal distribution. One method uses the t statistic to test the hypothesis. Sample size for a hypothesis test using a t statistic can be defined determining the minimum sample size that provides a t statistic that is at the minimum critical region of the distribution[57]. Can the t statistic be used to determine a sample size if the data is non-parametric? Lehman[58] determined the efficiency of the Wilcoxon rank sum test when compared to the two sample t test. Conover[59] also defined the asymptotic relative efficiency. Both found the maximum loss of efficiency

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[54] Montgomery, Douglas C. Design and Analysis of Experiments. New York: Wiley, 1991.

[55] Conover, W.J Practical Nonparametric Statistics. New York, NY: J. Wiley and Sons, 1999.

[56] Pappas, P. and DePuy, V., "An Overview of Non Parametric Tests in SAS®: When, Why, and How," Paper TU04, Duke Clinical Research Institute, Durham, North Carolina, USA, 2004

[57] NIST/SEMATECH e, National Institute of Standards and Technology, "Handbook of Statistical Methods," Engineering Statistics Handbook, 2011.

[58] Lehmann, E. and H. D'Abrera, "Nonparametrics: Statistical Methods Based on Ranks," Upper Saddle River, NJ: Prentice Hall, 1998.

[59] Conover, *Practical Nonparametric Statistics*, New York, NY: J. Wiley and Sons, 1999.

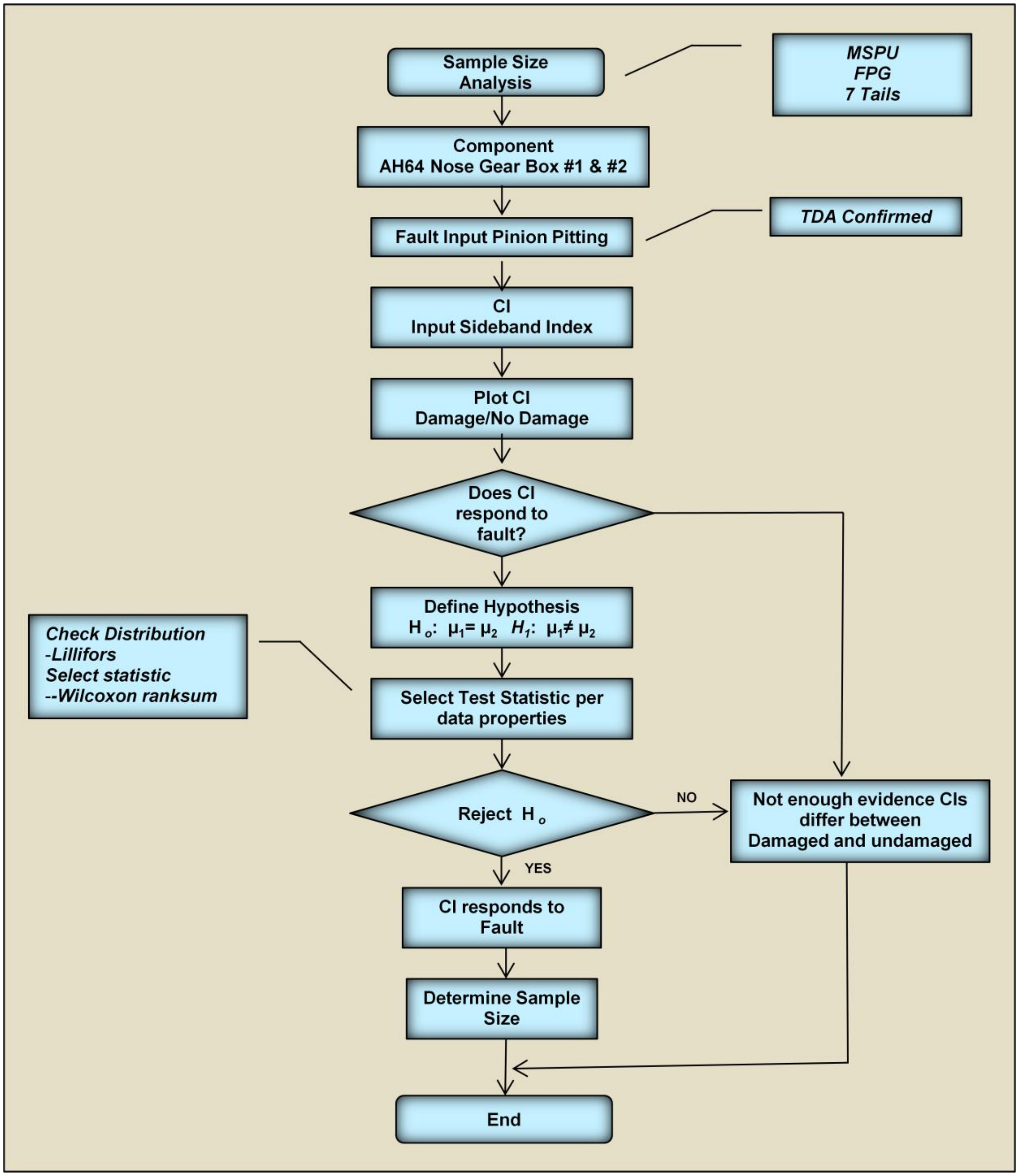


FIGURE I-14: Block diagram of sample size analysis steps

when using the Wilcoxon test instead of the t test was 0.864[60], indicating the sample size determined by the t test divided by the efficiency will determine the comparable sample size

60 Hollander, Myles, and Douglas A. Wolfe. *Nonparametric Statistical Methods*. New York: John Wiley & Sons, 1999.

required for the non normal distribution. Using the statistical properties of the CI data before replacement, when the gear was damaged, and after replacement, a t test can be used for determining the minimum sample size to reproduce the results of the hypothesis test. The t statistic is calculated as:

$$t = \frac{(\overline{y_1} - \overline{y_2})}{S_p \sqrt{\frac{1}{n_1} + \frac{1}{n_2}}} \qquad \text{(Equation 3)}$$

Where
$\bar{y}_1$*=sample* mean of undamaged gear teeth
n_1= *sample* size of undamaged gears
$\bar{y}_2$ =*sample* mean of gear teeth with pitting damage on two or more pinion teeth
n_2= *sample* size of damaged gears
Sp = pooled or average standard deviation

Using the pooled standard deviation estimated within the population, sample size is determined by setting the t statistic equal to the critical t values. Solving for *n* requires iterating through several *n* values to determine the sample size that solves the equation:

$$[t(1-\frac{\alpha}{2};\ n_1 + n_2 - 2) + t(1-\beta;\ n_1 + n_2 - 2)] = \frac{(\overline{y_1} - \overline{y_2})}{S_p \sqrt{\frac{1}{n_1} + \frac{1}{n_2}}} \qquad \text{(Equation 4)}$$

A minimum sample size for *n* using the data from ten helicopters was calculated to be 7.75 from these iterations. Due to the non-normality of data distribution, the minimum sample size must be 14% larger than the minimum sample size defined by the t test (as previously discussed above with the maximum loss of efficiency at 0.864). Therefore, 7.75/.864 = approximately 9 samples for minimum sample size when using the non normal distributed data from the NGB CIs.

Per Table I-I, α is equal to the type I error rate, reject null hypothesis when null hypothesis is true. For this hypothesis it means to claim a significant difference in means when there is not. It should be noted that $\alpha/2$ is used for this hypothesis because $\mu_1 \neq \mu_2$ could result in $\mu_1 > \mu_2$ or $\mu_1 < \mu_2$. Also per Table I-I, β is equal to type II error rate, fail to reject null hypothesis when null hypothesis is false (missed hit). For this hypothesis it means to claim no discernable difference in means when differences exist.

Per the CI data from both distributions, with $\alpha = 0.05$ and $\beta = 0.05$, a sample size of 9 undamaged gears (***n_1***) and 9 damaged gears (***n_2***) are required. However, if the data from the helicopter shown in Figure I-7, with the CI that responded poorly to the damage, was removed from the dataset, bringing our dataset down to 9, only 7 damaged and undamaged gears would be required. This means that a minimum of CI data from 7 faulted components and 7 unfaulted components are required to confirm the CI can differentiate between faulted and unfaulted gears. This exercise indicates the importance of assessing the overall performance of the CI across the fleet, including false negatives and false positive indications, when determining sample sizes for going "on-condition." As more data is acquired on damaged components to add to the distributions of the damaged and undamaged data sets, this number will change.

The example described provides a method to determine the minimum sample size required to evaluate the response of a CI to a specific fault. However, the example does not indicate if the CI responds with high confidence and reliability. To determine if the CI meets the 90% reliability metrics per equations 1 and 2, additional steps are required. The first step is to define an optimum threshold. This allows one to separate the data into the two conditions, faulted and unfaulted. A curve, referred to as Receiver Operating Characteristic (ROC), can be used for defining this optimum threshold.[61] ROC curves are plots of the false alarm rate (probability of false alarm or false positive rate) on the horizontal axis (x) versus the hit rate (probability of detection-true alarm or true positive rate) on the vertical axis (y), providing a visual comparison of thresholds on a common scale. Since this is for demonstration purposed only, the data from helicopters plotted on the following figures will be used: I-2, I-3, I-4, I-5, I-6, and I-8. The ROC curve for this dataset is shown in Figure I-15. Figure I-15 also provides a visual illustration of how the ROC plot is obtained. The two distributions represent a no damage response and a damage response of a CI. The threshold line separates the graph into correct indication/TN-true negative (no damage—no indication), FN-false negative (damage present—no indication), false alarms/FP-false positive (no damage—indicated) and hits/TP-true positive (damage—indicated). The probability of detection would equal the area under the damage distribution curve to the right of the threshold line. The false alarm rate would equal the area of the no damage distribution to the right of the threshold line. The optimum threshold for this CI, component and fault is 1.7. To determine the reliability of this system, the number of CI readings in the damaged dataset that exceed 1.7 is divided by the total readings. This determined the probability of detection equal to 92%. The false alarm rate was calculated by determining the number of readings in the undamaged dataset when the CI value exceeded 1.7. This determined the probability of correctly classifying an undamaged component as 88% or a false alarm rate of 12%. This CI does not meet the minimum demonstrated reliability of 90% for correctly classifying an undamaged component. Note the number of samples required to assess the performance of the CI must also meet confidence and reliability metrics.

I.4.3 Sample Size Method 3

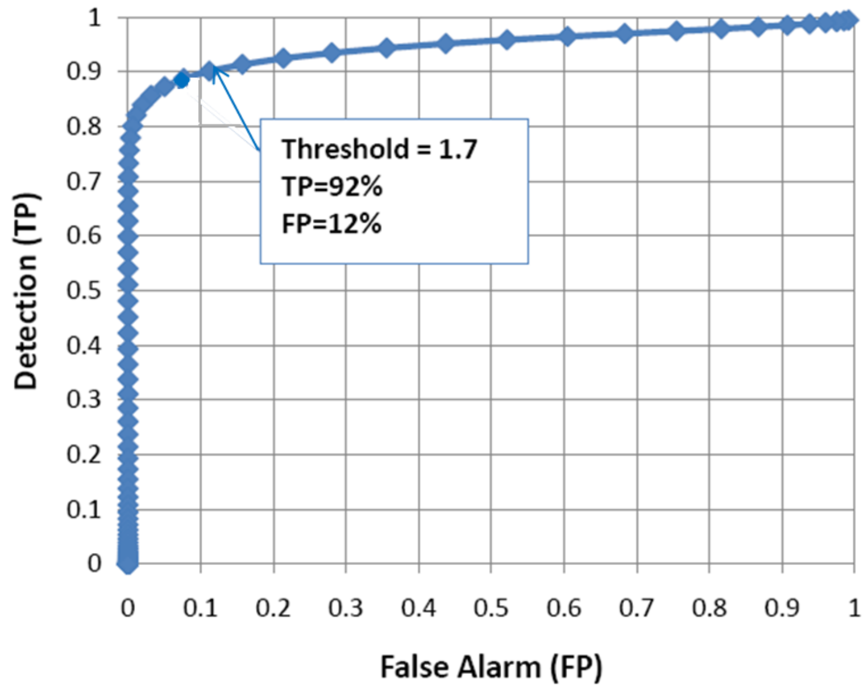
I.4.3.1 Weibull Analysis[62]

The sample size methodology in this section evaluates the distribution of CI data to see if the data may be incorporated into a Weibull distribution. For a CBM-monitored component, if the distribution of current CI values from all items is Weibull, then a method exists to establish

[61] Dempsey, Paula, J.; Jonathon, A. Keller, Daniel R Wade, *Signal Detection Theory Applied to Helicopter Transmission Diagnostic Thresholds*, NASA/TM-2008-215262; AMRDEC PAO Control Number FN 3597, July 2008.

[62] Abernethy, Robert B., *The New Weibull Handbook – Reliability & Statistical Analysis for Predicting Life, Safety, Survivability, Risk, Cost, and Warranty Claims,* Fourth Edition, 2000.

CI thresholds that meet the specified reliability and confidence requirements. Note also, in this example, the colors green, yellow, and red are utilized to describe different CI thresholds.



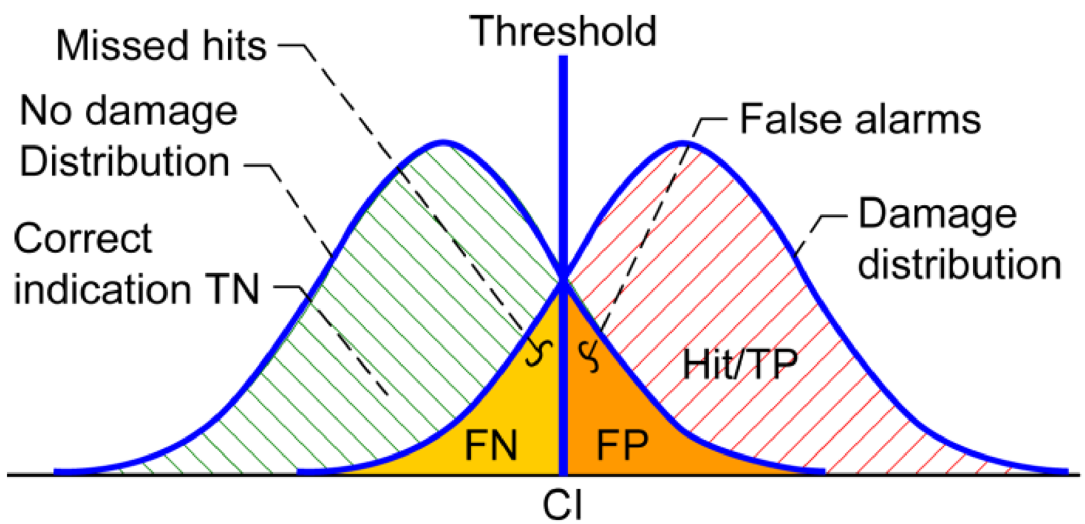


FIGURE I-15: ROC curve

Weibull analysis is a tool common to life data analysis, where operating times and failure times of individual items are analyzed and Weibull distribution parameters are identified to characterize the failure mode(s) of interest. In particular, the value of the Weibull slope parameter, β, indicates whether the failure rate is decreasing ($\beta < 1$), constant ($\beta = 1$), or increasing ($\beta > 1$). Applied to CBM, if the assumption is made that an increase in CI value (magnitude) corresponds to an increase in the probability of the presence of a fault ($\beta > 1$), then Weibull analysis should provide a reasonably accurate assessment of CI data and component health.

I.4.3.2 Minimum Sample Size Derivation

The cumulative distribution function for the two-parameter Weibull distribution is given by:

$$F(x) = 1 - e^{-\left(\frac{x}{\eta}\right)^{\beta}} \qquad \text{(Equation 5)}$$

Where:
x = the CI value to assess
β = the Weibull slope (or shape) parameter
η = the Weibull scale parameter

Applied to CBM, *F(x)* is the probability of encountering a fault up to a CI value of *x*. The complement of *F(x)* is reliability, *R(x)*, the probability of not encountering a fault up to a CI value of *x*:

$$R(x) = e^{-\left(\frac{x}{\eta}\right)^{\beta}}$$

(Equation 6)

If the natural logarithm is taken of both sides of ***Equation 6***, then we can solve for *x*:

$$ln(R(x)) = ln\left(e^{-\left(\frac{x}{\eta}\right)^{\beta}}\right) = -\left(\frac{x}{\eta}\right)^{\beta}$$

$$x = \eta\left[-ln(R(x))\right]^{\left(\frac{1}{\beta}\right)}$$

(Equation 7)

Using ***Equation 7***, we know β and η from the Weibull distribution, and we are interested in knowing the threshold CI value, X_0, where $R(X_0)$ = 90%, the required reliability at X_0. So the threshold CI value is given by:

$$X_0 = \eta\left[-ln(R(X_0))\right]^{\left(\frac{1}{\beta}\right)}$$

(Equation 8)

Or, rearranging the terms of ***Equation 8***, we are able to express η in terms of the threshold CI value and its associated reliability:

$$\eta = X_0\left[-ln(R(X_0))\right]^{-\left(\frac{1}{\beta}\right)}$$

(Equation 9)

If a null hypothesis H_0 is formed by assuming $R(x)$, the probability of not encountering a fault up to a CI value of x, is less than the required reliability of 90%, then N, the minimum number of fault-free items with a CI value = x needed to reject H_0 at the $1\text{-}C$ significance level, is given by:

$$N = \frac{ln(1-C)}{ln\big(R(x)\big)}$$

(Equation 10)

Where:
C = the required one-sided lower confidence bound to reliability
x = the CI value to which we test

If we substitute ***Equation 6***, the Weibull reliability equation, into ***Equation 10***, then the minimum sample size when faulted CI values follow a Weibull distribution is given by:

$$N = -ln(1-C)\left(\frac{\eta}{x}\right)^{\beta}$$

(Equation 11)

Finally, recall ***Equation 9***, where η is expressed in terms of the threshold CI value and its associated reliability. Substituting ***Equation 9*** into ***Equation 11*** and simplifying gives the minimum number of fault-free items with a CI value = x needed to demonstrate the required reliability $R(X_0)$ at X_0 with the required confidence bound C:

$$N = \left(\frac{ln(1-C)}{ln\big(R(X_0)\big)}\right)\left(\frac{X_0}{x}\right)^{\beta}$$

(Equation 12)

There are several noteworthy things about ***Equation 12***:

a. As mentioned earlier, for CBM applications, β should always be greater than one ($\beta > 1$). If β is found to be less than or equal to one, then we can conclude that the associated CI is a poor indicator of component health.

b. The CI value we are testing each item to, x, must always be greater than or equal to our threshold X_0.

c. When $x = X_0$, ***Equation 12*** reduces to ***Equation 10***, which provides a sample size based on the binomial distribution (with N successes out of N trials) that is independent of the CI distribution.

d. Most importantly, this method relies on N false positives to demonstrate the required reliability at X_0. And when x, the CI value we test to, is allowed to increase above our threshold X_0, the required sample size N decreases.

I.4.3.3 CBM Thresholds and Treatment of CI Data The same set of CI values is used to demonstrate reliability at the green-yellow threshold, X_G, and at the yellow-red threshold, X_R. However, CI values from true yellow items are treated differently at X_G and X_R. At X_G, the Weibull analysis treats CI values from true green items as right-censored data, while CI values from true yellow (optional maintenance) or true red (mandatory maintenance) items are treated as faulted data. At X_R, using the same data, the Weibull analysis treats CI values from true green or yellow items as right-censored data, while only CI values from true red items are treated as faulted data.

(Censored data exists when the value of an observation is only partially known. A right-censored CI value is the CI value for an item that has yet to incur a fault. We do not know at what CI value a fault will occur; we only know the most recent healthy CI value.)

For either threshold, if a Weibull distribution is found to provide an acceptable fit, then the value of the lower confidence bound at 90% reliability can be read directly from the Weibull plot. This value is the threshold. If the threshold is too low, an unacceptable number of false positives will occur. To remedy this, we can select a higher CI value as a potential threshold, setting it equal to X_0 in ***Equation 12***. Once a CI value to test to is chosen (x), we can use ***Equation 12*** to determine the minimum sample size required to demonstrate 90% reliability at the potential threshold.

Many monitored components have yet to experience a true yellow or red item. When there are only CI values from unconfirmed green items, a Weibayes analysis may provide useful insight into the selection of the green threshold. A Weibayes analysis is the same as a Weibull analysis, but with an assumed β slope parameter. A known β from a similar CI, or a known β from the same CI for a similar component, may provide a reasonable estimate of β for our component/CI being analyzed. However, if no basis for an estimate exists, assume β to be equal to 1.1. When Weibayes analysis is applied to reliability demonstration and minimum sample size calculations, assuming $\beta = 1.1$ is considered best practice. Using this value for β acknowledges that a positive correlation should exist between CI value and probability of a fault, while at the same time provides conservatism to the calculation of the minimum sample size required.

To perform a Weibayes analysis at the green threshold, there will only be right-censored data (the CI values from unconfirmed green items) and our assumed β. The value of the lower confidence bound at 90% reliability can be read directly from the Weibayes plot. As green items in the fleet continue to age, it is expected this lower confidence bound will increase. While it is ill-advised to establish the green threshold based solely on Weibayes results, this technique does provide useful information about the fleet of healthy items. Finally, nothing can be ascertained about the red threshold with only green items. However, if green and yellow items exist, then we can apply Weibayes in a similar manner to provide useful information about the fleet as items begin to wear to the point where faults are present.

I.4.4 Case Study Using Sample Size Method 3: Analysis of the Input Data Algorithm 1 (DA1) CI for the Apache Nose Gearbox (NGB) The Apache NGB is not a candidate for on condition maintenance. This component was selected for our case study because a large amount of CBM data is available for it. The fault detected involves pitting damage on gear teeth of the NGB spiral bevel gears. The CI used is Input DA1. This CI was recorded from the time when damage was thought to have occurred until the suspect gears were removed. An increase in the magnitude of the CI value indicates a fault. After TDA to confirm damage, new gears were installed and the CI was again recorded to confirm a healthy component.

A total of 16 NGBs are included in this study. The TDA results on suspect gears for these NGBs are 4 green, 2 yellow, and 10 red. In our analysis, CI values are categorized according to the color of the NGB from which they were recorded. CI values recorded after gear replacement are all considered green.

Three separate analyses were performed with different subsets of the CI data. Dataset 1 is comprised of the last recorded CI value before gear replacement and the first recorded CI value after gear replacement, for each NGB. After recognizing that multiple CI recordings may be taken on a single day, the maximum CI values recorded the day before gear replacement and the day after gear replacement were identified for each NGB. This is Dataset 2. Finally, the maximum CI value recorded over the entire suspect gear interval was identified for each NGB, along with the maximum CI value recorded the day after gear replacement. This comprises Dataset 3.

Figure I-16**,** below, provides the Weibull plots for both the green and red thresholds using Dataset 1. As it turns out, these thresholds are essentially identical, and the same occurrence is observed using Dataset 2 and Dataset 3. A yellow category does not exist for Input DA1 CI data for the Apache NGB. Therefore, only the green threshold will be considered for the remainder of this case study. Summary statistics for both thresholds are given in Table I-IV at the end of this section.

Figures 17, 18, and 19 provide the Weibull plots for Datasets 1, 2, and 3. For each figure, faulted CI values (yellow and red) are depicted on the plot, but right-censored CI values (green) are not. The straight line is the Weibull fit, and the curved line to the left is the 90% lower confidence bound. The CI value that provides 90% reliability with 90% confidence is found by locating the point where the line reliability = 90 (*y*-axis) intersects the lower confidence bound line, then reading the point's value from the *x*-axis (Input DA1).

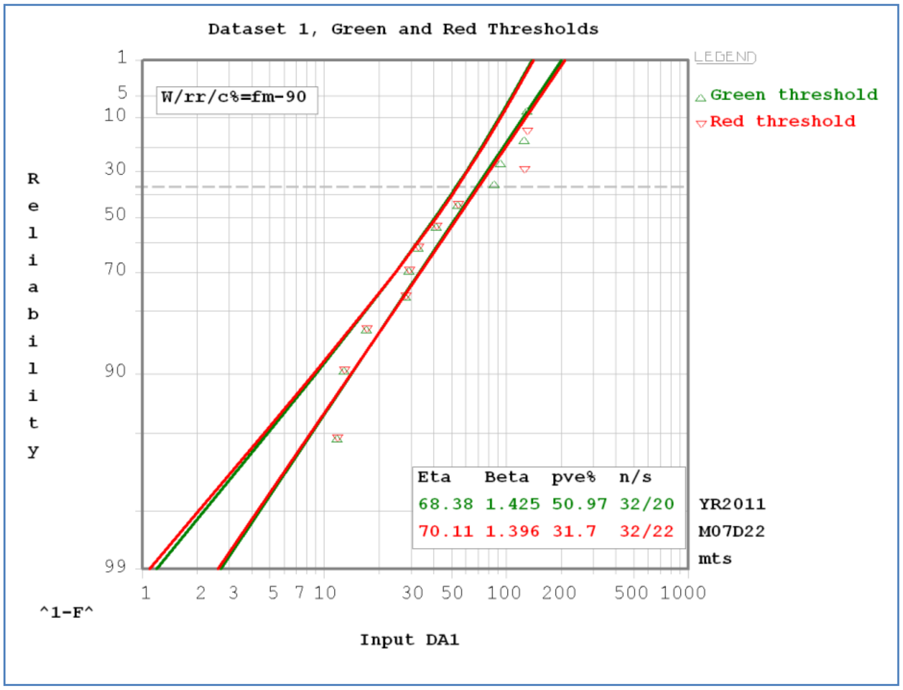


FIGURE I-16 Dataset 1, Weibull plot with green and red thresholds[63]

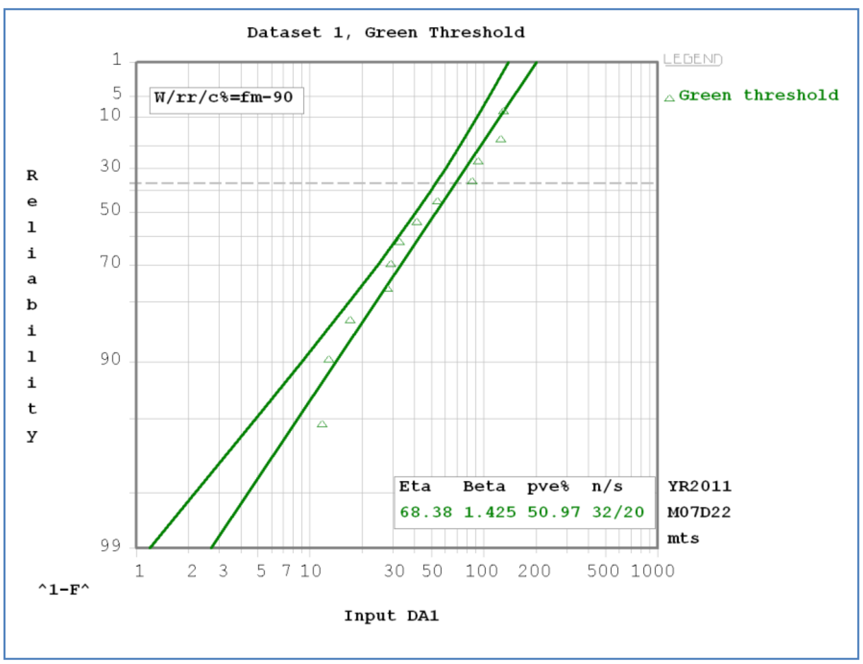


FIGURE I-17. Dataset 1, Weibull plot with green threshold

[63] W/rr/c% = fm-90 = Weibull/Rank Regression/Confidence Percent is equal to Fisher Matrix for 90% confidence
pve = P Value Estimated
n/s = Number of data points / Number of Healthy (green) data points
^1-F^ = Reliability

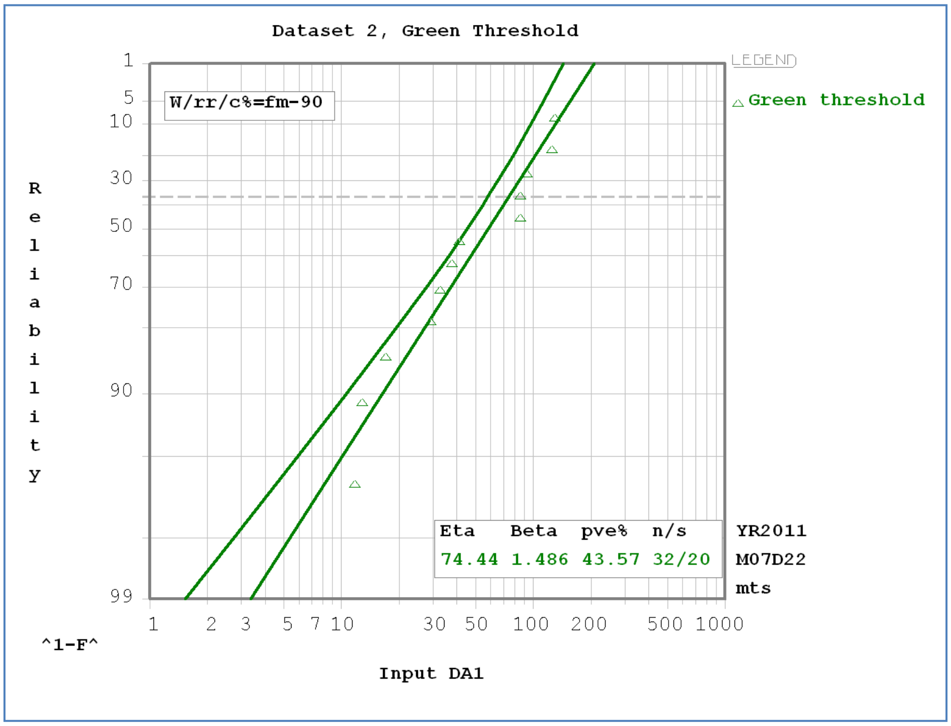


FIGURE I-18. Dataset 2, Weibull plot with green threshold

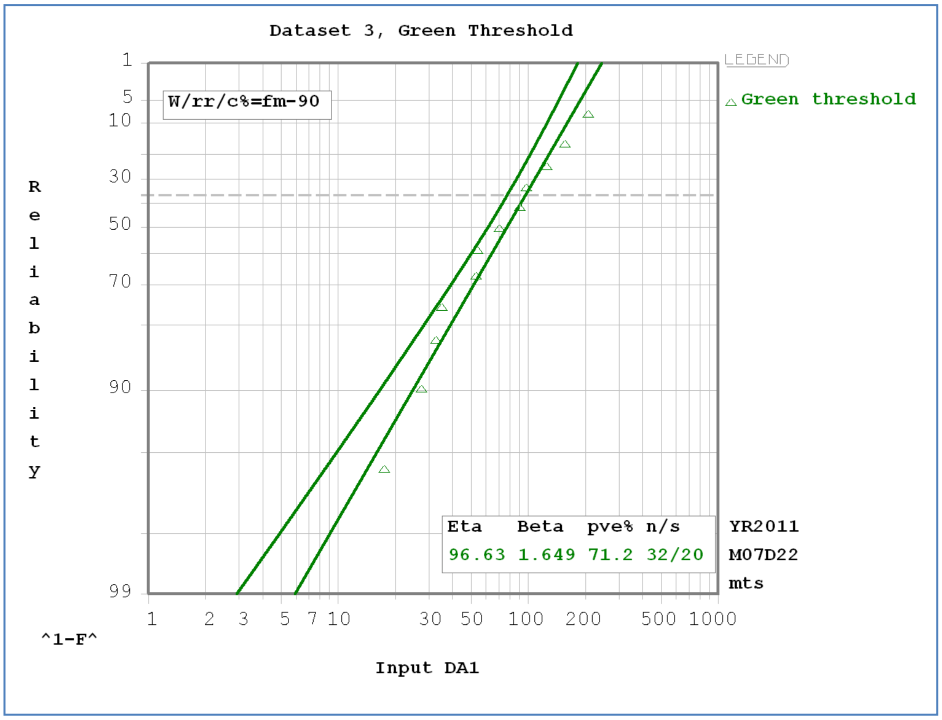


FIGURE I-19. Dataset 3, Weibull plot with green threshold

A summary of the results of the Weibull analyses is provided in Table I-IV. Note that any p-value estimate greater than 0.10 indicates an acceptable fit for a Weibull distribution. An r^2 value close to 1.0 also indicates an acceptable fit.

TABLE I-IV. Weibull Results Summary

Data	Before Gear Replaced	After Gear Replaced	Threshold	β	η	p-value Estimate	r^2	Acceptable Fit?	Threshold CI
Data set1	last CI recorded	first CI recorded	green	1.43	68.38	0.51	0.95	yes	8.95
			red	1.40	70.11	0.32	0.92	yes	8.71
Data set2	last day max CI	first day max CI	green	1.49	74.44	0.44	0.94	yes	10.62
			red	1.44	77.51	0.35	0.92	yes	10.26
Data set3	overall max CI	first day max CI	green	1.65	96.63	0.71	0.97	yes	16.50
			red	1.67	96.58	0.59	0.95	yes	16.65

To illustrate how sample size is determined, suppose we wish to demonstrate 90% reliability with 90% confidence for our green threshold = 20.0. With the β values from each dataset, we can compute sample sizes using ***Equation 12*** for various CI values ≥ 20.0. Minimum sample sizes without encountering a fault for each β at different "test to" CI values are provided in Table I-V.

TABLE I-V. Minimum sample sizes for CIs

Reliability = 90%
Confidence = 90%
Green Threshold = 20.0

CI To Test To	Minimum Sample Size		
	Dataset 1, β = 1.43	**Dataset 2, β = 1.49**	**Dataset 3, β = 1.65**
20.0	22	22	22
25.0	16	16	16
30.0	13	12	12
35.0	10	10	9
40.0	9	8	7

It should be noted that neither Table I-IV nor Table I-V are associated with the example and data from the Nose Gear Box described above.

There is evidence to suggest that the CI value to test should be chosen so that it is greater than or equal to the lowest faulted CI value. However, more research would be required before this conclusion can be reached definitively.

a. If CI data follows a Weibull distribution, then this method can be used to reduce the minimum sample size required to demonstrate CI reliability at a desired level of confidence.

b. This method does not specify a minimum number of faulted items; rather, it requires that a minimum number of healthy items are able to exceed the desired CI threshold while still remaining healthy.

c. This method can also be used to compute a minimum required sample size when one or more faults (true yellow or true red) are encountered in the tested items.

d. Finally, it may be possible to develop similar methods for other distributions, such as the lognormal distribution.

I.5 SUMMARY

It is the goal of this appendix to further provoke thoughts and encourage others to perpetuate research in maintenance credits for extending rotorcraft drive/engine systems time on wing with, and without, CBM.

Further research, documentation, and experience need to be accrued to achieve the full benefits of both extending component service time on wing and CBM. The examples within this appendix provide practical implementation methods to meet requirements for maintenance credits intended to modify or replace legacy inspections or TBOs.

When selecting examples, important considerations are:

a. the engineering rigor utilized to establish original maintenance on legacy rotorcraft prior to pursuing maintenance modification/replacement methods;

b. what is at stake when attempting to modify/replace legacy maintenance practices on legacy rotorcraft components.

c. technical variables surrounding a specific form of vibratory CBM monitoring device. Metrics for monitoring should be handled carefully so as to promote the objective for TBO extensions or paths to On Condition. Variables with Vibratory CBM can involve: data ski slopes indicative of bad accelerometers, wiring, or amplifiers; noise which may mask or simulate fault signals; and harmonics which may register a false fault alert.

d. sample sizes necessary to validate a specific vibratory CI/HI for modifying/replacing legacy rotorcraft maintenance. Sample size calculations are based on assumptions that should be tested for validity.

e. CI data following a Weibull distribution may be capable of reducing a minimum sample size required to demonstrate CI reliability at a desired level of confidence.

f. continuous field data assessment is necessary to ensure future faults follow the same distribution established during the initial sample size evaluation.

APPENDIX J

SEEDED FAULT TESTING

J.1 SCOPE

This 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.

J.2 REFERENCES AND 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.

DATA ITEM DESCRIPTIONS	
DI-NDTI-80566A	Test Plan – Data Item Description, 14 November 2006.
DI-NDTI-80809B	Test/Inspection Report, 24 January 1997.

(Copies of these documents are available at https://assist.daps.dla.mil)

VARIOUS REFERENCES	
CBM Test Requirements	US Army Aviation Engineering Directorate (AED) Condition Based Maintenance (CBM) Office, June 2009.
Decker, H.J., and D.G. Lewicki,	"Spiral Bevel Pinion Crack Detection in a Helicopter Gearbox", NASA Glenn Research Center, US Army Research Laboratory, June 2003.
Keller, J.A., and P. Grabill	"Inserted Fault Vibration Monitoring Tests for a CH-47D Aft Swashplate Bearing", US Army RDECOM, June 2005.
Prinzinger, J., and T. Rickmeyer	"Summary of US Army Seeded Fault Tests for Helicopter Bearings," US Army Aviation Engineering Directorate (AED) Propulsion Division, September 2012.

(Copies of these documents are available from sources as noted.)

J.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.

J.4 GENERAL GUIDANCE

Test stand Seeded Fault Testing (SFT) provides a means to acquire the empirical information needed to verify the fault indication(s) in support of on-aircraft CBM validation. SFT can be used to advance the development or refinement of CIs when the specific failure modes are not occurring naturally in the field or in the quantities desired for statistical significance due to legacy maintenance practices. SFT permits measurement and observation of a component in a controlled laboratory environment with a known faulted condition as it degrades towards failure. Further, condition and health indicators can be tested with SFT for their ability to reliably and accurately recognize fault signatures.

SFT can be used for a variety of reasons. One purpose could be to down select among a candidate list of sensors or location of sensors. Another purpose of SFT could be to develop or refine CIs and CI threshold values for achieving an acceptable tradeoff between probability of a false positive (P_{FP}) and false negative (P_{FN}) indications.

Furthermore, SFT can be used to demonstrate 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 and quantify variability of fielded aircraft, CIs should also be tested on multiple aircraft.

An essential diagnostic purpose of laboratory SFT is the ability to accurately correlate an indication level from a CI to a known damage condition. When the specific goal of SFT is to develop or verify a prognostic model, it is necessary to measure the rate of failure progression (i.e. crack growth) and the corresponding rate of change in measured indicators. Note, laboratory testing may confirm some failure modes and fault conditions are not reliably detectable by measured indicators and should not be transitioned to a CBM system. Laboratory testing may also reveal an impending fault may not exhibit any measurable indication prior to complete failure, and, therefore, it also may not be a good CBM candidate.

SFT involves most steps normally associated with aircraft component qualification testing. Figure J-1 and reference J.2(c) outline example SFT and qualification processes used by the US Army.

As shown in the Figure, the process is organized into four general steps.

J.4.1 Step 1: Foundation. Initial test planning begins with determination of goals and objectives for the experiment and should be clearly defined in a Statement of Work (SoW) for the effort. These goals and objectives should be coordinated through the respective aircraft platform's Project Manager's Office and all stakeholders. The primary paths SFT can take are to create or develop CI's, refine or mature the CI's, or to validate CI's for on condition maintenance progression (ref. para. 5.9). During the initial planning step, it is customary that the test articles be procured by the appropriate agency.

a. Failure mode review – The failure mode review should include all available resources of information pertaining to the component's known or anticipated fault mechanisms leading to failure modes. As a minimum, this should include the OEM's initial and updated Failure Mode Effects and Criticality Analysis (FMECA), applicable DA Form 2410 reported failures, and any reported failures from both the Joint Deficiency Reporting System and the U.S. Army Combat Readiness/Safety Center.

b. Seeding the part with a fault – Once the failure mode review has defined the applicable failure modes of interest, representative fault mechanisms should be selected for SFT that will accurately manifest themselves into the failure mode(s) to provide the anticipated result(s) during testing. For example, if the failure mode is bearing thermal runaway (or plastic flow) typical fault mechanisms leading to this failure mode are usually grease degradation/depletion or excessive loading. If the failure mode is Fatigue/Spalling, typical fault mechanisms leading to this failure are corrosion or excessive loading. These fault mechanisms are the subject of focus for the test in question. Depending on the goals and objectives of testing, the test should be provided a specimen that will either operate at an initially measured level of fault damage/degradation to enable fault diagnosis or allow a progression to failure in such a way to permit accurate prognosis of the rate of degradation during the course of the experiment. Introduction of a specimen which is degraded or deformed in a specified or controlled manner will help to ensure the desired fault condition will occur during the test or, if desired, failure will

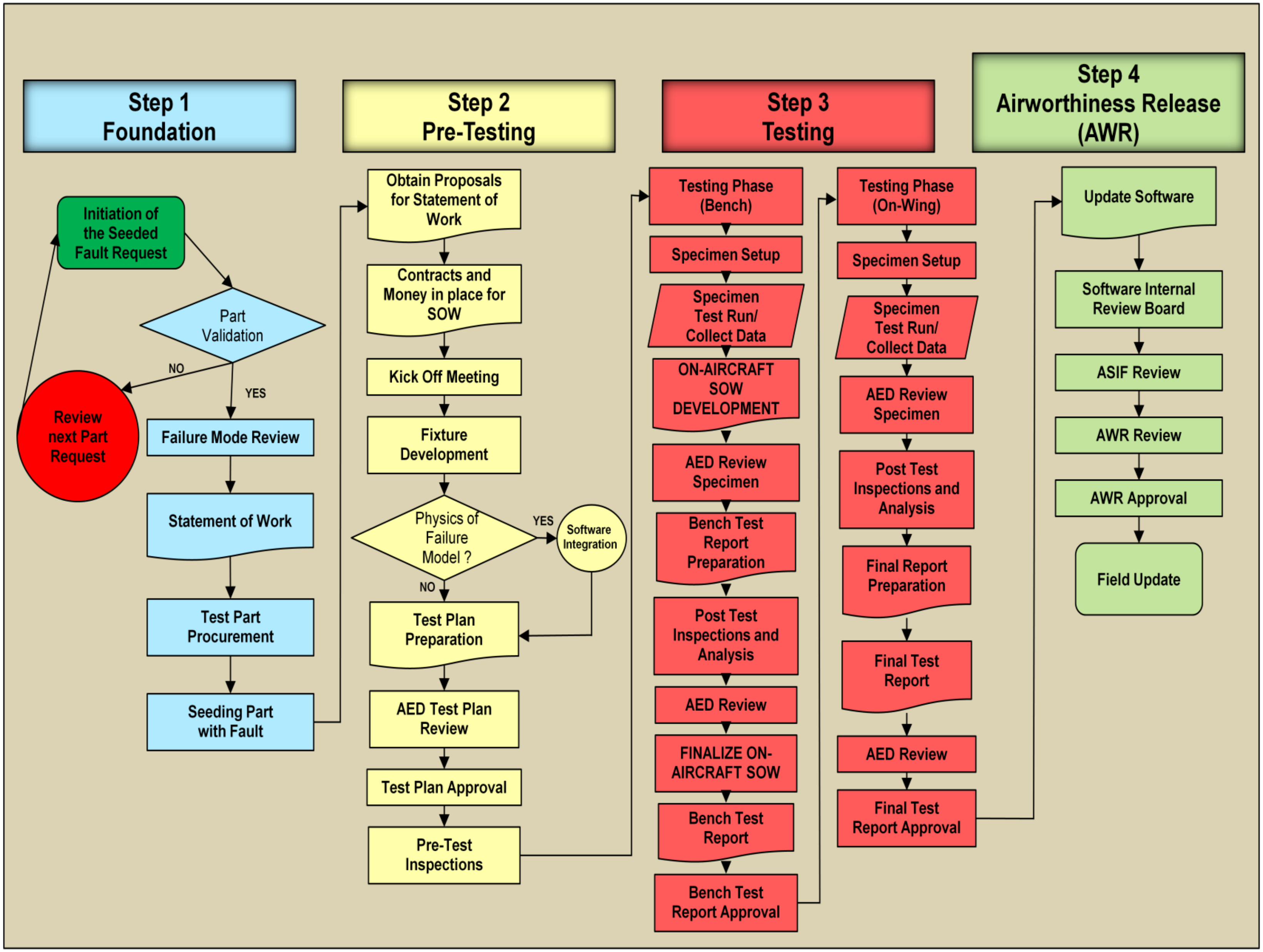


FIGURE J-1. Example seeded fault testing and qualification process

occur within a reasonable test timeframe. The component may be seeded by manually faulting the part in the laboratory, for example, notching the part to initiate a crack at a desired location. Alternatively, an SFT may employ a used, worn, or deformed part from the field which may ultimately result in the anticipated failure when under the induced stress of the laboratory test setup. Additional effort should be considered to ensure initiation of artificially induced faults (i.e. EDM notches, acid etched pitting) that aid in predicting results are representative of expected outcomes. This can be accomplished through pre-testing on material coupons or actual components.

i) Component 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. The point at which it is possible to detect the fault that leads to failure will also determine how much time is remaining before progressing to a complete failure. If faults are not observed early enough prior to failure, the component under test may not be an appropriate CBM candidate.

ii) A second classification of Component 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 or on the platform during a test cycle. Safe operating limits and inspections are imposed on the test to ensure test progression does not cause harm to equipment or personnel.

Note, component failure can involve considerations of the method of aircraft component lifing: safe life or damage tolerance. If a safe life approach was applied to an aircraft component design then a component with a flaw in it under this design approach is considered to be unacceptable for further aircraft usage due to failing inspection for safety. As a result, prior to introducing a faulted, safe-lifed component on an aircraft for limited on-aircraft SFT, the safe-life of the component (time to failure) should be accurately measured in the laboratory. This will provide some assurance the test article will not progress to a state where component failure during on-aircraft SFT.

Also note the statistical significance while analyzing the margin of indication (e.g. delta in CI magnitude differentiating Green to Yellow to Red) of a faulted component for transitioning to on condition monitoring. Analysis may illustrate it may not be beneficial to take the component to failure under the definitions noted in paragraphs 1(i) and (ii), above. For instance, it may prove to be more beneficial to take the component to a known condition that meets the Red criteria (as defined in D.4.2, Appendix D).

J.4.2 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. The laboratory nature of seeded fault testing should allow for the careful isolation of a specific failure condition without interference from other fault conditions. Typically, the test stand should be designed to simulate on-aircraft operating conditions so fault progression and condition indication tracking can proceed as it would in a normal environment. However, at times it may be necessary to exceed normal component operating conditions to achieve a reasonable time limit on the experiment. It is important, though, that test conditions do not specify operation outside of the test fixture safety limits. SFT design should not call for exceeding test stand operating thresholds which 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. During the course of test stand development, a progressive review process should be used to ensure the testing agency will be able to meet the goals and objectives of the test. These progressive reviews the US Army uses are Preliminary Design Review (PDR), Critical Design Review (CDR), and Test Readiness Review (TRR). Through these progressive reviews all stakeholders are involved and provide approval at predetermined developmental criteria.

b. Physics of failure model – A complete SFT analysis would include development of a physics of failure model. This would include the use of the best available modeling and analytical tools to predict fault initiation, fault growth, and component failure under specified test conditions. A rigorous mathematical characterization of the experiment also 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 preparations / review / approval – The SFT Plan should be written using Data Item Description DI-NDTI-80566A as a guide. To synopsize, the SFT Plan should clearly identify the: (1) Test Stand & Component configurations, (2) Calibration requirements (3) Baseline measurements, (4) Fault(s) under test, (5) Condition Indicators being evaluated, (6) Data Format/Requirements, (7) Component Failure Modes, (8) Intended Testing Milestones, (9) Government & Contractor Participation Roles, (10) Test Facilities/Locations, (11) Anticipated Schedule, (12) Safety & Security Guidelines & Responsibilities, (13) Number of Cycles, (14) Success/Failure Criteria and(15) Reporting Requirements/Responsibilities. The test plan should incorporate all stakeholder inputs as to ensure the most comprehensive plan that meet the testing goals and objectives. The overall SFT Plan should be reviewed by associated CBM peers prior to execution. This review should confirm the faulted condition(s) to be induced in the part, the manner in which the fault(s) is generated in the laboratory, and the condition(s) the faulted component will operate under. Therefore, the review should cover both the 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 either produce anticipated results, or fail as expected in the test. Also, a final inspection of the test fixture should be performed to ensure it will operate properly over the entirety of the test and impose the controlled stress needed to induce the expected fault and monitor fault progression/component failure. The US Army uses a review called a Test Stand Verification (TSV). The purpose of the TSV is to document and review the items required to verify the functional capability, data integrity, and safety of the newly constructed test stand/platform, as designed and built by the test facility.

J.4.3 Step 3: Testing. The testing phase proceeds from specimen setup through experiment conduct to test report documentation. Ideally, the seeded fault laboratory experiment should either be positively correlated/scaled to on-aircraft values in the report or 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 the actual operating environment before introducing the application on field 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 within applicable aircraft flight load survey results until the appropriate level of fault degradation and related monitoring 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 – Upon completion of all specimen test runs, a comprehensive test report should be created to document all observed events, findings, and analytic results of the laboratory experiment. The findings should include summary of 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 within a specified timeframe. The specified timeframe should be discussed relative to the anticipated download intervals from the field aircraft. The report should also document the original condition of the test specimen, test fixture, all pre-test analysis, and Tear Down results. The comprehensive test report should be written using Data Item Description DI-NDTI-80809B as a guide.

During the laboratory bench testing phase, considerations should be given to conducting on-aircraft testing of the proposed CBM technique. These considerations can be weighed against the robustness of testing results and the applicability of the results to the aircraft. The purpose of on-aircraft testing is to confirm the implemented technique is sufficiently robust to detect the fault and monitor fault progression in the noise environment of normal aircraft operation. Following a review of bench test results and the decision to validate the CBM hardware on aircraft, an 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 limited repeat of the bench testing with a seeded specimen placed in a test aircraft for evaluation using a combination of legacy maintenance and CBM prior to fleetwide implementation. 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 recognizable fault condition by the HUMS. This will allow for a reasonable amount of normal operation to progress the fault to a detectable level and provide data for again evaluating the condition or health indicator's ability to measure fault progression. Because a faulted, or compromised, component is being introduced into the aircraft, the on-aircraft testing should be conducted with ample consideration given to vehicle and operator safety (e.g. ground run only). In fact, the earlier laboratory testing should provide as accurate an estimate as possible to the remaining safe-life using legacy maintenance methods for the specimen prior to installation on the aircraft. This will provide some continued airworthiness assurance 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 when possible. Therefore, the aircraft testing should normally be performed as part of an experimental flight with a trained test pilot unless circumstances justify otherwise.

J.4.4 Step 4: Follow-on efforts. Pending the conclusions and results found in the bench and on-aircraft test reports, an AWR or Safety Message may be generated to alert the fleet as to any changes required in a fielded CBM system, and to include any CBM maintenance manual changes. Depending on the purpose and intended goals of the SFT, 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.

J.5 DETAIL GUIDANCE

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

In the NASA / US Army Research Laboratory study on crack detection reference sited in Section J.2, 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 main 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 field solution.

The US Army RDECOM report "Inserted Fault Vibration Monitoring Tests for a CH-47D Aft Swashplate Bearing," cited in Section J.2, involves the detection of a swashplate bearing failure in a CH-47D Chinook. This test method offers an acceptable alternative to Reference 21 for obtaining a test specimen for initial SFT. In Reference 22, heavily worn, used components returning from field operation 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 referenced 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 in these tests. It would be expected, however, following the example process guidance in Figure H-1, that, in situ, on-aircraft seeded testing be used to validate any laboratory findings before issuing a flight/fielding AWR for a CBM system on US Army aircraft.

The US Army RDECOM report "Summary of US Army Seeded Fault Tests for Helicopter Bearings," cited in Section J.2, addresses AH-64 SFT for both a main rotor swashplate (MRSP) and a tail rotor hanger bearing. The SFT report communicates the methodologies; emphasizes the necessity of seeded fault testing the US Army employs in pursuit of CBM; outlines SFT applications; and documents how the SFT applications facilitate HUMS validation on actual components.

While the MRSP testing is still ongoing, the TRDS results and analyses provide substantiation for an actual US Army approved maintenance credit when using a HUMS device. This document substantiates a maintenance credit from a 2500 flight hour TBO to 6487 hours for a hanger bearing using a probabilistic assessment. In contrast to the previous two examples cited for the OH-58 and CH-47 aircraft, above, this report provides actual monitored aircraft data along with the SFT bench data. Note also, confidence and reliability play a key role in analyzing the aircraft/bench data, developing statistically significant sample sizes, and establishing a basis for changing from legacy aircraft maintenance practice to CBM for continued airworthiness.

APPENDIX K

VERIFICATION AND VALIDATION OF CBM PROCESSES

K.1 SCOPE

This appendix provides the general guidance for the Verification and Validation of diagnostics-based Maintenance Credits and Prognostics Processes. . These approaches are not the only methods. This guidance material does not address Validation & Verification (V&V) best practices that would normally be utilized throughout product and system development cycles. To be specific about our objective and these terms, Verification provides testing or other evidence that the application meets its specifications. Validation provides testing or other evidence that the application performs as intended in the operational and maintenance environment of the aircraft.

K.2 REFERENCES AND APPLICABLE DOCUMENTS

MILITARY STANDARDS	
MIL-HDBK-1823	Non-Destructive Evaluation System Reliability Assessment. Department of Defense, 2009

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

FEDERAL AVIATION ADMINISTRATION	
FAA AC 29-2C MG15-	Airworthiness Approval of Rotorcraft Health Usage Monitoring Systems (HUMS)

(Copies of these documents are available online at http://www.faa.gov/regulations_policies/)

FEDERAL AVIATION REGULATION	
FAR/JAR Parts 27, 29 -	Airworthiness Standards: Normal Category Rotorcraft, Transport category rotorcraft.

(Copies of these documents are available online at or https://dap.dau.mil/policy/Pages/overview.aspx)

VERIFICATION AND VALIDATION (V&V)	
SAND 2003 – 3769	Verification, Validation, and Predictive Capability in Computational Engineering and Physics, Sandia National Laboratories, 2003.

(Copies of this document are available from http://prod.sandia.gov)

K.3 DEFINITIONS

Accuracy: A measure of how close predicted failure times are to the actual (observed) failure times.

Age: A measure of exposure to stress computed from the moment an item or component enters service when new or re-enters service after a task designed to restore its initial capability, and can be measured in terms of calendar time, running time, distance traveled, duty cycles or units of output or throughput. (SAE JA1012)

Applicable Task: A task that is capable of preventing or mitigating the consequences of failure based on the technical characteristics of that failure. (SAE JA1011)

Appropriate Task: A task that is both technically feasible and worth doing (applicable and effective). (SAE JA1012)

Commercial Off The Shelf (COTS): This term defines equipment hardware and software that is not qualified to aircraft standards. An example of COTS equipment hardware and software is a personal computer (PC) and its operational software. (AC 29-2C)

Complexity: A measure of effectiveness defined as the ratio between computational time and expected time to failure.

Conditional Probability of Failure: The probability that a failure will occur in a specific period provided that the item concerned has survived to the beginning of that period. (SAE JA1012)

Confidence: The probability that the true reliability is at least as high as what is stated, equal to one minus the probability of a false negative. The target confidence is 90%. (ADS-79C)

Criticality: This term describes the severity of the end result of an item or process failure/malfunction. Criticality is determined by an assessment that considers the safety effect that an item or process can have on the aircraft. (ADS79D Appendix L Section 5.1)

Damage Fraction: An estimate of fatigue damage due to identified usage and/or fatigue loading where a damage fraction of 1.0 is typically associated with a probability of fatigue crack initiation.

Desired Performance: The level of performance desired by the owner or user of a physical asset or system. (SAE JA1012)

Economic Consequences: A classification assigned to failure modes, or multiple failures in the case of hidden failure modes, that do not adversely affect safety, the environment, or operations, but increase cost either from repair or from lost or degraded operations. (SAE JA1011)

Effective Task: A task that reduces the probability or consequences of failure to an acceptable level and is feasible to perform. (SAE JA1011)

Environmental Consequences: A classification assigned to failure modes, or multiple failures in the case of hidden failure modes, that could result in a breach of any industry or government environmental standard or regulation. (SAE JA1011)

Evident Failure: A failure mode whose effects become apparent to the operator(s) under normal circumstances if the failure mode occurs on its own. (SAE JA1011)

Evident Function: A function whose failure on its own becomes apparent to the operator(s) under normal circumstances. (SAE JA1011)

Failure Consequences: A classification of the failure effects of failure modes into categories based on evidence of failure, impact on safety, the environment, operational capability, and cost. (SAE JA1011)

Failure Effect: What happens when a failure mode occurs. (SAE JA1011)

Failure-Finding Task: A scheduled task used to determine whether a specific hidden failure has occurred. (SAE JA1011)

Failure Management Policy: A generic term that encompasses on-condition tasks, scheduled restoration, scheduled discard, failure-finding, run-to-failure, and one-time changes. (SAE JA1011)

Failure Mode: A single event, which causes a functional failure. (SAE JA1011)

Function: What the owner or user of a physical asset or system wants it to do. (SAE JA1011)

Functional Failure: A state in which a physical asset or system is unable to perform a specific function to a desired level of performance. (SAE JA1011)

Hidden Failure: A failure mode whose effects do not become evident to the operator(s) under normal circumstances if the failure mode occurs on its own. (SAE JA1011)

Hidden Function: A function whose failure on its own does not become evident to the operator(s) under normal circumstances. (SAE JA1011)

Independent Verification Means: An independent process to verify the correct functionality of a HUMS application on a ground station that utilizes COTS. The intent of independent verification is to gain some degree of confidence in the COTS operational reliability. **Note:** This process may be discontinued when sufficient confidence in the application has been achieved. (AC 29-2C)

Initial Capability: The level of performance that a physical asset or system is capable of achieving at the moment it enters service. (SAE JA1011)

Integrity: Attribute of a system or a component that can be relied upon to function as required by the criticality determined by the Functional Hazard Assessment (FHA). (AC-29-2C)

Maintainer: A person or organization that may either suffer or be held accountable for the consequences of a functional failure or multiple failures by virtue of performing maintenance functions on behalf of the User and/or owner of the asset or system. (SAE JA1011)

Mitigating Action: An autonomous and continuing compensating factor which may modify the level of qualification associated with certification of a HUMS application. This action becomes a part of the certification requirements and, as such, is required to be performed as long as that certification requirement is not changed by a subsequent re-certification. An example of a mitigating action is a pilot's comparison of airborne HUMS data with aircraft instrument data. (AC 29-2C)

Multiple Failure: An event that occurs if a protected function fails while its protective device or protective system is in a failed state. (SAE JA1011)

Net P-F Interval: The minimum interval likely to elapse between the discovery of a potential failure and the occurrence of the functional failure. (SAE JA1012)

Non-Operational Consequences: A classification assigned to failure modes that do not adversely affect safety, the environment, or operations, but only require repair or replacement of any item(s) that may be affected by the failure. (SAE JA1011)

On-Condition Task: A periodic or continuous task used to detect a potential failure. (SAE JA1011)

One-Time Change: Any action taken to change the physical configuration of an asset or system (redesign or modification), to change the method used by an operator or maintainer to perform a specific task, to change the operational context of the system, or to change the capability of an operator or maintainer (training). (SAE JA1011)

Operational Context: The circumstances in which a physical asset or system is expected to operate. (SAE JA1011)

Operational Consequences: A classification assigned to failure modes that adversely affect the operational capability of a physical asset or system (output, product quality, customer service, military capability, or operating costs in addition to the cost of repair). SAE JA1011)

Owner: A person or organization that may either suffer or be held accountable for the consequences of a functional failure or multiple failure of an asset or system by virtue of ownership of that asset or system. (SAE JA1011)

P-F Interval: The period between the point at which a potential failure becomes detectable and the point at which it degrades into a functional failure. (SAE JA1011)

Prevalence: Prevalence is defined as the fraction of defectives in a given population at a specific time.

Protective Device or Protective System: A device or system which is intended to avoid, eliminate, or minimize the consequences of failure of some other system. (SAE JA1011)

Potential Failure: An identifiable condition that indicates that a functional failure is either about to occur or is in the process of occurring. (SAE JA1012, SAE JA1011)

Precision: A measure of narrowness of dispersion associated with the predicted results for multiple experiments.

Primary Function(s): The function(s) which constitute the main reason(s) why a physical asset or system is acquired by its owner or user. (SAE JA1011, SAE JA1012)

Proactive Maintenance**:** Maintenance undertaken before a failure occurs, in order to prevent the item from getting into a failed state (scheduled restoration, scheduled discard, and on-condition maintenance). (SAE JA1012)

Positive Predictive Value (PPV): The probability that a part actually has defect if sensor indicates a positive result. (MIL-HDBK 1823)

Negative Predictive Value (NPV): The probability that a part is defect-free if sensor indicates a negative result. (MIL-HDBK 1823)

Prognostics: Identifies predictable periods of operation free of functional failures with an estimate of time to failure, maintenance action, or remaining useful life.

Reliability Centered Maintenance (RCM): A process to ensure that assets continue to do what their users require in their present operating context. (ADS-79D) RCM is a specific process used to identify the policies which must be implemented to manage the failure modes which could cause the functional failure of any physical asset in a given operational context. (SAE JA1011)

Robustness: A measure of capability to perform intended functions without failure under a wide range of conditions specified.

Run-to-Failure: A failure management policy that permits a specific failure mode to occur without any attempt to anticipate or prevent it. (SAE JA1011)

Safety Consequences: A classification of failure modes that could injure or kill a human being. (SAE JA1011)

Scheduled: Performed at fixed, predetermined intervals, including "continuous monitoring" (where the interval is effectively zero). (SAE JA1012)

Scheduled Task: Maintenance tasks performed at fixed, predetermined intervals, or through the use of "continuous monitoring" (where the interval is effectively zero) to prevent or mitigate the consequences of failure or multiple failure. (SAE JA1011)

Scheduled Discard: A scheduled task that entails replacing an item at or before a specified age limit regardless of its condition at the time. (SAE JA1011)

Scheduled Restoration: A scheduled task that restores the capability of an item at or before a specified interval (age limit), regardless of its condition at the time, to a level that provides an acceptable probability of survival to the end of another specified interval. (SAE JA1011)

Secondary Functions: Functions which a physical asset or system has to fulfill in addition to its primary function(s), such as those needed to fulfill regulatory requirements and those which concern issues such as protection, control, containment, comfort, appearance, energy efficiency, and structural integrity. (SAE JA1011)

Sensitivity: Sensitivity is defined as the probability of a true positive, P(detection | target present).

Similarity: Comparison of predicted time series to real (ground truth) time series.

Specificity: Specificity is defined as the probability of a true negative, P(no detection | no target present).

Structural Health: The "state" of the constituent materials, of the different parts, and of the full assembly of these parts constituting the structure as a whole.

Synthesis: The process of evaluating service history and any other relevant data with the objective of validating and, if necessary, refining the performance of an approved credit. (AC 29-2C)

Unscheduled Maintenance: Those unpredictable maintenance requirements that had not been previously planned or programmed but require prompt attention and must be added to, integrated with, or substituted for previously scheduled workloads.

Uncertainty: Measure of variability associated with the predictive outcomes and experiments.

User: A person or organization that operates and/or maintains an asset or system or may either suffer from or be held accountable for the consequences of a failure of that asset or system. (SAE JA1011)

K.4 GENERAL GUIDANCE

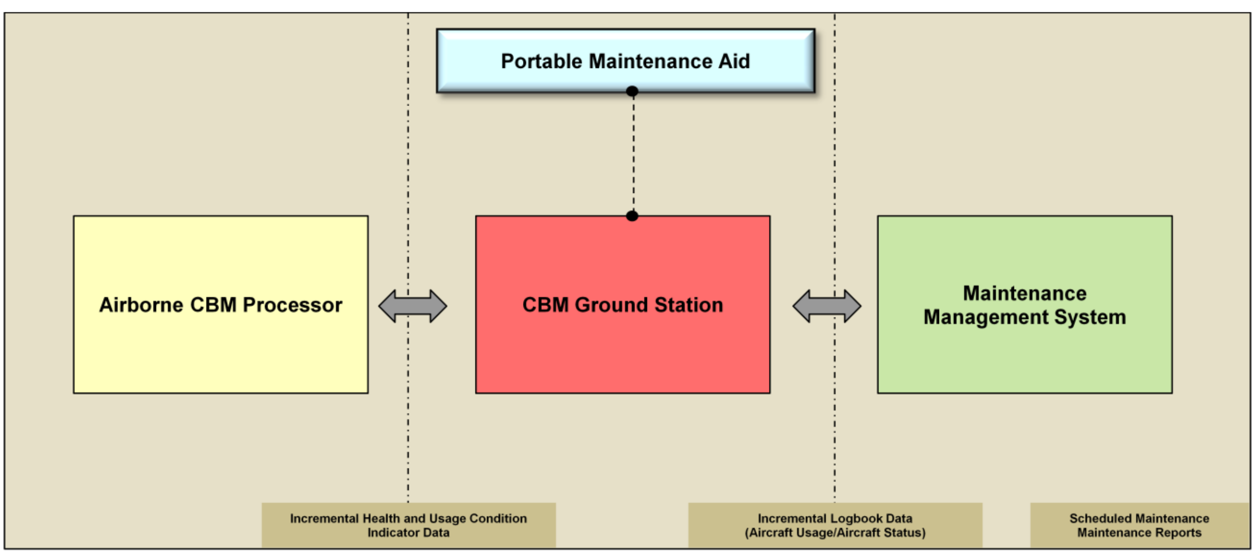
K.4.1 Purpose

This section of the V&V document was prepared to outline the hardware and software processing elements in the Continuing Airworthiness Plan, as well as user documentation that must be identified and controlled as part of the V&V process.

K.4.2 Discussion There are many factors that must be controlled to ensure that a baseline credit configuration has been defined and that proper documentation exists prior to formal V&V Credit testing.

CBM systems are unique in that both airborne and ground elements very often make up the end-to-end Credit processing application. In general a credit application will need to show that all of the elements *contributing* to the final credit result are defined and have been controlled. We use the word "contributing" to note that it may not be possible or practical to have collected all of the data with systems of the *exact configuration*. For example, the software used during V&V might not be configured to measure a Conditions Indicator for an unrelated measurement, or thresholds for unrelated CI's might have changed over time. It is likely that some analysis will be required to show that the configuration baseline is controlled and all testing is relevant to the credit being pursued.

Figure K-1 shows the top level elements of a generic CBM system architecture. The figure shows that a typical CBM system is made up of 3 or 4 different major processing elements. Incremental CBM data is collected in the airborne system and further processed in the ground station or portable maintenance aid. In some cases the end-to-end credit processing is not completed until the data is passed to the operator's Maintenance Management System.



Elements of a CBM System Architecture

FIGURE K-1. Top level system elements

It is in the Maintenance Management System where the individual aircraft parts configuration would traditionally be kept and maintenance schedules would be generated. Unscheduled maintenance activity and trouble-shooting support is likely to be generated by the CBM Ground Station.

Figure K-2 breaks these major elements down into other definable and configurable system elements including dedicated CBM Sensors or remote “smart” boxes that may be part of the end-to-end credit path. Mixed in with the application software are other software components such as operating system software and databases. In some cases these are COTS components which must be controlled with respect to their influence on the credit being sought. After all of the major software modules are defined and controlled, the system will still not perform the intended credit processing unless the CBM configuration files defining the aircraft configuration, CBM data acquisition requirements, acquisition schedules, and finally specific CI thresholds are controlled.

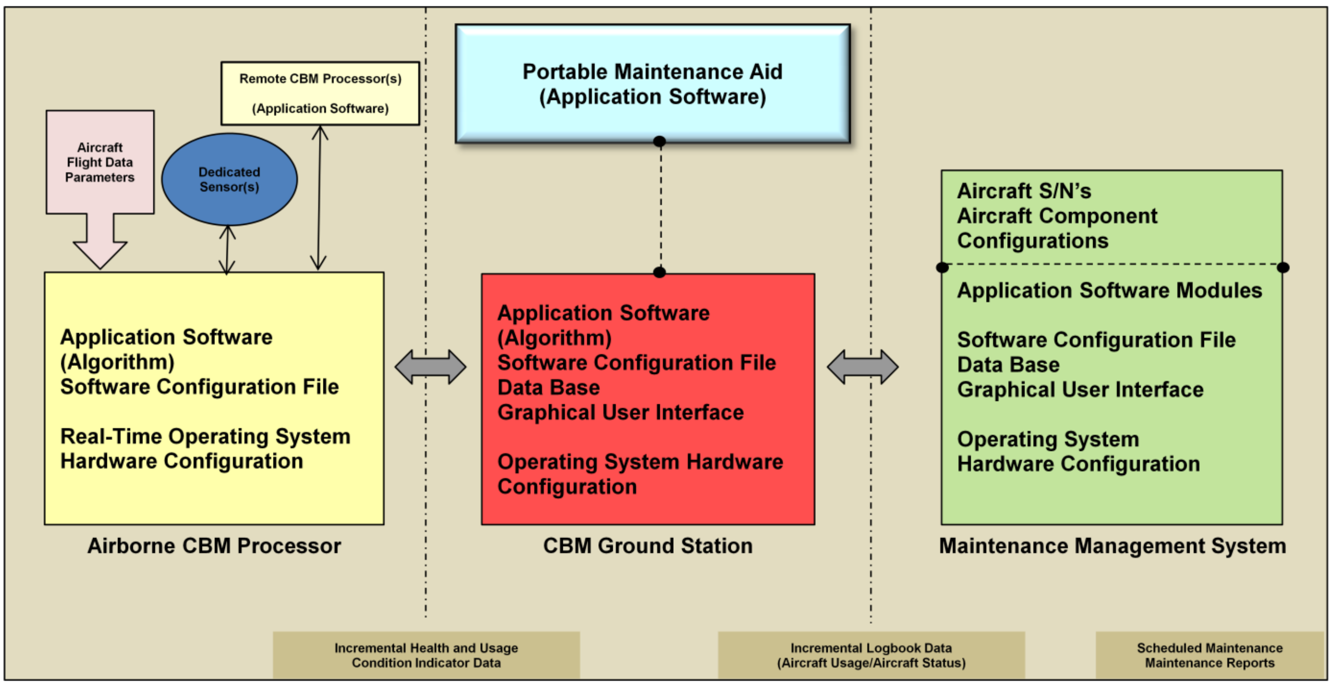


FIGURE K-2. Elements of CBM system architecture

These elements must be carefully tracked and controlled during the design process and documented before formal V&V testing. The final V&V of a Credit must be verified and validated on a production or production representative test configuration.

Table K-I provides a listing that can be used as a checklist to ensure that all elements in the end-to-end credit process have been accounted for.

TABLE K-I. End-to-End Credit Process Checklist
System Level Baseline for Maintenance Credits
-Specific aircraft Model with a defined certification basis
-Maintenance Manual
-CBM Airborne Data Acquisition and Monitoring System
-CBM Ground Station
-Operator's Maintenance Management System
CBM Sensor and Flight Data Elements
-Integrity of the source data (System of origination, qualification level, accuracy, resolution, validity checks)
-Integrity of the sensor in aerospace environment (Performance Specification and Qualification as well as Calibration)
-Verification and Validation of these elements are the same as any other hardware that is added to the aircraft and require no unique V&V process and should follow standard V&V processes that similar elements follow.
CBM Documentation Elements
-Maintenance Credit Hazard Analysis
-CBM user manual
-CBM flight manual supplement
-CBM Maintenance Manual Supplement
-CBM Continuing Airworthiness Plan
-Verification and Validation require no unique processes for this element
CBM Airborne Monitoring System Elements
-CI algorithms (Reference App D for Guidance)
○ Detection SNR/Resolution/Accuracy
○ Maintenance Limit (yellow)
○ Functional or Airworthiness Limit (red)
-Application Software
-Operating System Software
-Data Acquisition Configuration File for aircraft physical configuration, sensor configuration, engineering units, data acquisition schedule, data acquisition conditions, and analyses applied
-Airborne Processing Hardware (hardware qualified for environment)
-HUMS systems DSC

TABLE K-I. End-to-End Credit Process Checklist (Cont'd)
-For detail on V&V processes reference section K.5
CBM Ground Station Elements
-PC hardware configuration
-Operating System software
-Database software
-CBM Application software
-Graphic User Interfaces
-Data review and collection procedures
-Backup and Archival procedures
-Interface to Maintenance Management System
-Software Description Document
-Software Requirements Specification
-Software Test Plan
-Acceptance Test procedure (ATP)
-For Ground Station V&V details reference section K.6
Operations Maintenance Manual
-For further detail refer to ULLSA-E documentation as it is a separate program of record

K.4.3 General Considerations for Verification and Validation. The terms "verification" and "validation" have been used interchangeably in casual conversation as synonyms for the collection of corroborating evidence. In the context of modeling and simulation, per ASME V&V 10, "verification" is the process of gathering evidence to establish that the computational implementation of the mathematical model and its associated solution are correct. "Validation," on the other hand, is the process of compiling evidence to establish that the appropriate mathematical models were chosen to answer the questions of interest by comparing simulation results with experimental data. Therefore, a complete and comprehensive V&V process should address both verification and validation of the model.

A notional hierarchy of the V&V process for modeling and simulation, as recommended by ASME V&V 10, is shown in Figure K-3. The process consists of mathematical modeling and physical modeling branches. Design engineers follow the left branch to develop, exercise, and evaluate the model. Testing engineers follow the right branch to obtain the relevant experimental data via physical testing. Modelers and experimenters collaborate in developing the conceptual model, conducting preliminary calculations for the design of experiments, and specifying initial and boundary conditions for calculations for validation.

In the modeling perspective (shown as the left branch), a mathematical model representing the physics nature of the target problem is developed, and detailed numerical algorithms are implemented into the computational model. Verification of the computational model is performed to check functionality and reliability of the code. Once the fidelity of the computational model is satisfied, numerical simulations are performed to generate numerical outcomes. Further verification of the computational model is followed to ensure desirable robustness and accuracy of the code. Often, actual data and/or case studies with known solutions are used at this stage. Due to the inherent scatter associated with input parameters and model assumptions and simplification of the mathematical models resulting from incomplete knowledge of the physical nature of the problem, variability in the response of interest is often expected and rigorous uncertainty quantification needs to be carried out. The simulation outcomes obtained from verified and calibrated computational frameworks form a database for further comparison against the outcomes of physical modeling.

The right branch in Figure K-3 highlights the process of physical testing. The essential element in the process is to validate mathematical models through physical experiments. The purpose of validation experiments is to generate baseline "ground truth" data needed to assess the accuracy of the mathematical model; therefore, all assumptions behind the mathematical model should be understood, well defined and controlled. The process starts with a basic physics-based model describing the underlying problem. Often, preliminary assessment using the computational model would be performed to identify the critical factors for further experimental design. During this stage, preliminary sensitivity analysis and associated uncertainty quantification are performed to help determine the location and type of measurements needed for validation. Once a validation testing plan has been established, the experiments will be performed and measurement data from various instruments will be collected and further processed. The extracted features will form the baseline for direct comparison with the simulation outcomes. Repeat experiments are generally required to quantify uncertainty due to lack of repeatability and inherent variability. The experimenter then performs uncertainty quantification to quantify the effects of various sources of uncertainty on the experimental data.

Once experimental outcomes and simulation outcomes for the actual test conditions have been generated, the validation assessment will be conducted by comparing these two sets of outcomes. The metrics for comparing experimental outcomes and simulation outcomes, as well as the criteria for acceptable agreement, will have been specified in V&V plan stage. The degree to which the model accurately predicts the data from validation experiments is the essential component of the overall assessment of the model's predictive capability. This leads to further accredit the benefit to be obtained from a CBM using the predictive model. Note, however, the "acceptable agreement" provides a subjective decision point for initiating improvements in the conceptual, mathematical, and computational models and in the experimental designs.

Sometimes, the outcomes of simulation and physical observation may not result in agreement. If the discrepancy between the mathematical model and physical reality exceeds a pre-defined acceptance limit, the outcomes from the mathematical model and the physical model need to be carefully examined. The key assumptions for the conceptual model, its mathematical representation, and the computational framework must be reviewed. The scope and setup of physical experiments for the purpose of validation are also subjected to further scrutiny. If an excessive discrepancy is found and attributed to aforementioned aspects, appropriate adjustments or modifications in the mathematical and /or physical model are necessary. In addition, the computational framework may need further calibration to improve agreement with respect to a chosen set of benchmarks through the adjustment of parameters implemented in the solution.

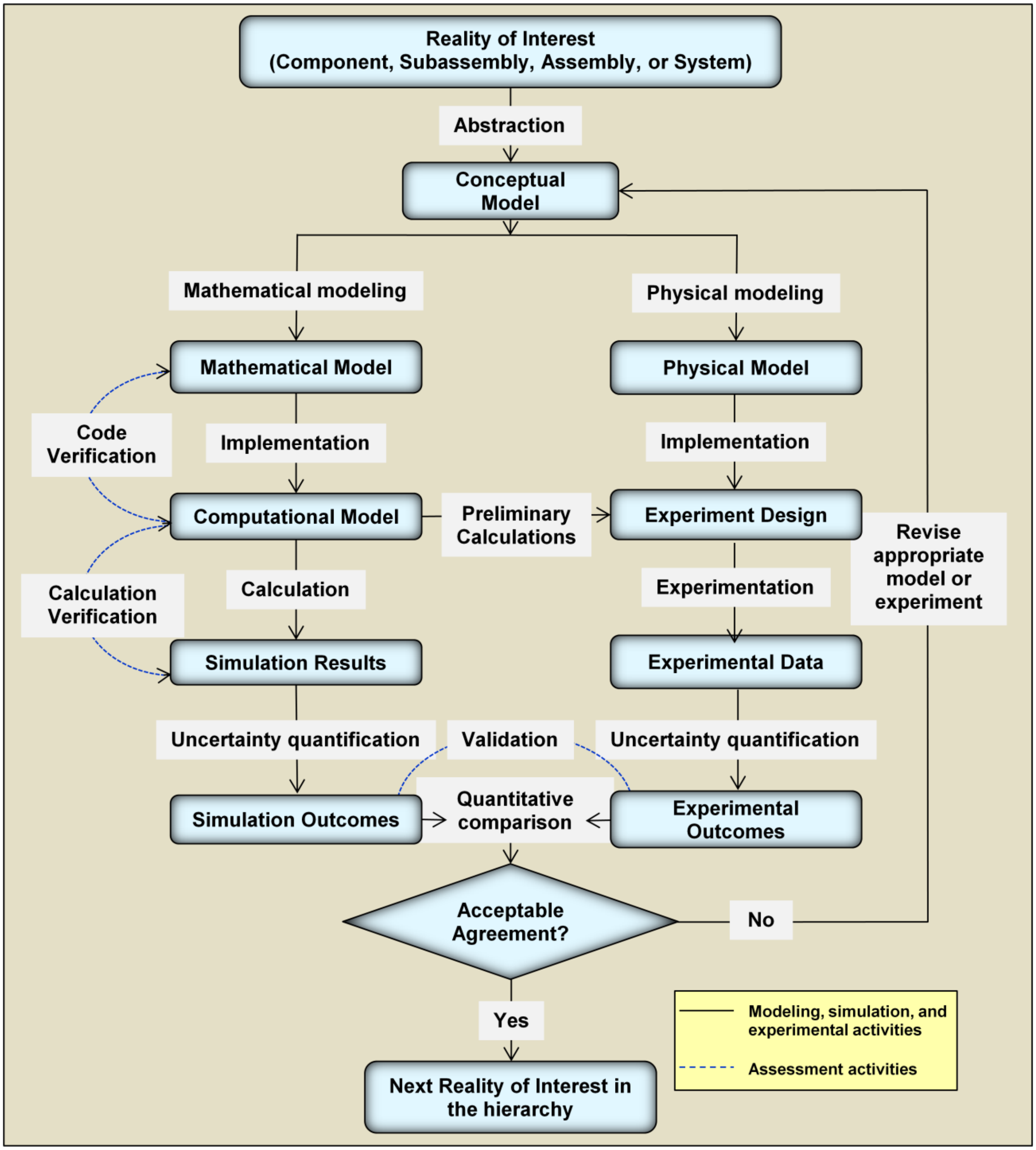


FIGURE K-3. V&V activities and flow for modeling and simulation (ASME V&V 10)

The V&V activities involve an iterative process. To effectively validate a model of interest, verification needs to be performed first. The proof-of-concept and verified computational model is employed to conduct preliminary analysis to facilitate proper setup for validation experiments. The experimental outcomes are used to benchmark simulation outcomes obtained from the computational model. Based on the outcomes of comparisons between simulation and experiments, various model calibrations or enhancements will be made to improve the model's predictive capability, if necessary.

The primary objective of validation is to establish a desirable level of confidence that a model or algorithm under consideration reasonably represents intended physical reality. Therefore, validation needs to be carefully planned and performed to cover the region of primary interest. To establish a database for further validation, the scope of validation, in terms of validation domain, needs to be determined first. Ideally, the validation domain should cover entirely the domain of intended application. Practically, however, there is often a budgetary or schedule limit to prevent excessive scope of validation. This is particularly important for high fidelity physics based computational models for a complex system. Therefore, understanding the nature of the physics of the underlying computational model plays an essential role in establishing a feasible and tangible validation plan to ensure the accuracy and applicability of the model.

Validation strategies can be further categorized based on amount of overlap between the validation domain and application domain. As shown in Figure K-4, the "Application Domain" designates the region where predictive capability from a computational model is needed for potential applications, while the "Validation Domain" represents a region where the proper understanding of relevant physics is well established and confidence of the associated computational model has been quantitatively demonstrated by satisfactory agreement between predictive model and experiments.

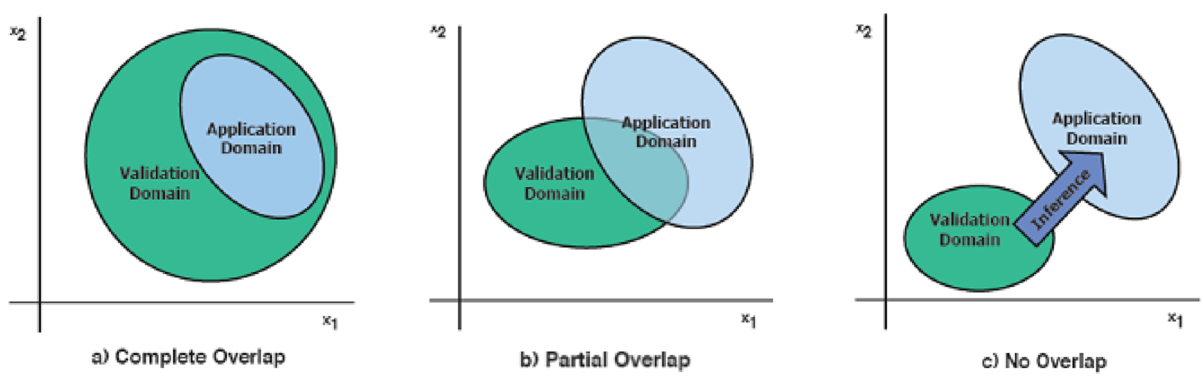


FIGURE K-4. Illustration of Various Validation Sceneries[64]

[64] SAND 2003 – 3769, Verification, Validation, and Predictive Capability in Computational Engineering and Physics, Sandia National Laboratories, 2003.

In general, the validation schemes can be classed into three categories: 1) Complete overlap, 2) Partial overlap, and 3) No overlap. The first category represents a vast majority of validation cases for most engineering issues. This validation scheme engages a broad validation domain which covers the entire application domain. It is the prevalent situation in validation since the entire region of application has been validated and additional validation data beyond the application domain provides the means for future extension of applicability of the computational model. Sometimes, extensive validations covering the entire application domain may not always be feasible. As depicted in Figure K-4 (b), the domain of validation addresses only a portion of the application domain. This validation scheme represents an engineering situation where a comprehensive validation is infeasible or prohibitive and therefore the focus of validation is on the application region of primary interest. In the third validation scheme, shown in Figure K-4 (c), there is no overlap between validation domain and application domain. This scenario highlights a rare engineering situation in which direct validation of the applicability of a model is not possible. As the only alternate, indirect validations are conducted to confirm the basic assumptions of the model and inference from the validation domain is made using both physics based models and statistical methods. Although the aforementioned validation scheme may seem to lack rigor, it may be the only option for validation of some complicated engineering issues. A typical example of using the inference-based approach to validate a predictive model is to validate an estimate of probability of failure or quantitative risk assessment. In the aircraft industry, highly reliable components are expected. Therefore, direct evidence for extremely small probability failures are very difficult to obtain. To validate probabilistic prediction for a specific component, indirect evidence such as limited fielded data for an entire fleet as well as data for similar components are collected and pooled to form the basis for further statistical inference. Once the statistical confidence of the pooled database is shown to demonstrate similar behavior, validation of the probabilistic prediction can be made accordingly. As discussed by Oberkampf, et.al the need to perform extrapolation beyond a model's validation domain reinforces the need for models to be judged on the basis of achieving the right answers for the right reasons in the intended validation regime.[65]

Validation, as one of the vital steps to qualify Maintenance Credits, is generally carried out through physical experiments. A variety of validation methodologies exist. Validation testing can be performed by either physical or numerical experiments. Physical experiments are an ideal way to conduct validation testing. Various forms of physical experiments have been used for validation, including sub-component/component test, sub-system test, seeded fault test, and fielded system level test. Each of these forms has its own advantages and disadvantages. Selection of a specific form of physical experiment depends on the intended scope of study, requirements of model accuracy, criticality of potential failure mode under the investigation, budget and schedule of the validation program. Clearly, performing fielded system level test increases the complexity and model fidelity. But this also decreases feasibility of validation and causes potential delay of implementation. If feasible and affordable, physical experiments offer great opportunities to generate direct evidence to compare simulation prediction against physical

[65] SAND 2003 – 3769, Verification, Validation, and Predictive Capability in Computational Engineering and Physics, Sandia National Laboratories, 2003.

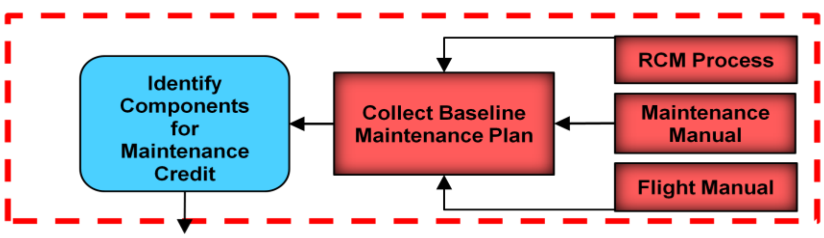
reality. They provide realistic data with usually well controlled and documented ground truth information.

Sometimes, physical experiment may not be a viable option for validation due to complexity of the testing, excessive cost of physical experiment, or even infeasibility to perform such a test. As an alternative, numerical experiments may be used to generate simulated features or signatures for further comparison with the computational model. This approach has been used in the nuclear industry and gains more popularity in validating diagnostic and/or prognostic algorithms. Often, its benefit in cost effectiveness may be offset by the lack of representative realistic data. If numerical experiment is used in validation, efforts needs to be made to ensure the relevance and quality of the simulation model utilized.

K.5 MAINTENANCE CREDIT V&V PROCESSES AND PROGNOSTICS

Maintenance Credits are acquired when a HUMS system DSC can modify or replace the existing industry standard maintenance interval for a given component. Thus, the HUMS system DSC must thoroughly prove that its estimation of the component's condition, that is to modify or replace the current standard, is accurate. A verification and validation process is provided in this section that is to be used to achieve Maintenance Credit at a component level.

For the sake of simplicity and flow of explanation, the process below has been partitioned into subsections. The subsections are ordered in congruence with Figure K-5.



Sub-Section 1: The overall Maintenance Credit V&V process begins with the identification of the component for Maintenance Credit. The identification step should account for the component's current maintenance practice as outlined in its RCM process, the helicopter platform's maintenance manual and its flight manual. The analysis of these three components will help collect the baseline maintenance plan.

Sub-Section 2: Once the component has been identified, the next step is to determine the severity of the component. The severity classification will later be used in determining the probability of false positive (PFA) of the credit methodology during development. For more information on the severity categories, refer to Appendix D of ADS-79D, "Minimum Guidance for Determining CIs/HIs for Propulsion Systems."

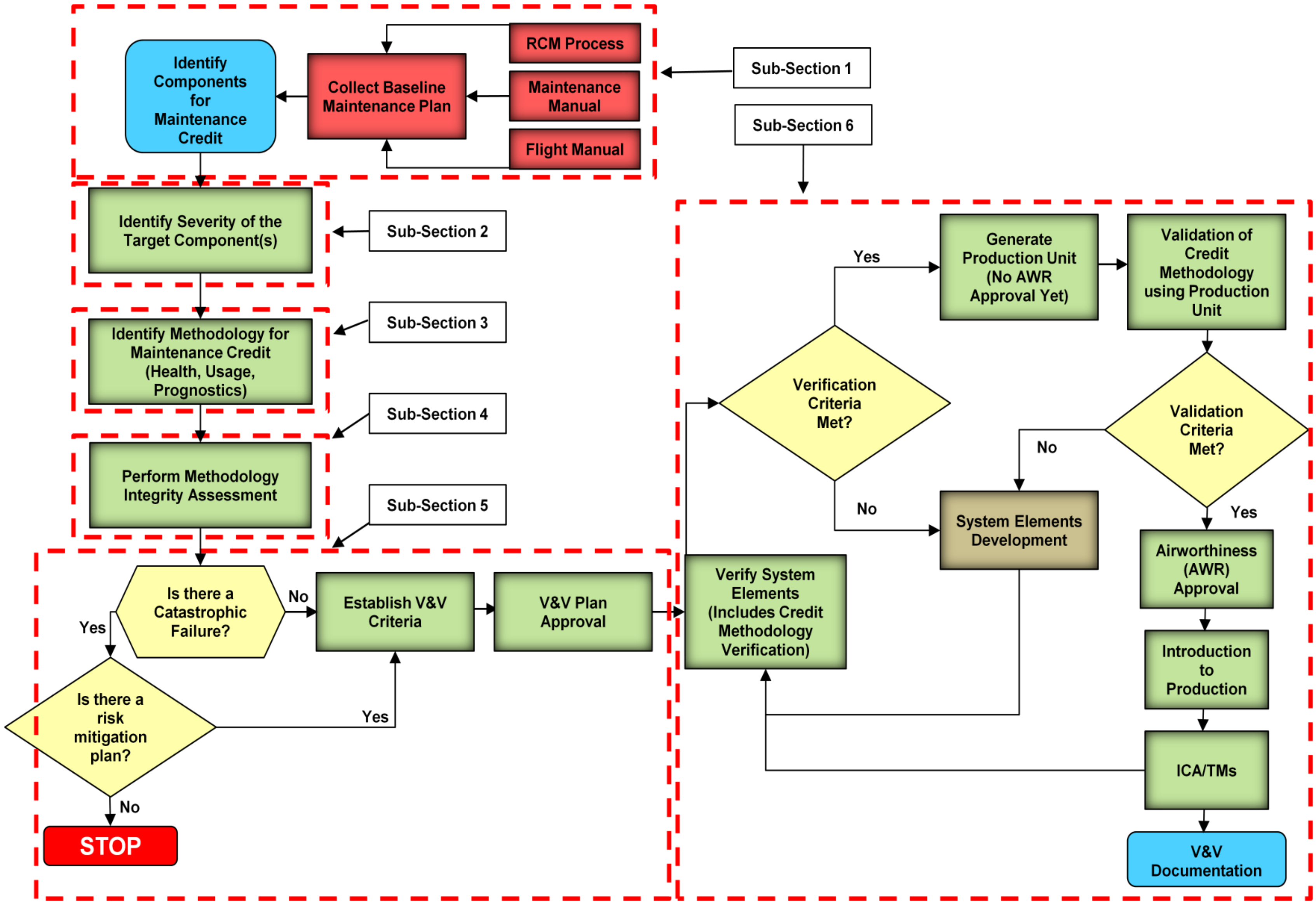
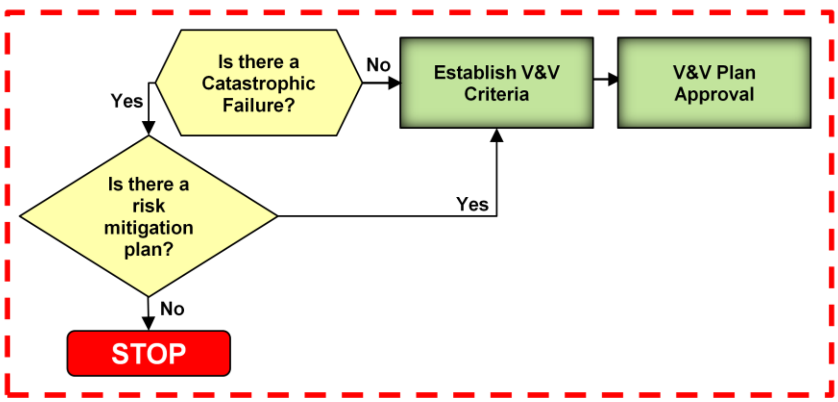


FIGURE K-5. Maintenance credit verification & validation process

Sub-Section 3: The next step is to determine the methodology for credit. Here, one can choose a Mechanical Health approach, a Mechanical Health-based Prognostics approach, a Usage/Structural Health Monitoring (SHM) approach or a Usage/SHM-based Prognostics approach. In this step, the algorithm is proposed and all necessary development activities can be conducted.

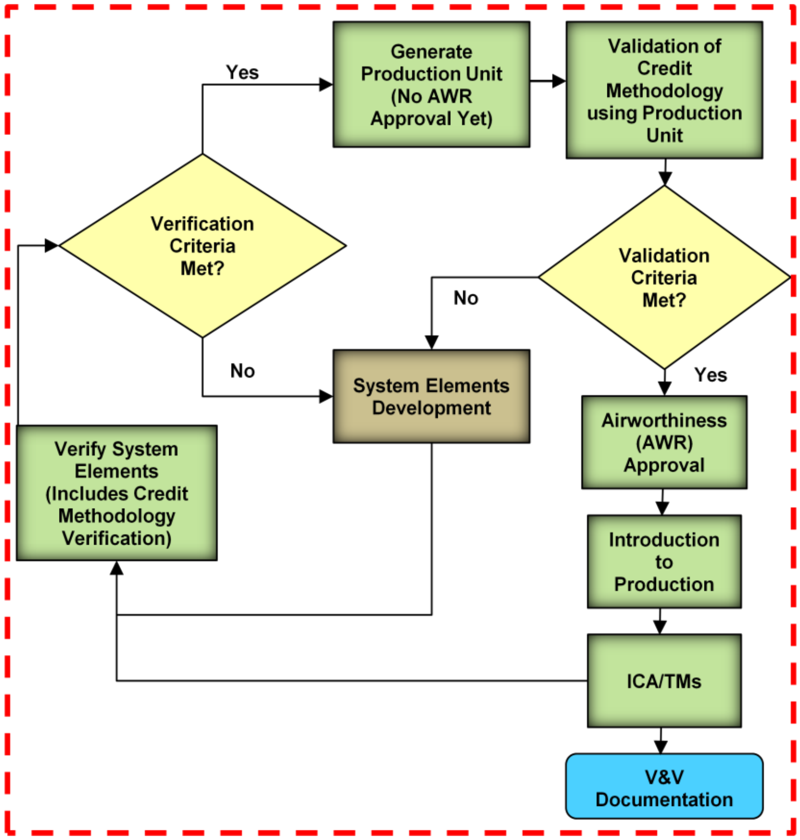
In order to go down the Mechanical Health and/or Usage/SHM path, the candidate HUMS system DSC must have been installed previously and must have collected the appropriate data. If the system was not previously installed, credit for the methodology cannot be acquired. Similarly, for a Prognostics Credit, the candidate system must have a validated health credit in place for the target component.

Sub-Section 4: The determination of the credit methodology leads into performing an *Integrity Assessment* (IA) of the methodology. This assessment should focus on the potential functional failures of the health/prognostics algorithm such as false negatives and false positives. The IA should also spell out the mitigations for these functional failures.



Sub-Section 5: At this point in the process, all the preliminary items required for Maintenance Credit have been established. Thus, this is an appropriate spot to make the decision whether or not to proceed with Verification and Validation activities, based on whether or not a catastrophic failure would occur if the credit methodology fails to perform. The determination of catastrophic failure should have been made previously when the component severity was established. The IA should have spelled out all possible functional failures of the algorithm. If the component severity is catastrophic and the IA does not provide mitigation for an algorithm failure that can misdiagnose this component, then Maintenance Credit cannot be acquired for this component. However, if the IA has mitigation spelled out for all possible functional failures of the algorithm, then one can proceed with the next V&V steps, i.e. establishing the V&V criteria and getting the V&V plan approved.

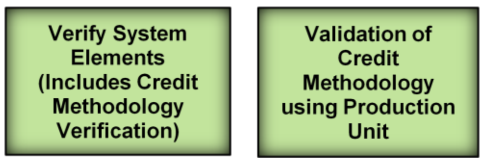
The V&V criteria should describe the plan for verification and validation. It should spell out all appropriate test cases and test plans to complete the verification and validation specifications for Maintenance Credit for the target component. The level of verification will be based on the severity of the component. Once the criteria have been established, the V&V plan needs to be approved by aviation authority, who will determine whether the V&V plan meets the specifications for Maintenance Credit.



Sub-Section 6: The next steps in the Maintenance Credit V&V process are depicted in the flow chart above. The approval of the V&V plan is the entry criteria for the V&V activities. This part of the process begins with the verification of the credit methodology.

Upon completion of the verification steps, determine whether the verification criteria outlined in the plan have been met. If no, then the system element, i.e. the algorithm and corresponding configuration, needs to be redesigned and re-verified. If yes, move onto the next step in the Maintenance Credit process – generation of production unit. It is to be noted that at this point the Air-Worthiness Report (AWR) has not yet been written for the credit methodology.

The next step in the process is validation of the credit methodology. A point to be highlighted here is that validation can only be conducted using the verified production unit. Similar to the verification completion part of the process, it needs to be determined whether the validation criteria outlined in the V&V plan have been met. If no, then the system element, i.e. the algorithm and corresponding configuration needs to be redesigned, re-verified and re-validated. If the validation was successful, then an AWR for the methodology can be written and the unit can be officially introduced into production. For Continued Airworthiness (CA), the credit methodology will have to follow the Maintenance Credit process, starting at verification. To conclude this process, an official document or a series of interlinked documents need to be created that detail the Verification and Validation process.



K.5.1 <u>Mechanical Health Credit V&V Process</u>. Mechanical health of a helicopter can be based on Condition Indicators (CI), or rolled-up Health Indicators (HI). Both the aforementioned methods are similar in nature as they provide an instantaneous look at the health of the aircraft. It is to be noted that in order for the system to get this type of Maintenance Credit, the candidate system must have previously been installed on the aircraft. It also must have collected data for this aircraft. This section will provide a detailed view of the processes of Verification and Validation, two of the steps in the Maintenance Credit Process (as seen in the

Sub-Section 6 figure), for acquiring Maintenance Credit for a component using the Mechanical Health approach.

K.5.1.1 Mechanical Health Credit Verification Process. In this section, a detailed view of the Mechanical Health Credit Verification process (Figure K-6) is provided. This process also discusses certain development steps needed to have a successful verification.

The entry criteria for starting the health credit verification process are: 1) that the candidate system must be installed on the target aircraft platform; and, 2) that the candidate system must have collected data for the credit-seeking component. This data can be collected from different hardware and software versions at an overall system level, as long as the different versions do not affect the incoming data for the target component. For example, say that a HI-based maintenance health credit is being sought for a bearing. Furthermore, the HI-algorithm fuses the four bearing energies (inner race, outer race, ball and cage) and provides a normalized value for health. So the data collected in this case would be the bearing energies. Let's also assume that the overall system has undergone various changes. As long as the system changes do not affect the bearing energies, it is appropriate to use them for HI development. If, however, the system level changes do affect the bearing energy data, then the HI development process can only use the data from the point in time where the change was fielded. If such a situation arises, analysis should be performed to assess the appropriateness of the data. This is to ensure that the development and verification steps that follow will be specific and accurate for the target platform. If these criteria are met, the next step is to obtain the collected data.

At this point it is important to determine whether sufficient data is available. The determination methodology needs to be approved by the aviation authority. If the results of this analysis prove that a sufficient amount of data is present, then the existing database can be used for development and verification. If the results, however, prove that the amount of data is insufficient, then the decision needs to be made whether the potential risk of inaccurate development can be mitigated by either additional testing or by obtaining more data. If the "Design Additional Testing and Approval" path is picked, then the existing database can be used for development and verification, as long as additional tests are designed and approved by AED. On the other hand, if the "Obtain Additional Database" path is chosen, then the additional data need to be gathered and reanalyzed for sufficiency.

Once the database to be used is determined, metrics such as Probability of Detection (POD) and Probability of False positive (PFA) need to be determined. Depending upon the severity established for the component, follow the CI/HI guidelines as outlined in Appendix D of ADS-79D, "Minimum Guidance for Determining CIs/HIs for Propulsion Systems," to determine the POD and PFA. This will lead to the development of CI & HI thresholds and algorithm that should be treated as requirements. Next, capture these requirements in a Configuration Specification Document. Upon completion of the Configuration Specification Document, the algorithm and configuration can be implemented.

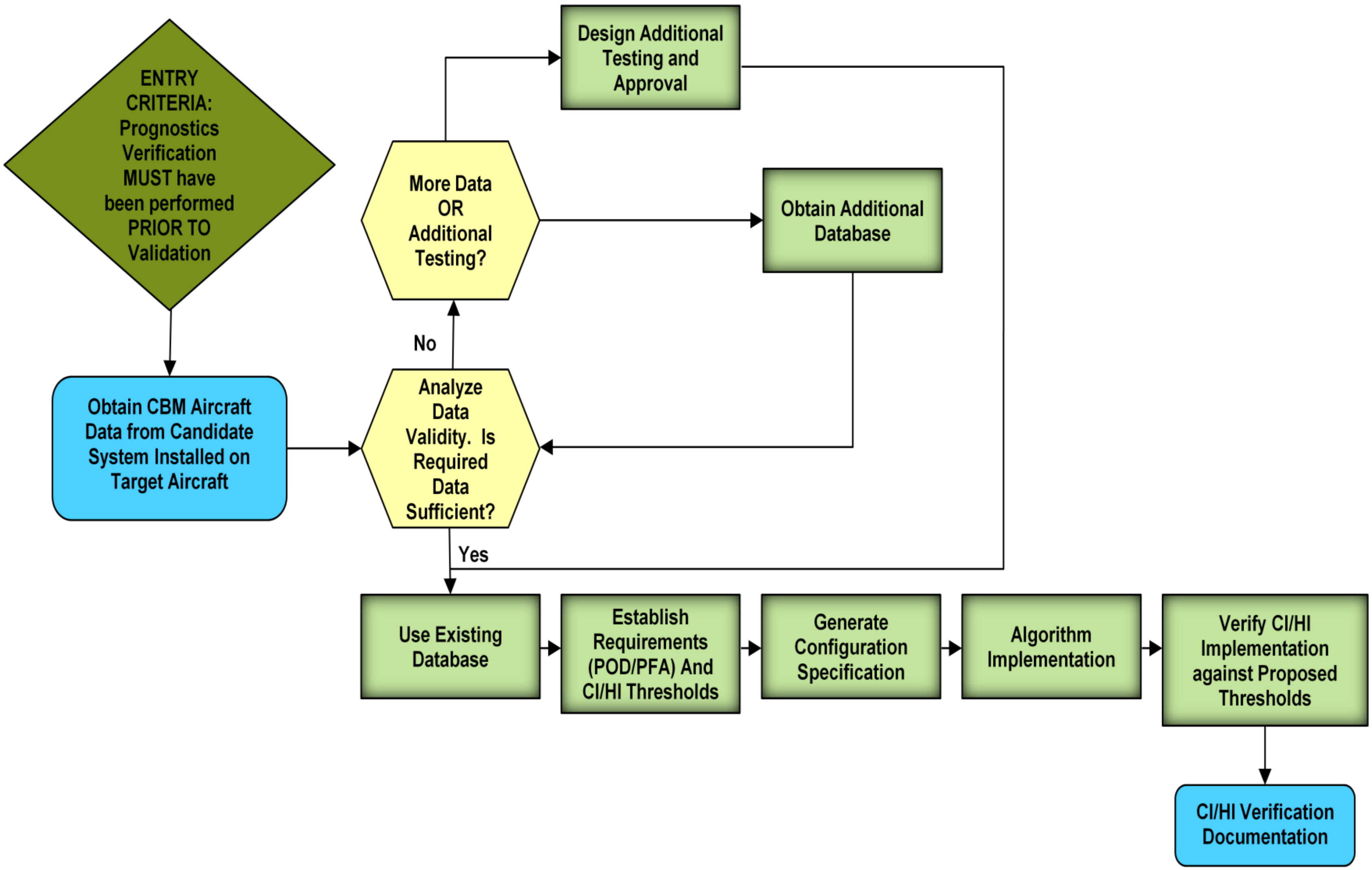


FIGURE K-6. Mechanical health credit detailed verification process

It is important to note that the term verification refers to testing against the requirements as stated in the Configuration Specification Document. The algorithm, configuration and other system elements designed for the component's health credit will be tested to assure correct implementation. Whether or not they function appropriately in the real world application is determined during validation. Given this definition, the "Verify CI/HI Implementation" step only needs to prove that the system elements work as specified. Such verification can be conducted on a bench-type or simulation-type test. If the verification proves that the requirements of the system elements are met, verification is complete. At the end of verification, a "CI/HI Verification Document" needs to be generated that captures the verification tests and process.

K.5.1.2 Mechanical Health Credit Validation Process Upon successfully verifying the credit methodology, the production unit is generated. The production unit is essential for validating the methodology. This section explains the process by which the Mechanical Health Credit methodology should be validated. See Figure K-7 for the flow chart.

The entry criterion for the Mechanical Health Credit Validation Process is that the CI/HI verification process must have been completed. The starting assumption here is that upon successful completion of verification, the system elements will now be production software. Validation can only be performed on production software.

To continue with the validation process, the next step is to establish validation requirements and metrics. Figure K-8 has mentioned a few metrics. At this point, a decision regarding which validation path should be used. Validation of a health credit can be achieved by one of the following three processes: Data Validation, In-Service Validation or Seeded/Natural Fault Test Validation.

Data Validation: To go down this process path, samples of fault cases for the target component should be available. If this data is available, the fault cases can be used to determine whether the CI/HI methodology is able to accurately diagnose the health of the component. However, if such a dataset is not available, one of the other validation processes needs to be followed.

The data validation is successfully complete if the candidate system can accurately (based on criteria established in steps before) diagnose the health of the component. Upon successful validation, the candidate system can acquire Maintenance Credit for the component. A document about the validation process will have to be generated to close out the process. Refer to Appendix D for more details on identifying candidate features.

In-Service Validation: In order to perform in-service validation, the candidate algorithm and software has to be installed on the target platform. Aircraft operations will continue normally. The idea behind this validation process is that the system will acquire Maintenance Credit when it successfully diagnoses the target component's health through normal aircraft operation.

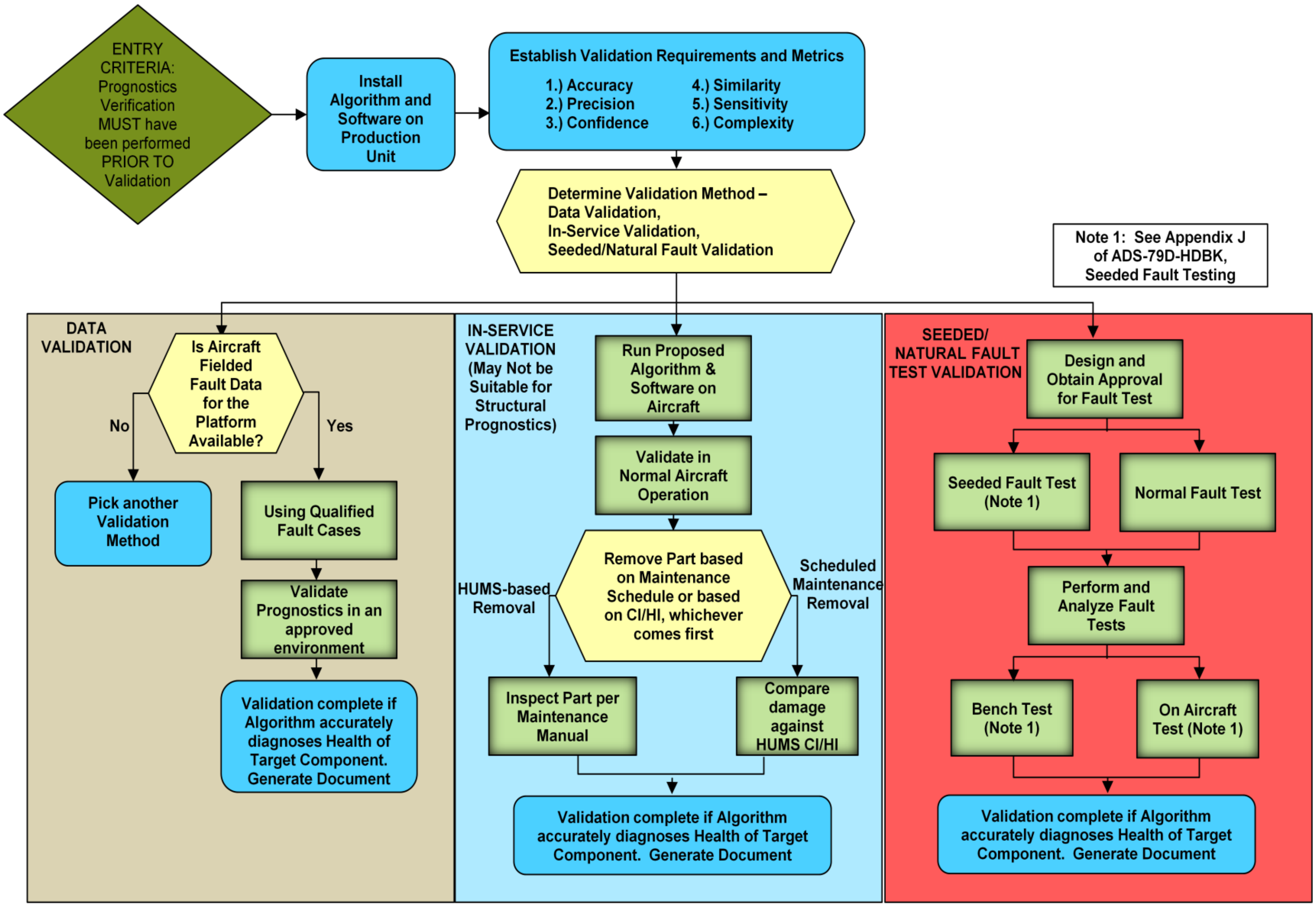


FIGURE K-7. Mechanical health credit validation process

At this point, the question arises, what if the health algorithm doesn't accurately diagnose the component? The fall-back here is regularly scheduled maintenance. If maintenance calls for the part to be removed, it should be removed even if the diagnostics algorithm doesn't say it should. If the part is removed due to maintenance, check whether the algorithm's diagnosis is on par with maintenance. If, however, the part is removed based on the health algorithm's recommendation, then check the part against the maintenance manual to ensure that the removal was necessary. There is no time limit for in-service validation. Refer to Appendix D for more details on algorithm validation.

Seeded/Natural Fault Test Validation: This is a viable option for health credit validation. To start off, validation tests need to be designed by the supplier and the tests need to be approved by AED. These tests can be either a seeded fault test or a natural fault test.

A seeded fault test is where a fault is integrated into the target component and the component is used for the test. In a natural fault test, the component used for the test must have been removed previously from the aircraft due to a fault.

Once the test methodology is determined and the tests approved, it is time to perform the validation. The validation process also has two options: Bench Test or On-Aircraft Test.

Upon successful validation, the candidate system can acquire Maintenance Credit for the component. A document about the validation process will have to be generated to close out the process. For more details on seeded fault testing, reference Appendix J.

K.5.2 <u>Mechanical Health-based Prognostics V&V.</u> The determination of Remaining Useful Life (RUL) of a component based on its mechanical health is the definition of Mechanical Health-based Prognostics. This approach is a viable option for a system attempting to achieve prognostics for mechanical components. It is to be noted that for a system to conduct prognostics, the candidate system must have at least one mature or validated HI for the target component. This section will provide a detailed view of the processes of Verification and Validation for acquiring prognostics for a component using the Mechanical Health approach.

K.5.2.1 <u>Mechanical Health-based Prognostics Verification Process.</u> In this section, a detailed view of the Mechanical Health-based Prognostics Verification process (See Figure K-8 for flow chart) is provided. This process also discusses certain development steps needed to have a successful verification.

The entry criterion for starting the prognostics verification process is that the candidate system must have at least one mature HI established for the target component. This is to ensure that even if the prognostics fails to function accurately, maintenance still can be performed based on health diagnosis.

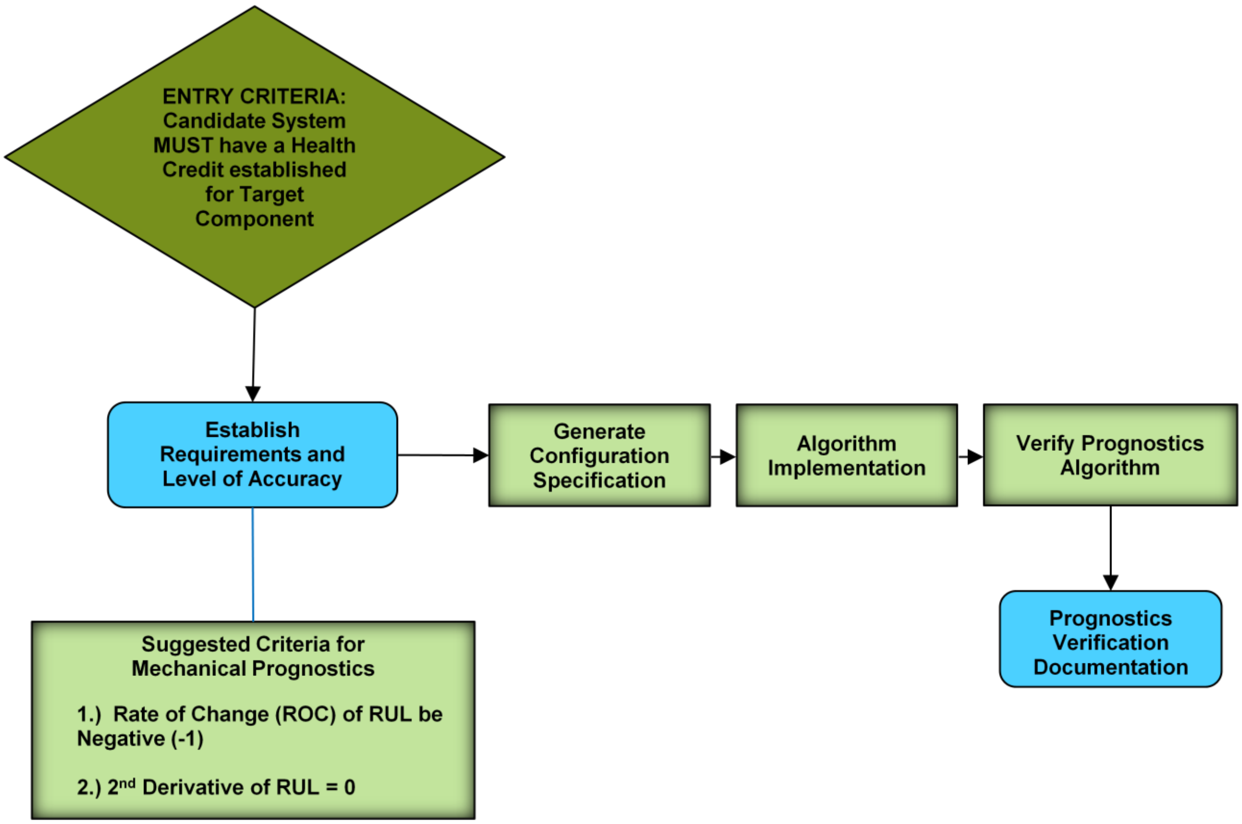


FIGURE K-8. Mechanical health-based prognostics detailed verification process

If the entry criterion is met, the next step is to establish the prognostics requirements and level of accuracy desired. Some suggested prognostics criteria are:

Rate of Change of RUL ≈ -1: In other words, if a component has 100 hours of RUL and it is run for 1 hour, the prognostics algorithm should predict 99 hours at the end of the 1 hour operation.

Second Derivative of the RUL ≈ 0: The value of the second derivative of the RUL is a measure of the stability of the algorithm. The closer it is to 0, the more stable the estimation.

Next, capture these requirements in a Configuration Specification Document. Upon completion of the Configuration Specification Document, the algorithm and configuration can be implemented.

It is important to note that the term verification refers to testing against the requirements as stated in the Configuration Specification Document. The algorithm, configuration and other system elements designed for the component’s health credit will be tested to assure correct implementation. Whether or not they function appropriately in the real world application is determined during validation. Given this definition, the “Verify Prognostics Algorithm” step is only used to test the algorithm against its requirements. The tests should verify and validate that the system elements work as designed. Such verification can be conducted on a bench-type or simulation-type test. If the verification proves that the requirements of the system elements are met, verification is complete. At the end of verification, a "Prognostics Verification Document" needs to be generated that captures the verification tests and process.

K.5.2.2 Mechanical Health-based Prognostics Validation Process. Upon successfully verifying the prognostics methodology, the production unit is generated. The production unit is essential for validating the methodology. This section explains the process by which the Mechanical Health-based Prognostics methodology should be validated (See Figure K-9).

The entry criterion for the Mechanical Health-based Prognostics Validation Process is that the prognostics verification process must have been completed. The starting assumption here is that upon successful completion of verification, the system elements will now be production software. Validation can only be performed on production software.

To continue with the validation process, the next step is to establish validation requirements and metrics. At this point, a decision should be made regarding which of the following three processes should be used: Data Validation, In-Service Validation or Seeded/Natural Fault Test Validation.

Data Validation: To go down this process path, samples of fault cases for the target component should be available. The available fault cases can be used to determine whether the prognostics methodology is able to accurately diagnose the RUL of the component. However, if such a dataset is not available, one of the other validation processes needs to be followed.

The data validation is successfully complete if the candidate system can accurately (based on criteria established in steps before) predict the RUL of the component. A document about the validation process will have to be generated to close out the process.

In-Service Validation: In order to perform in-service validation, the candidate algorithm and software has to be installed on the target platform. Aircraft operations will continue normally. The idea behind this validation process is that the system will acquire prognostics validation when it successfully predicts the target component's RUL through normal aircraft operation.

In this validation method, the health algorithm shall define when a part is to be removed from the aircraft; it should be removed even if the prognostics algorithm does not say it should. When the component is removed, check that the algorithm's RUL estimate is correct based on tear down analysis. There is no time limit for in-service validation.

The prognostics algorithm is considered validated when the system has successfully determined the RUL of an in-service component. A document about the validation process will have to be generated to close out the process.

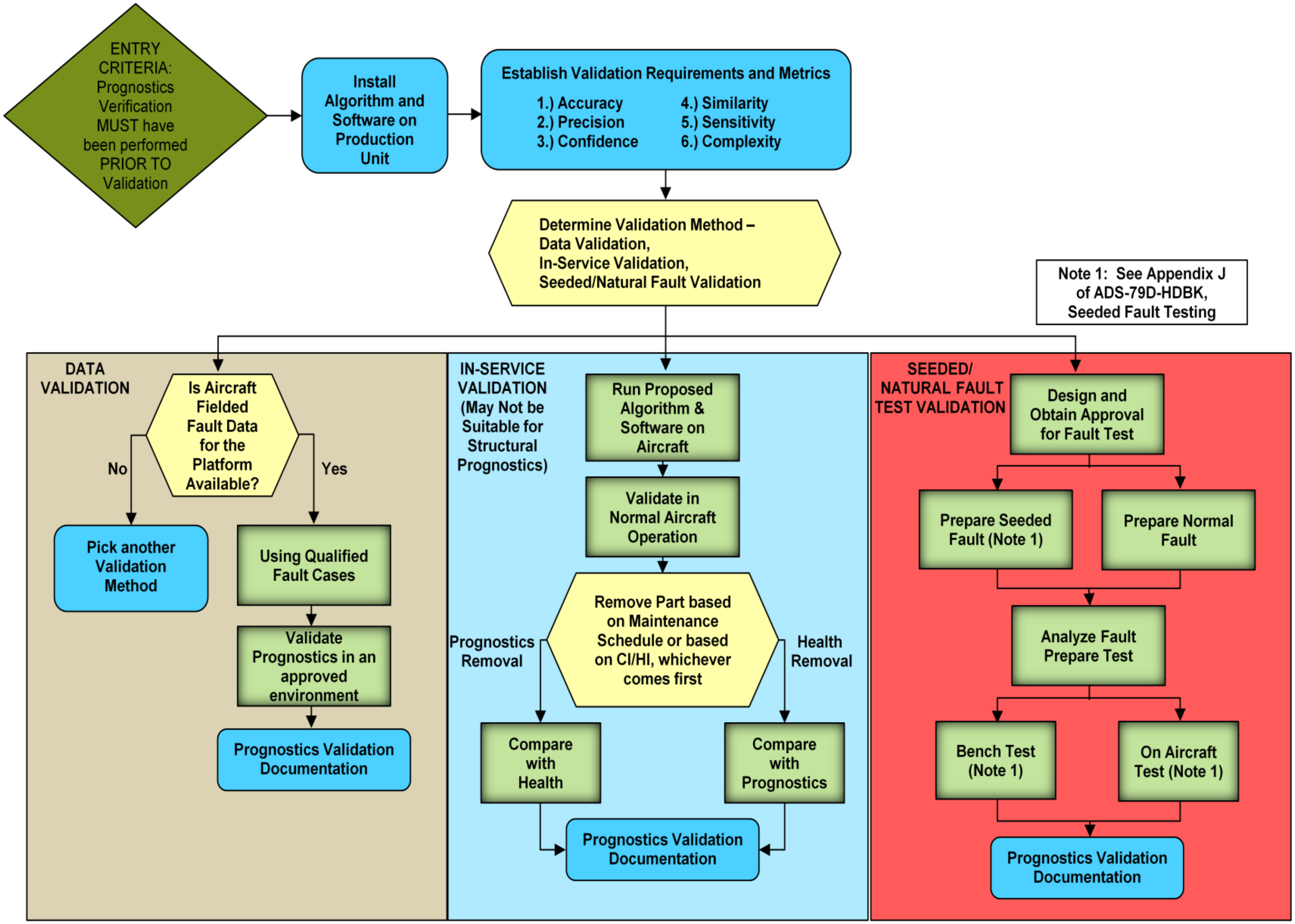


FIGURE K-9. Mechanical health-based prognostics detailed validation process

Seeded/Natural Fault Test Validation: This is a viable option for prognostics validation. To start off, validation tests need to be designed by the supplier and the tests need to be approved by the program manager. These tests can be either a seeded fault test or a natural fault test, as explained in Section K.5.1.2.

Once the test methodology is determined and the tests approved, the validation may begin. The validation process also has two options: Bench Test or On-Aircraft Test. For more information on these two types of test, refer to Appendix J of ADS-79D, "Seeded Fault Testing." A document about the validation process will have to be generated to close out the process.

K.5.3 Structural Usage Monitoring Credit V&V Process. Structural Usage Monitoring can provide Maintenance Credits in many forms including updating design usage spectra, satisfying structural integrity monitoring requirements, providing lifing options such as life factors for isolated sections of the fleet with unique missions, or providing for individual component fatigue life tracking.

Structural usage monitoring can be accomplished using regime recognition algorithms that quantify usage in terms of structurally significant regimes or by direct loads monitoring The achievable benefits or Maintenance Credits from usage monitoring are dependent on the availability of aircraft data parameters and the availability and quantity of data available across the fleet. This concept is shown notionally in Figure K-10. In the figure, the vertical axis

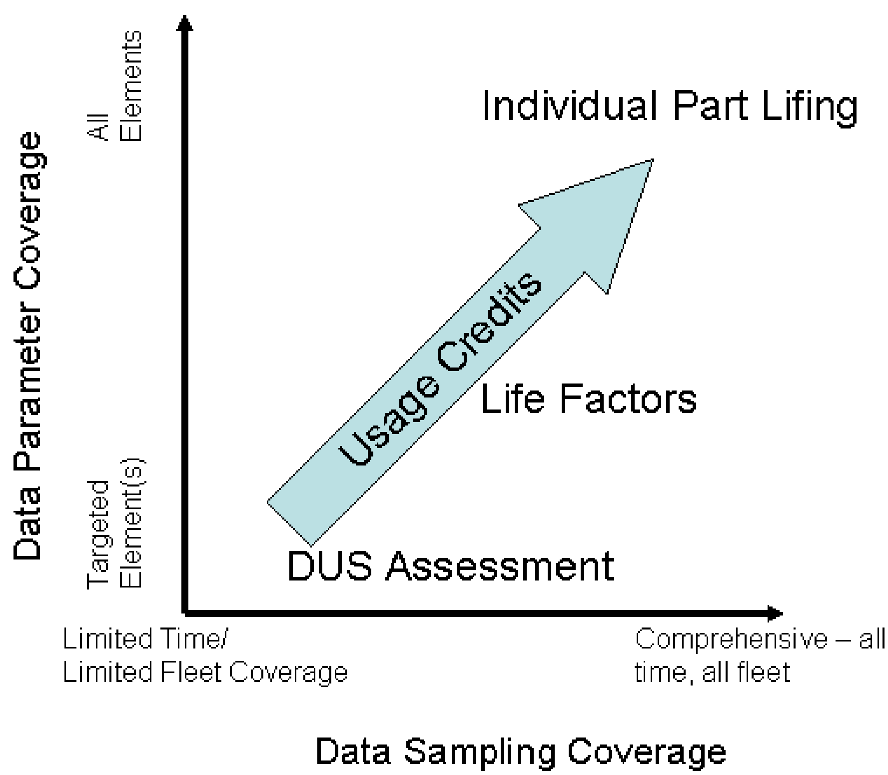


FIGURE K-10 Influence of data parameter coverage and data sampling coverage on achievable usage monitoring credits.

represents the availability of aircraft parameters used to quantify usage. For regime recognition based usage monitoring, the parameter coverage could range from a limited number such that only one or more key fatigue drivers can be recognized to a full complement which provides comprehensive regime recognition. The horizontal axis represents the extent to which available data provides coverage over time and across the fleet.

Under the traditional fatigue life management process, the Design Usage Spectrum (DUS) is utilized both in the design phase of a component and for fatigue life management of the component where a retirement time in terms of flight hours is established based on a calculated fatigue damage fraction associated with the DUS. The design usage spectrum is defined as a so-called "composite worst case" for the fleet in terms of variability across the fleet and with some consideration for potential variability over the fleet lifetime (mission creep). With the advent of structural usage monitoring, additional Operating Usage Spectra (OUS) can be defined for fatigue life management. The Maintenance Credits associated with the use of operating usage spectra are dependent on the sampling and parameter coverage. Possible examples include:

A more accurate fleet wide operating usage spectrum used to establish flight hour retirement lives for application across the fleet. Based on fleet sampling or comprehensive monitoring with scheduled reviews to protect against time variability, this operating spectrum should be less severe than the DUS.

A fleet wide operating usage spectrum in combination with an applied life factor with a local operating usage spectrum. The local operating environment could be more severe (training or deployed) and the use of an applied life factor for flight time in time in the local operating environment would allow for a less severe usage spectrum to be applied to the fleet in general.

With comprehensive usage monitoring across the fleet and with a full complement of data parameters being collected, it becomes possible to implement a fatigue life methodology that incorporates individual part fatigue lifing based on the actual part usage. Individual part fatigue lifing requires both the tracking of usage corresponding to the component in-service time and the correlation and tracking of the calculated fatigue damage fraction associated with the tracked usage.

Considerations of data parameter coverage, data sampling coverage, and the target usage credit should be included as part of the *Integrity Assessment* (IA), the establishment of the V&V criteria, and the V&V Plan.

For usage spectrum monitoring applications, calendar time related considerations can be important as summarized in Figure K-11. Usage is typically tracked by aircraft tail number. Usage can vary among aircraft, among units, and among operational theaters. Usage can also vary over time. Although usage is tracked by aircraft, the usage spectrum is applied to individually tracked components.

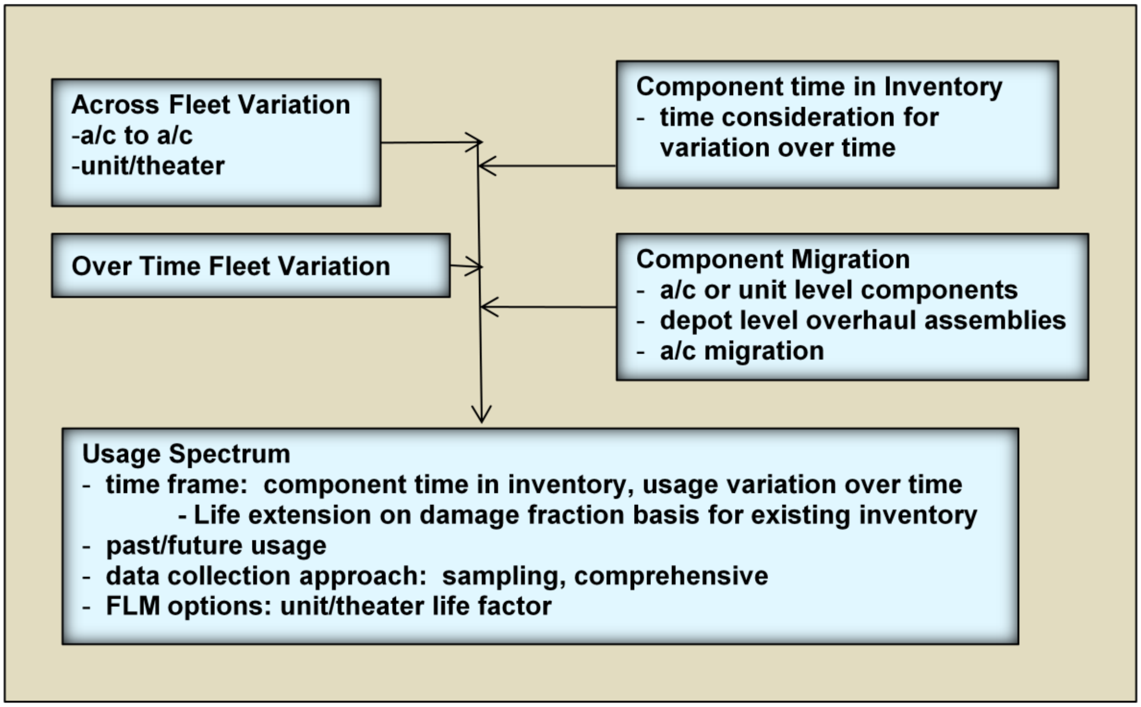


FIGURE K-11. Examples of time related considerations for usage spectrum monitoring.

Some of these components remain on a single airframe over their service life while others can migrate between aircraft within a unit (as a unit asset) and others can migrate anywhere within the fleet over their service life. This interaction of usage variability and component migration should be considered in data collection for spectrum updating and fatigue life management approaches used for the introduction of updated spectra and other options such as using fatigue life factors for unique fleet populations.

K.5.3.1 Structural Usage Monitoring Credit Verification Process. In this section, a detailed view of the Structural Usage Monitoring Credit Verification process is provided. This process also discusses certain development steps needed to have a successful verification.

The entry criterion for starting the usage credit verification process is that the target usage credit has been identified and the feasibility with regards to data coverage has been assessed as shown in Figure K-12.

The selected usage credit could be limited to usage tracking only, or could also include calculated part fatigue damage fraction tracking. Calculated fatigue damage fraction tracking is a requirement for usage credits where individual components are retired based on actual usage. Guidance on the verification requirements related to the fatigue life management (FLM) elements of the identified credit can be found in Appendix A. Guidance on the requirements related to the regime recognition elements of the identified credit can be found in Appendix B.

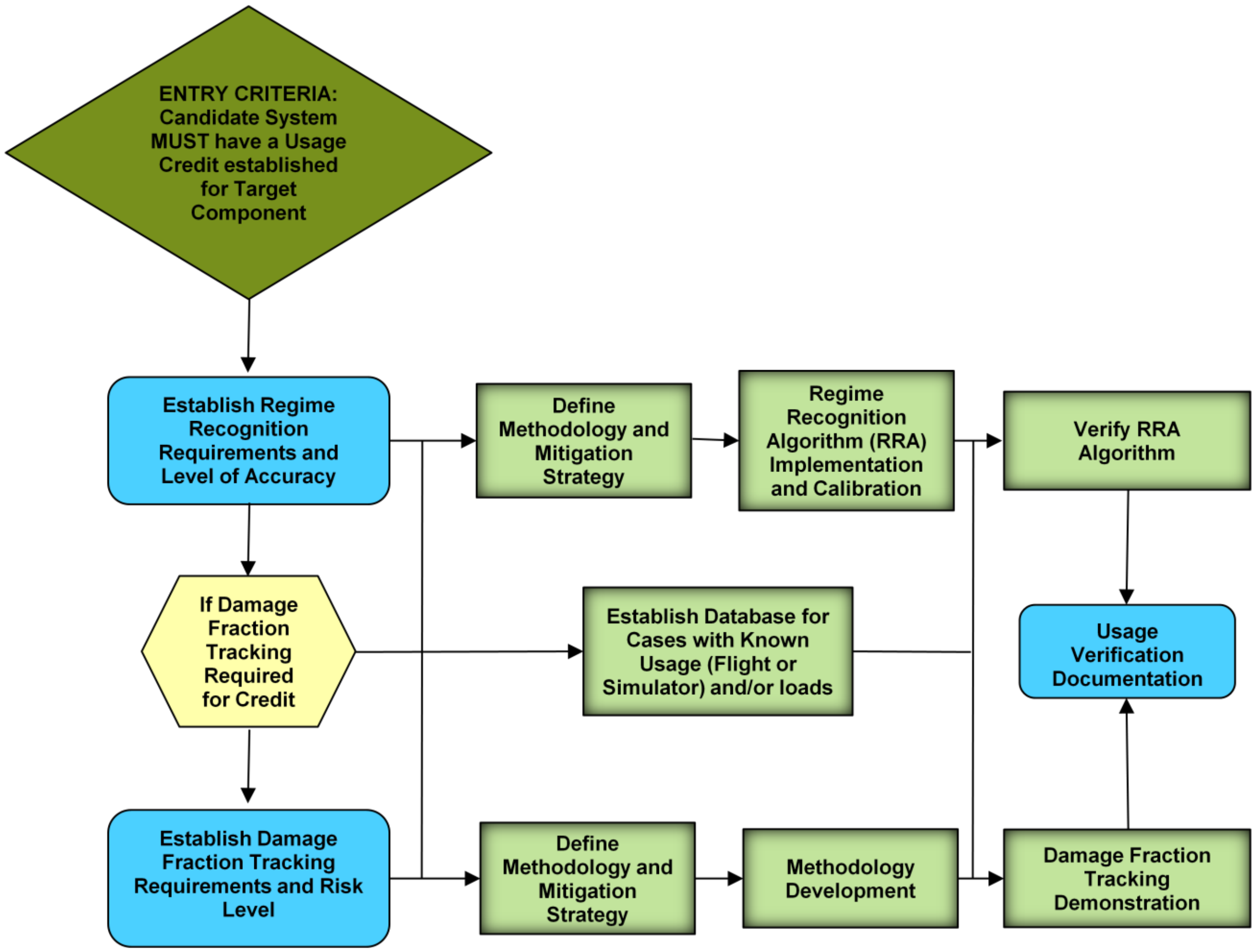


FIGURE K-12. Regime recognition-based structural usage monitoring credit detailed verification process

K.5.3.2 Structural Usage Monitoring Credit Validation Process. Upon successfully verifying the credit methodology, the production unit is generated. The production unit is essential for validating the methodology. This section explains the process by which the Usage-based Credit methodology should be validated. See Figure K-13 for the flow chart.

The entry criterion for the Usage-based Credit Validation Process is that the verification process must have been completed. The starting assumption here is that upon successful completion of verification, the system elements will now be production software. Validation can only be performed on production software.

Guidance on the validation requirements related to the fatigue life management (FLM) elements of the identified credit can be found in Appendix A. Guidance on the requirements related to the regime recognition elements of the identified credit can be found in Appendix B.

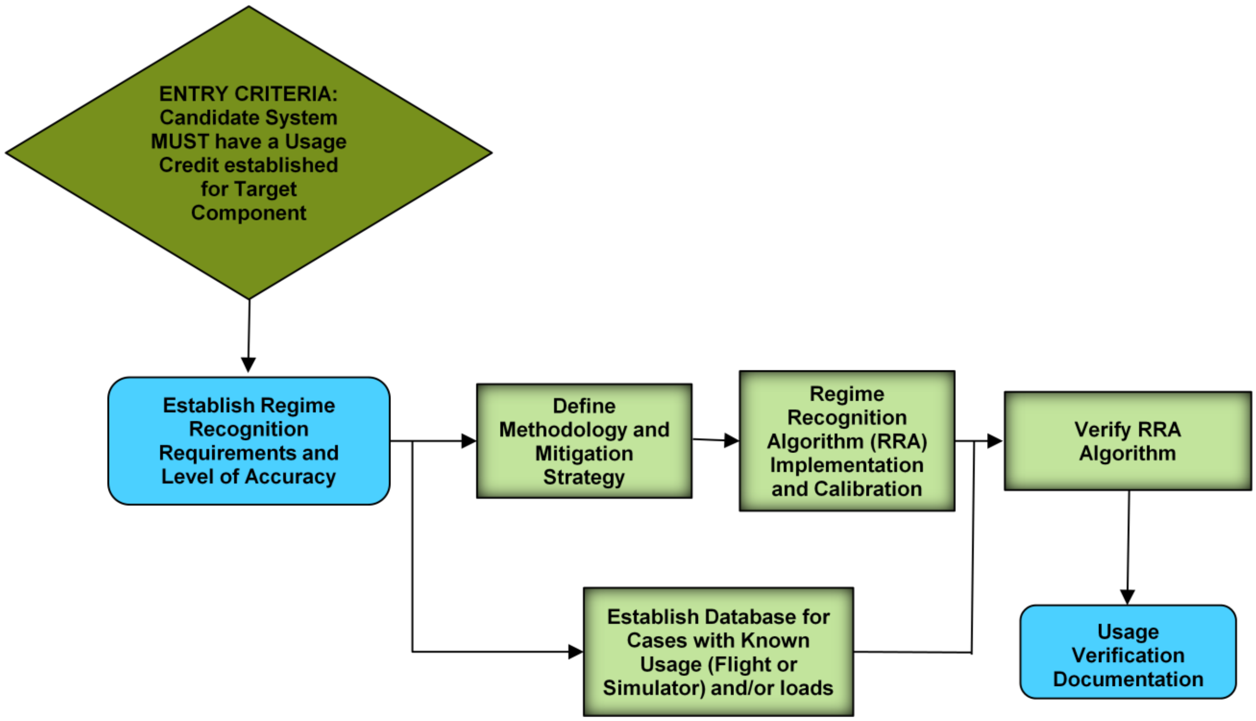


FIGURE K-13. Structural usage monitoring credit detailed validation process

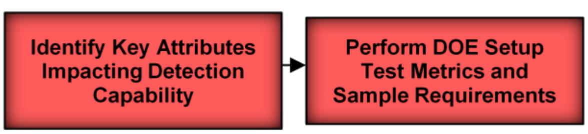
K.5.4 Structural Health Monitoring Credit V&V Process Structural health monitoring (SHM) becomes emergent technology in monitoring and assessing underlying structural integrity throughout its designed life cycle using monitored structural health data. A comprehensive structural health monitoring system contains several elements for structural integrity evaluation and capability assessment, including load tracking, virtual load monitoring, and damage detection. A qualified structural health monitoring methodology can provide Maintenance Credits in reducing or eliminating unnecessary inspections, extending inspection interval or time of overhaul / removal, and extending component design life. One of the key challenges to achieve aforementioned Maintenance Credits for structural health monitoring is establishing a process and technical path to verify and validate the SHM credit methodology.

Several structural damage detection technologies have been developed and applied in aerospace industry, including NDE methods and structural health monitoring sensors. In general NDE can be applied in-situ or at depot, while structural damage detection sensors are typically operated in an on-board basis. Owing to the non-destructive requirement, the damage detection sensors are regarded as a special case of integrated NDEs. The more detailed discussion of structural health monitoring methodology can be found in Appendix C. Due to the maturity of various structural health monitoring technologies, our discussion in this section will focus on structural health monitoring via damage detection.

K.5.4.1 Structural Health Monitoring Credit Verification Process. In this section, a detailed view of the Structural Health Credit Verification process is provided and the associated process flowchart is depicted in Figure K-14. The focus of discussion is on the verification of credit methodology assuming basic methodology developmental efforts have been performed and integrated into a candidate system. The candidate system must be installed on the target platform and must be ready to collect additional data for the credit-seeking component. Due to the complexity of structural health monitoring technologies and desire of improving level of maturity, the process also includes certain development steps needed to have a comprehensive verification.

In general, the process of verifying the credit methodology for structural health monitoring looks similar to the one associated with mechanical health monitoring. However, structural health monitoring possesses some unique technical challenges and therefore several additional steps need to be followed in its credit verification process. These steps are highlighted in the dashed oval in the process map and discussed below.

Structural damage detection capability is regarded as a critical aspect of overall damage tolerance (DT) based product design and operation management approach. Often, the requirements of detection and thresholds vary from one application to another. Sometimes, a characteristic detection capability, such as a_{NDE} (for example, $a_{90/95}$ representing detectable damage size of 90% probability of detection with 95% of confidence), would be sufficient to represent typical damage detection capability. In other cases, a more comprehensive requirement may follow, such as entire probability of detection curve as a function of damage size. Clearly, the difference in detection requirements dictates different level of rigor in V&V process. Therefore, the detection requirements and thresholds should be clearly defined at the beginning.



Once the required damage detection metrics and associated thresholds for the candidate application are defined, the next step is to identify the key attributes impacting desired detection capability. It is quite often that the credit methodology has been developed and further verified in sub-component / component which are the same as the credit seeking component. In other cases, the developmental work has been performed on similar components and the associated variation on similarity may alter the detection capability. This step is essentially to provide additional means to ensure all the key attributes which affect the detection capability are fully addressed for the subject credit seeking application. If additional factors are identified, design of experiment (DOE) needs to performed to setup test for additional experimental data.

Upon the completion of the aforementioned steps, the rest of the steps in Figure K-14 similar to the path used in Figure K-6 for Mechanical Health Monitoring Verification.

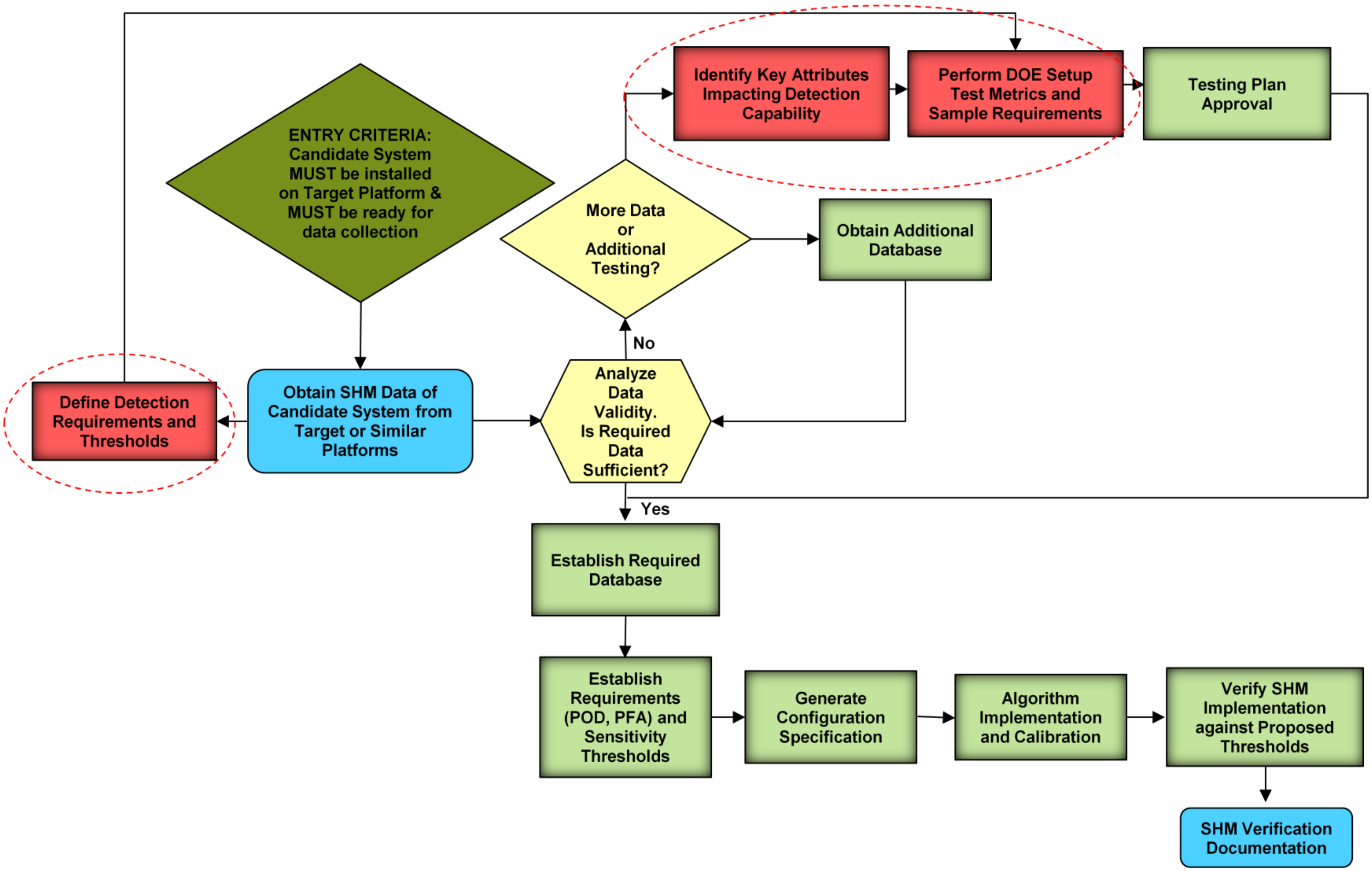
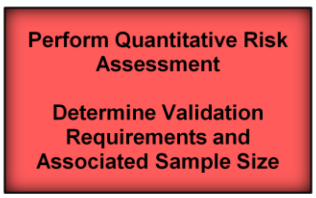


FIGURE K-14. Structural health monitoring credit detailed verification process

K.5.4.2 <u>Structural Health Monitoring Credit Validation</u> Process Upon successfully verifying the credit methodology, the production unit is generated. The production unit is essential for validating the methodology. This section explains the process by which the Structural Health Monitoring - based Credit methodology should be validated. A flowchart illustrating the process is presented in Figure K-15.

The entry criterion for the Structural Health Monitoring - based Credit Validation Process is that the verification process must have been completed. The basic assumption here is that upon successful completion of verification, the system elements, including both software and hardware, will be fully integrated into suitable production unit.

The validation process of SHM highlighted in Figure K-15 resembles the one in Figure K-9 for Validation of the Mechanical Health Monitoring. The most significant difference is that the SHM process begins with performing quantitative risk assessment.



One of the key characteristics associated with structural health application is related to its severity of the application. In general, structural health monitoring application is associated with the structural components possessing significant hazard to structural safety. Therefore, a more rigorous approach in determining the risk associated with potential failure of credit methodology and mitigation plan to manage the risk needs to be defined and followed. In this step, the risk associated with failing to perform required damage detection will be quantified first. The obtained information will be used in subsequent reliability allocations to establish target structural reliability. In addition, uncertainty quantification of damage detection system should be performed to establish confidence bounds of potential detection variation. Based on aforementioned assessment, validation requirements and associated sample size for the pertinent credit application are established. The remaining steps follow a similar path as used for Validation of Mechanical Health Monitoring. The final validation can be achieved via the Fielded Data Validation, Seeded Fault Testing, or In-Service Validation. Caution should be taken in the Seeded Fault Validation option. If the subject application has significant impact on flight safety and the anticipated damage growth rate is not slow enough in vicinity of the detection threshold, seeded fault validation with flight test may not be a viable choice. For specifics and examples on seeded fault testing, reference Appendix J.

K.5.5. <u>Structural Prognostics.</u> For structural elements and components which are critical safety items, prognostics involves setting appropriate component retirement intervals and inspection intervals to maintain component reliability, as discussed in Appendix A. Similarly, prognostics for damage tolerant structure (including slow crack growth structure, fail-safe multiple load path structure, and fail-safe crack arresting structure) is based on setting service lives and inspection intervals, as discussed in Appendix A. In either case, these intervals depend on usage monitoring and integrated-NDI based structural health processes discussed in Appendices B and C. Verification and validation of these usage monitoring and structural health processes discussed earlier in this appendix are necessary to enable structural prognostics.

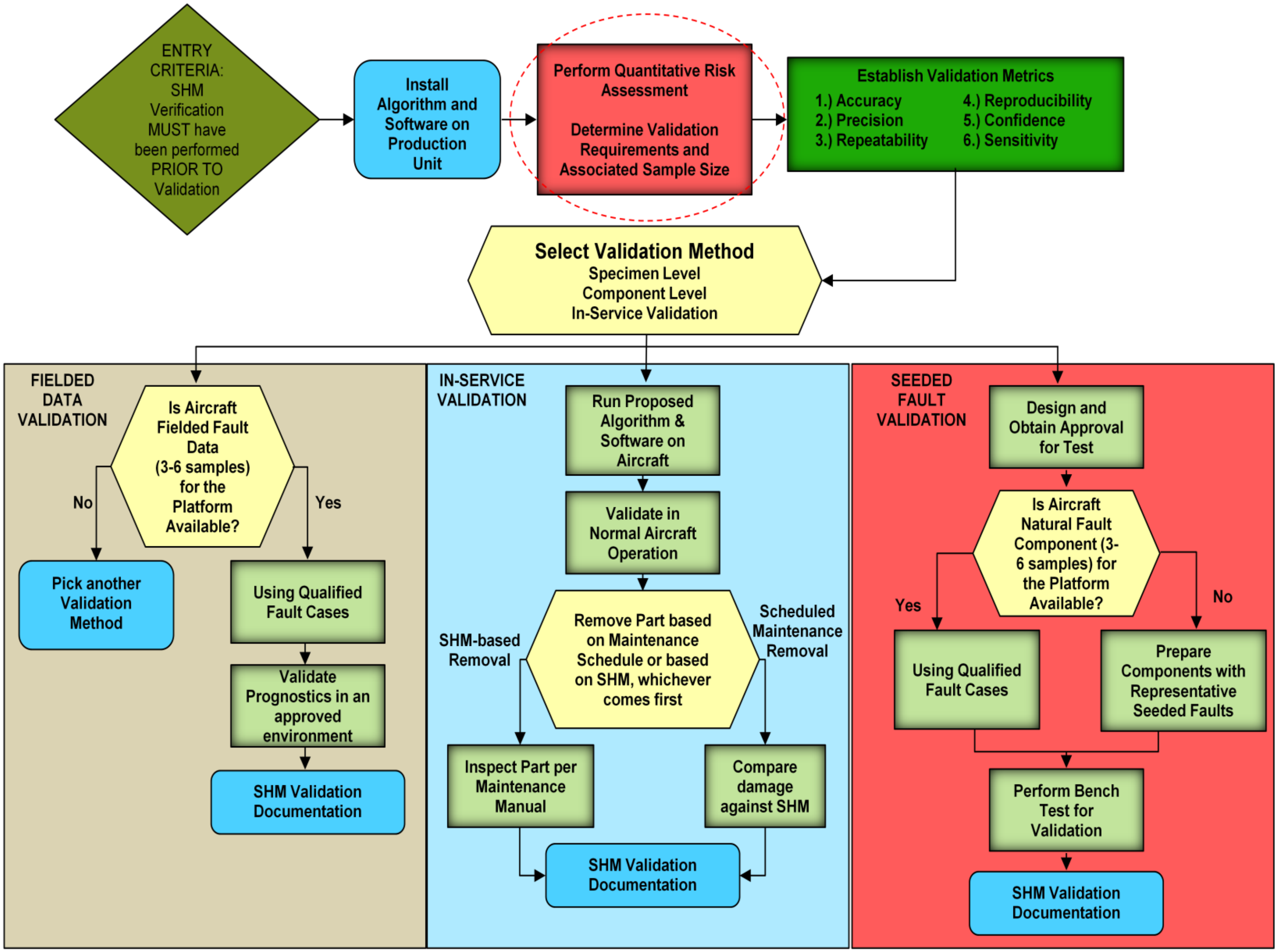


FIGURE K-15. Structural health monitoring credit detailed validation process

Structural reliability is based on the number of inspections and the probability of detection of each inspection. It is noted that slow crack growth damage tolerant structure is difficult to achieve in a rotorcraft load environment. As such, prognostics for US Army rotorcraft structure are not considered to include continued flight with known damage beyond any substantiated flaw tolerance for the component.

One method of improving structural prognostics would involve developing and introducing improved integrated-NDI (or traditional NDI) capabilities. Typically, NDI capabilities may be improved by introducing new methods, techniques, and equipment which reduce the size of detectable damage (for example the crack length with 90% probability of detection with 95% confidence).

A second method of improving structural prognostics would involve incorporation of individual component fatigue damage assessment into fleet management. By trending accumulated fatigue damage and seeking methods to correlate fatigue damage to controllable aspects of aircraft usage, commanders would be able to more efficiently and proactively manage aviation assets. Component replacement could be scheduled prior to component life expenditure based on the observable trends. Reduction in usage scatter would improve the prediction capability and increase the horizon to scheduled replacement.

A separate V&V Process for structural prognostics is not required. Although not an airworthiness concern, it is recommended that the general V&V Process described in Figure K-5 be followed for any formalized infrastructure investments in accumulated fatigue damage trending and fleet management tools which incorporate use of individual fatigue damage assessment data to predict fatigue life expenditure.

K.6. GROUND STATION VERIFICATION AND VALIDATION PROCESSES

One of the goals and desired benefits of HUMS is to be a transparent collector of information during normal operational missions. As such, the methods used to determine component health and aircraft usage statistics, and the approaches designed within the system in the form of algorithms and processing, should be shown to be proven and effective for collecting and analyzing such data. A very high correlation between the data collected and the HUMS-determined health of a specific component should be demonstrated to validate the data collection method and to verify the proper operation of the system. The Ground Station is designed to be the database and software tool where operational data and usage statistics are stored and analyzed to determine component health.

This section also outlines the methodologies used to verify and validate Ground Station (GS) systems to support overall HUMS functionality on Army aircraft. References to RTCA DO-178B will be used throughout this section to outline verification processes in relation to ground station approval.

Verification guidance and specific procedures developed for verification should remain consistent with the potential impact to the system of erroneous operation or hidden fault modes. Systems monitoring components that have a high degree of criticality to the safe and continued operation of the mission or aircraft should be subjected to a higher degree of verification than systems with less impact. The degree of scrutiny applied to the verification effort should be in step with the effects that could be generated by an improperly operating system. In the civil arena, a Functional Hazard Assessment (FHA) supplies the basis for determining the criticality of any given component within the system. A similar approach which provides a documented basis for the level of rigor needed when certifying HUMS, and the Ground Station specifically, is recommended to ensure that the cost of certifying a particular system is not out of line with the system's potential impact should a failure occur. For more information regarding this section see RTCA D0-178B, section 2.2.

K.6.1 Ground Station Verification. The Verification Process is designed to ensure that the developed system satisfies its requirements. In other words, it answers the question "Does the system do what we said it should do?" In normal practice, the customer and developers collaborate on the requirements defined for the system. The developer then begins implementing a system to satisfy those requirements, and the customer is then responsible for accepting the developed system and validating that it performs as required in the actual target environment. In contrast to verification, validation is the process of answering the question "Does the system do what it is supposed to do?"

This section of the document provides guidelines for verification of the system, which is usually performed by the developer of the system prior to delivery to the customer. The next major section covers the issues involved with validation.

K.6.1.1 Ground Station Requirements-Based Verification. For systems that have been designed using requirement-based development, requirements-based verification is a perfect fit. By verifying that the requirements for the system have been implemented, this phase of the process provides the necessary proof that the system performs as intended and as specified. Test Procedures, in the form of a Software Test Plan, which verify specific requirements, can be developed early-on in the development process, independently of the design and implementation of the software. Traceability between the design (requirements) and test (procedures) can be established and maintained to show full compliance of the system to its design. Test Reports can be developed to document the results of the verification process, and highlight any product deficiencies down to the design/requirement level if necessary.

K.6.1.2 Ground Station Software Verification Procedures and Cases. By focusing on requirements-based verification, software verification cases and procedures can be developed early-on and in parallel with software development (process shown in Figure K-16). Since the requirements for the system are established up-front, verification activities can proceed without the need for significant input from the actual software until the time that integration activities are performed.

In a typical development cycle, once the requirements for that cycle have been established, the development and verification teams can proceed independently and meet up at the integration and formal verification stages.

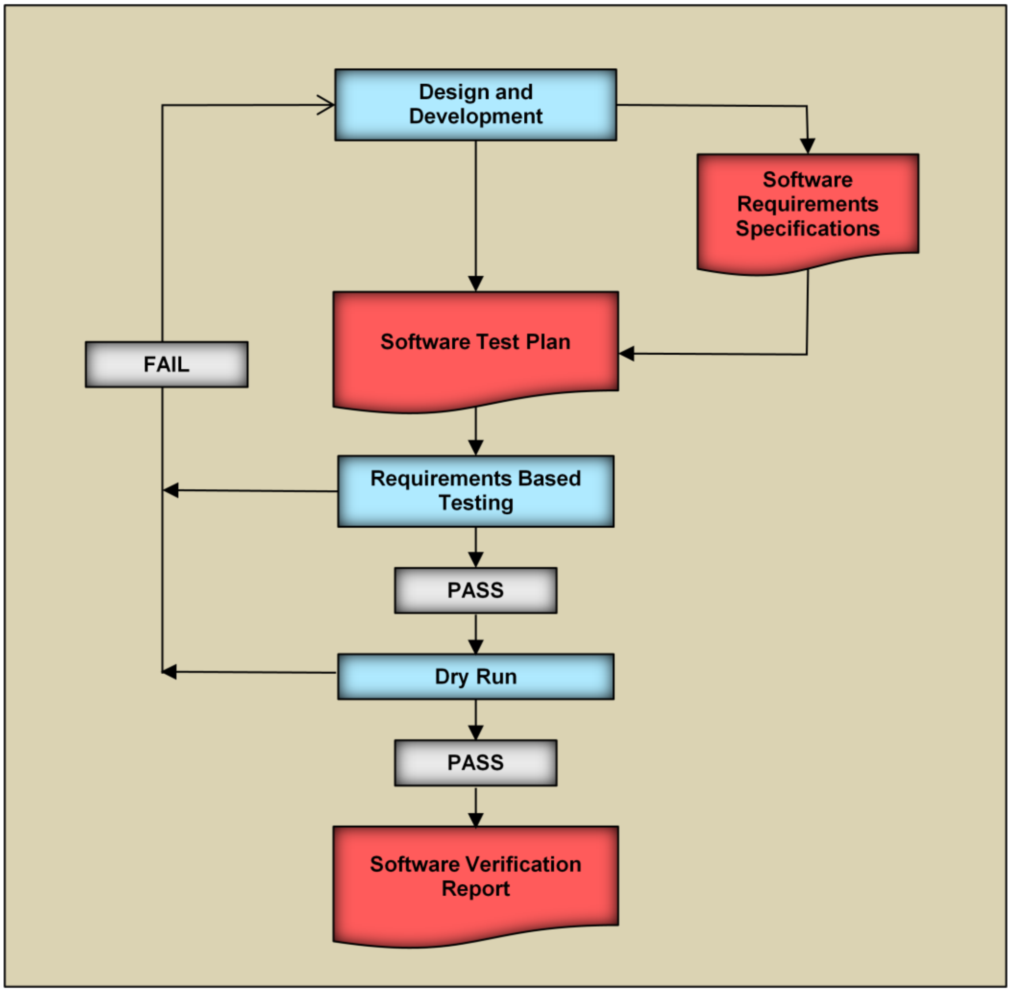


FIGURE K-16 Software verification process

The verification team is responsible for generating the test cases necessary to verify the established requirements for the system. RTCA DO-178B encourages testing at the highest level of integration possible. In other words, verify requirements at the system (end-to-end) level if at all possible and only proceed to lower integration levels (CSCI, CSC, etc.) as necessary to establish the required verification coverage. To this end, the verification team should focus on establishing test cases that verify requirements at the highest integration level possible.

An analysis of test case coverage against the high-level requirements is required for any criticality at Level D or higher. More extensive test coverage is required for higher criticality levels. This analysis is often documented in the form of a traceability of test cases to requirements, which is an excellent method for confirming test case coverage. Traceability not only identifies which test cases cover which requirements, but also provides an easy-to-understand method of identifying any missing test cases or unverified requirements.

Once the appropriate test cases have been allocated, test procedures can be developed to support the verification effort. Test procedures provide the detailed instructions to perform in order to verify the software under test. When a specific output is required for verification, that value should be specified in the test procedure along with any necessary tolerances (+/- values) to establish the PASS/FAIL criteria for each test. The test procedures also document the environment necessary in which to conduct the test; any prerequisites for the test; the pre- and post-test procedures required to perform the test; and any unique conditions, data, or environmental considerations required for the test.

Just as an integration phase is usually necessary to complete the software design phase, a ‘dry-run’ phase is usually necessary to complete and confirm the development of test procedures. Since test procedures are very detailed instructions on how to execute and manipulate the software under test, it is usually necessary to execute these procedures informally a few times in order to evaluate the procedure and make any necessary adjustments and clarifications prior to release and formal test execution.

K.6.1.2.1 Ground Station Automated Verification. When allowed by certificating authorities, automated tools to assist with the verification of the software under test are encouraged. Automated verification enables the developers to concentrate on desired changes rather than running test suites repeatedly. By automating the verification of the system, it can be tested more extensively and faster than via manual methods. Guidance for verification tools can be found in Section 12 of RTCA DO-178B.

K.6.1.2.2 Ground Station Regression Testing. As a part of verifying new functionality, any change to the system should also be considered a candidate for regression testing to ensure that functionality not intended for modification at a given change has indeed not changed. This is even more important for systems whose behavior depends on the aircraft model or other external indications. For more information on modifications to previously developed software reference RTCA DO-178B, section 12.1.1.

K.6.1.3 Ground Station Software Verification Results. The output of the verification process is captured in the Software Verification Results (also known as a Software Verification Report (SVR) or Software Test Report). The SVR documents the entire test environment, the unit under test (UUT) and its configuration, and the results of the execution of the software test procedures. If any failures occur during test, the information regarding those failures is captured, and documentation of the failure is captured (for example the creation of a Software Problem Report or equivalent defect tracking system input). The SVR provides a single source for determination of whether the unit under test meets its requirements or is deficient in any way. The airworthiness authority uses this document to either accept or reject the product based on the results of verification activities performed.

K.6.2 Ground Station Validation Process. As stated earlier, validation answers the question “Does the system do what it is supposed to do?” It occurs sequentially following verification (which is assumed to already have proven that the system functions as it was specified to work) and is vital to the fielding of a completed system because it focuses on end-user functionality in the designed target environment, rather than on simulated environments and test cases. As such, validation is typically performed by the end-user or customer representative, on actual or representative hardware running in the end-user’s normal hardware environment.

Validation is the final phase of development and confirms the ability of the developed system to perform in real-world conditions.

K.6.2.1 Ground Station System Validation. System Validation represents the final V&V phase for developed products. The purpose of system validation is to determine whether the system is indeed performing the functions for which it was created, in the real-world production environment.

The system should be installed, calibrated, and operated in the production environment using established training and maintenance procedures, run by the end-user operators for which it was designed. Consideration should be given to introducing this stage of validation in a controlled manner (for example, a designated unit or units) and ensuring that all personnel involved with the system receive adequate training and preparation for its use. The time period for system validation should be long enough to establish the system in routine use, and to evaluate the operations and maintenance recommendations it will generate during that period.

Supplier input and assistance during this phase should be limited to that which would normally be provided to any production environment. If Contractor Logistic Support (CLS) is envisioned for the system, CLS personnel should be allowed to participate to the same extent they would in the production environment. If the system is designed for organic support, then the necessary personnel should be available to supply that support. There should be no difference between the supplier and customer support provided to the unit(s) in validation and those used in any other production system.

In order to benefit from the system validation phase, a report should be generated at its conclusion highlighting the degree to which the system performed as intended, and identifying any issues or problems with the system, as well as any unanticipated benefits or other unplanned impacts. By providing an objective evaluation of the overall system, the validation report can serve as a basis for planned updates, future modifications, or enhanced attention to the product. It can also serve as a baseline scorecard for the development effort to assess its overall degree of success in meeting the customer's intent.

A successful system validation would normally be followed by introducing the system into the full production environment. The airworthiness authority should evaluate the system validation report and provide its guidance on fleet-wide implementation and the need for any subsequent modifications or alterations to the system.

Experimental or developmental approaches and methods, though certainly useful for the continued development of HUMS technology, should be either isolated within the data collection and reporting system (i.e. 'background' data collection or analysis performed which is transparent to the end user and which does not interfere with the normal operation of the system), or should be assigned to test aircraft on an as-available basis.

Similarly, on the Ground Station, acceptable approaches and methods related to software development, verification, configuration management, and application maintenance should be employed as a means of developing, fielding, and maintaining certifiable verified and validated software. Guidance in the form of RTCA DO-178B and similar publications can provide a background for acceptable approaches towards the development and verification of software applicable to the entire system.

In addition, an alternate means of compliance to RTCA DO-178B may be pursued. Since the function of the HUMS may essentially perform nondestructive inspection (NDI) on critical components – rather than control the aircraft or provide actionable information to the pilot for control of the aircraft – the HUMS software requirements may not necessarily include compliance with RTCA DO-178B level A or B DAL. Instead, similar to other NDI equipment employed in aviation, the HUMS may undergo a periodic functional check via fault stimulation to verify it will display the applicable alert to the maintainer when a fault is present on critical components. Analogous to stimulating an eddy current or ultra sound machine using a pre notched/cracked block to ensure proper equipment function before inspecting components, stimulation software with known fault signals may be executed on ground stations. The stimulation software can consist of raw data from a known fault obtained from seeded fault validation and fed into the ground station to assure the applicable alerts for known faults are provided as output alerts from the ground station. If applicable alerts are not provided by the ground based stations, then the ground station software is not functioning properly. Otherwise, ground station functionality is verified and download of raw data from the aircraft may occur with confidence the ground station software is providing the correct output.

K.6.2.2 Ground Station Acceptance Test Procedures (ATP). Whether deploying a newly-developed system for the first time, or updating an existing system that is already in production use, some method of determining the acceptable performance of the delivered system is necessary. Acceptance Test Procedures (ATPs) are the recommended method of verifying system functionality whenever an update or modification to the system is fielded.

Unlike the Verification phase testing, ATPs generally are simpler tests which provide the end-user confidence that the updated system is performing adequately and has not been damaged or incorrectly altered by the modification. ATPs are designed to exercise the standard functionality of the system in a representative environment, using known inputs and expecting given outputs. They exercise the majority of the normal use cases of the system and are designed to provide sufficient confidence in the system (rather than full coverage of every feature and exception).

By establishing a standard suite of functional testing that covers normal use of the system, an ATP can be developed that can be used repeatedly with little or no modification over time. By having a standard ATP, changes to the system (whether they are functional software changes, maintenance updates, or even updates to the supporting environment—such as Operating System or Database patches, new printers or network hardware, etc.) can be easily and quickly verified and assessed for any unanticipated impact to the operation of the HUMS Ground Station and supporting software.

A process that includes execution of the standard suite of ATP tests should be included in any plans for software updates, maintenance patches, or other system environment changes to fielded production systems. The ATP itself should be evaluated at any change in system functionality (new features, etc.) to ensure that the ATP continues to cover the majority of the expected system capability over time.

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APPENDIX L

DATA INTEGRITY

L.1 SCOPE

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

L.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.

RADIO TECHNICAL COMMISSION FOR AERONAUTICS (RTCA)	
RTCA DO-178B.	Software Considerations in Airborne Systems and Equipment Certification.
RTCA DO-200A. .	Standards for Processing Aeronautical Data
RTCA DO-201A.	Standards for Aeronautical Information, 19 April 2000
RTCA DO-278	Guidelines for Communication, Navigation, Surveillance, and Air Traffic Management (CNS/ATM) Systems Software Integrity Assurance, 5 March 2002.
RTCA Report:	Future Flight Data Collection Committee Final Report, Issued 4 December 2001.

(Copies of this document are available from http://www.rtca.org/ or RTCA, Inc., 1150 18th Street, NW Suite 910, Washington, DC 20036, Tel: 202-833-9339, Fax: 202-833-9434 info@rtca.org)

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

L.3 DEFINITIONS

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.

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.

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).

Data Reliability. Data Reliability refers to the provisions taken so the data can be used for its purposes in the CBM system as a result of steps taken to ensure its integrity and availability. Data that the end user receives must be consistent with its origination.

Data Security. Data Security refers to the provisions taken to ensure that the data is protected from corruption by malicious acts, unauthorized access, or accidental mishandling.

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).

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.

L.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, 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 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 required magnitude of the efforts to ensure data integrity 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.[66] 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.

L.5 SPECIFIC GUIDANCE

L.5.1 Criticality. The measures and procedures taken to ensure data integrity in a CBM system should be determined by the resultant severity of the safety effects caused by a compromise in data integrity. The severity of effects of the use of each data parameter should be classified 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.

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

Criticality may be determined by performing a Functional Hazard Assessment (FHA). 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. The FHA may be followed by a preliminary document to the Preliminary System Safety Assessment (PSSA) to further refine hazards and their safety requirements.

For each topic in the following subsections, prevention of corruption and 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 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.

L.5.2 Data Acquisition. Data corruption and 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 captured at a rate that will prevent distortion. Also, any filtering or smoothing should not mask features or characteristics.

In most CBM systems, persistent data will ultimately reside in a 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 (RDBMS) 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. Integrity constraints must follow four basic rules

a. Null Rule: Columns (fields) will disallow INSERTs or UPDATEs to rows (records) containing a NULL (absence of a value) entry. A value can be invalid for several reasons. For example, it might have the wrong data type for the column, or it might be out of range. A value is missing when a new row to be inserted does not contain a value for a non-NULL column that has no explicit DEFAULT clause in its definition.

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.

d. Zero Rules: In strict mode, date entries do not permit '0000-00-00' as a valid date. Dates are disallowed where the year part is nonzero but the month or day part is 0 (for example, '0000-00-00' is legal but '2010-00-01' and '2010-01-00' are not). Check that the month is in the range from 1 to 12 and the day is in the range from 1 to 31. Simple range checking does not apply to TIMESTAMP columns, which always require a valid date. Produce an error in strict mode (otherwise a warning) when a division by zero (or MOD(X,0)) occurs during an INSERT or UPDATE.

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.

L.5.3 <u>Data Computation.</u> Data corruption and 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. Other value errors can include missing fields, out-of-bound entry values, ASCII dashes, hyphens, underscores, and disallowed characters involving backslashes, database functions (SELECT) used as column fields, and timestamp configurations.

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.

System-level computations that can be verified include compression algorithms (gzip), data and chunking algorithms (rsync, etc.).

L.5.4 Data Transmission. Data corruption and 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. Check-sum digital hash signatures can be generated using accepted algorithms such as MD5, CRC, SHA-1, etc. Unrecoverable data lost in the course of transmission may be resolved with protocols such as automatic re-transmission and transmit/receive handshaking. Built-in Windows OS testing utilities for file comparisons include the DOS commands: fc, comp, cksum, diff, shasum, and dir for byte comparisons as common data integrity checks.

In addition to physical ‘line noise’, other possible transmission errors include factors affecting completeness of TCP/IP session errors 404 (string to long), 403 (missing data), and time-outs for large files particularly. Data chunking and compression are one way to reduce time-out errors. Common checks for transmission errors include header checksums, end-of-file (EOF) markers and byte-alignment checks as described in checksum algorithms.

L.5.5 Data Storage. Data corruption and loss may occur during storage; therefore, the design should incorporate the necessary precautions to ensure data integrity during aircraft onboard and off-board storage. Typical corruption scenarios include unsafe partitioning of storage media or use of multiple-version dissimilar software.

Built-in Windows OS testing utilities for file comparisons include the DOS commands: fc, comp, cksum, diff, shasum, and dir for byte comparisons as common data integrity checks.

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. Many different techniques have been developed to optimize the backup procedure. These include optimizations for dealing with open files and live data sources as well as compression, encryption, backup file rotation and de-duplication, among others. An incremental backup copies everything that has changed since the last backup (full, differential or incremental). While magnetic tape has long been the most commonly used backup media, hard disks including RAID configurations, optical storage and geographically distributed remote storage are also flexible options for managing data recovery.

It should be noted that while data backup is one of the most valuable integrity tools, it has limitations related to cost (hardware, software, and labor), performance (particularly for encryption, compression and indexing) and bandwidth-limited network transfers.

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.

L.5.6 Security. In addition to accidental data corruption and 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.

L.5.7 Data Retrieval. Data corruption and 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.

L.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. The data should be oriented and formatted in a manner that allows access to the variety of authorized Army maintenance and analysis systems.

However, as discussed as part of Data Retrieval (L.5.7), measures should be taken to ensure 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.

L5.9 Data traceability Each critical parameter must be traceable from its source to the final usage destination (See Figure L-1).

All data events to include translation transformation, or user manipulation should be traceable. A record that includes the description of the data, the date of the activity, and the identification of the person executing the activity should be logged. This will coincide with the end-to-end process.

Data traceability can provide details or a tracking mechanism from the originator to the end-user. Traceability is critical in identifying whether a component meets a required standard. Being able to trace information on a particular component can be very useful in determining if the data is consistent and accurate.

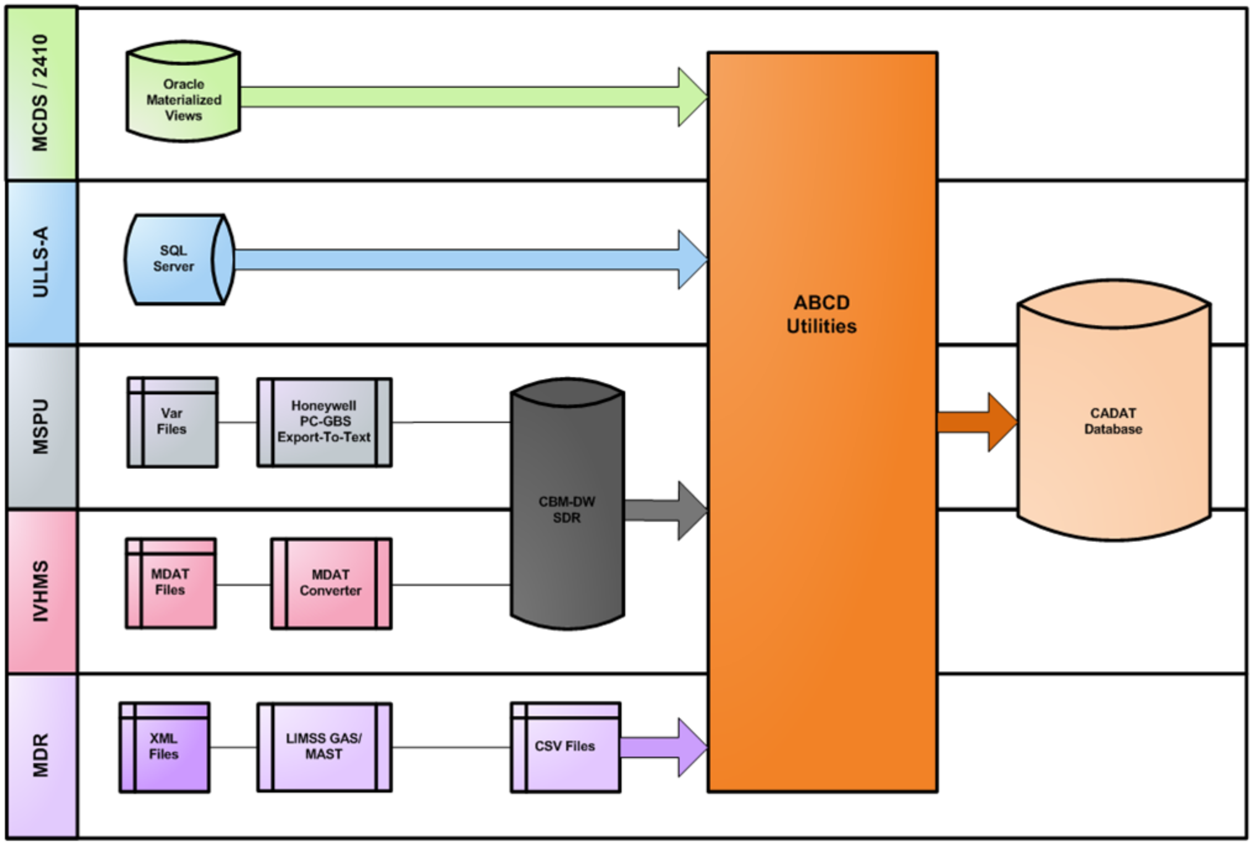

FIGURE L-1. Data orientation and formatting

Traceability shall have readily available the data transmittal information from its origin until it reaches the end-user. To ensure data integrity, traceability logs must be available with information that supports data verification and validation.

L.5.10 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 M

OIL CONDITION AND DEBRIS MONITORING

M.1 PURPOSE AND SCOPE

The purpose of this Appendix is to provide methodology and guidance to implement oil debris and oil condition capabilities for the detection, identification and characterization of faults in oil-wetted aircraft components where oil monitoring is deemed an appropriate risk mitigation strategy. This Appendix covers the use of oil sampling, on-line oil debris sensors and at-line test equipment for oil condition and debris monitoring. Component usage monitoring, limits and trending, diagnostics and prognostic algorithms, and methodology verification and validation are also included. Furthermore it recommends the minimum technical requirements for utilizing oil debris and condition systems for condition based maintenance. Condition based health monitoring on greased and hydraulic components are not specifically addressed in this Appendix but may be added at a later date.

M.2 REFERENCES AND APPLICABLE DOCUMENTS

M.2.1 Standards:

US ARMY AERONAUTICAL DESIGN STANDARD	
ADS-50-PRF	Rotorcraft Propulsion Performance and Qualification Requirements and Guidelines, 1996

(Copies of this document are available at http://www.redstone.army.mil/amrdec/rdmr-se/tdmd/StandardAero.htm)

DEFENSE ACQUISITION UNIVERSITY (DAU)	
AMC-R11-47	Army Oil Analysis Program (AOAP) 19 June 2006

(Copies of this document are available from https://acc.dau.mil/cbm-guidebook or Defense Acquisition University, DAU-GLTC, 9820 Belvoir Road, Ft. Belvoir, VA 22060-5565)

ARMY REGULATIONS	
AR 700-132	Joint Oil Analysis Program 28 Aug 2008 (RAR 31 May 2010)
AR 750-1	Army Materiel Maintenance Policy 10 Apr 2007 (RAR 11 Oct 2007)
Army Pamphlet 750-8	The Army Maintenance Management System (TAMMS) User Manual, 22 Aug 2005 (Rev 14 Sep 2011)

(Copies of these documents are available online at http://www.apd.army.mil/)

ASTM INTERNATIONAL (AMERICAN SOCIETY FOR TESTING AND MATERIALS)	
ASTM D445-11A	Standard Test Method for Kinematic Viscosity of Transparent and Opaque Liquids (and Calculation of Dynamic Viscosity).
ASTM D664-11A	Standard Test Method for Acid Number of Petroleum Products by Potentiometric Titration.
ASTM D974-11	Standard Test Method for Acid and Base Number by Color-Indicator Titration
ASTM D2276	Standard Test Method for Particulate Contaminant in Aviation Fuel by Line Sampling.
ASTM D4057-06	Standard Practice for Manual Sampling of Petroleum and Petroleum Products, 2011.
ASTM D6304-07	Standard Test Method for Determination of Water in Petroleum Products. Lubricating Oils and Additives by Coulometric Karl Fischer Titration.
ASTM D6595-00	Standard Test Method for Determination of Wear Metals and Contaminants in Used Lubricating Oils or Used Hydraulic Fluids by Rotating Disc Electrode Atomic Emission Spectrometry, 2011.
ASTM D7596-10	Standard Test Method for Automatic Particle Counting and Particle Shape Classification of Oils Using a Direct Imaging Integrated Tester
ASTM D7669-11	Standard Guide for Practical Condition Data Trend Analysis
ASTM D7684-11	Standard Guide for Microscopic Characterization of Particles from In Service Lubricants.
ASTM D7685-11	Standard Practice for In-Line, Full Flow, Inductive Sensor for Ferromagnetic and Non-ferromagnetic Wear Debris Determination and Diagnostics for Aero Derivative and Aircraft Gas Turbine Engine Bearings
ASTM D7690-11	Standard Practice for Microscopic Characterization of Particles from In Service Lubricants by Analytical Ferrography.
ASTM D7720-11	Standard Guide for Statistically Evaluating Measure and Alarm Limits when Using Oil Analysis to Monitor Equipment and Oil for Fitness and Contamination
ASTM E122-09	Standard Practice for Calculating Sample Size to Estimate, with Specified Precision, the Average for a Characteristic of a Lot or Process.
ASTM E2412-10	Standard Practice for Condition Monitoring of In-Service Lubricants by Trend Analysis Using Fourier Transform Infrared (FT-IR) Spectrometry.

(Copies of these documents are available online at http://www.astm.org or from the ASTM International, 100 Barr Harbor Drive, P.O. Box C700, West Conshohocken, PA 19428-2959.)

MILITARY STANDARDS (MIL-STDs)	
DOD-PRF-85734A	Performance Specification: Lubricating Oil, Helicopter Transmission System, Synthetic Base (29 Jun 2004) [Superseding DOD-L-85734]
MIL-PRF-23699F	Performance Specification: Lubricating Oil, Aircraft Turbine Engine, Synthetic Base, NATO Code Number O-156 (21 MAY 1997) [Superseding MIL-L-23699E].
MIL-STD-3004C	Department Of Defense Standard Practice: Quality Assurance/Surveillance For Fuels, Lubricants And Related Products (10 Aug 2011)

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

INTERNATIONAL ORGANIZATION FOR STANDARDIZATION (ISO)	
ISO/IEC 17025:2005	General Requirements for the Competence of Testing and Calibration Laboratories

(Copies of this document are available from http://www.iso.org/iso/catalogue_detail?csnumber=21832 or contact International Organization for Standardization ISO Central Secretariat 1, ch. de la Voie-Creuse CP 56 CH-1211 Geneva 20 Switzerland.)

SOCIETY OF AUTOMOTIVE ENGINEERS (SAE) INTERNATIONAL	
SAE AIR 1828	Guide to Engine Lubrication System Monitoring. 27 June 2005.
SAE AIR 1873	Guide to Limited Engine Monitoring Systems for Aircraft Gas Turbine Engines. 5 May 1988. Reaffirmed March 2012.
SAE JA1012 A	Guide to the Reliability-Centered Maintenance (RCM) Standard. 22 Aug 2011

(Copies of this document are available from http://www.sae.org/standards/ or SAE World Headquarters, 400 Commonwealth Drive, Warrendale, PA 15096-0001 USA. Phone (US) 1-877-606-7323)

TECHNICAL MANUALS	
TB 43-0211	AOAP Guide for Leaders and Users 4 Nov 1998 (30 Apr 2010)
TM 38-301-1	Joint Oil Analysis Program Manual, Vol. 1: Introduction, Theory, Benefits, Customer Sampling, Procedures, Programs and Reports, 1 July 2005 (12 Sep 2008)

TM 38-301-2	Joint Oil Analysis Program Manual, Vol. 2: Spectrometric and Physical Test Laboratory Operating Requirements and Procedures, 1 July 2005
TM 38-301-3	Joint Oil Analysis Program Manual, Vol. 3: Laboratory Analytical Methodology and Equipment Criteria (Aeronautical), 1 Aug 2007

(Copies of this document are available online at http://www.armyproperty.com/tm/TB%2043-0211 or 505 E. Huron Street, Suite 202; Ann Arbor, MI 48104 2011 Crystal Drive, Suite 400; Arlington, VA 22202 DUNS Number: 829504880 / CAGE Code: 5BMR7 (703) 269-0013 / (734) 585-5061)

M.2.2 Papers and Books:

REFERENCES	
AOAP/JOAP	"AOAP/JOAP Plans for Hand-held/Portable Oil Analysis Devices", JOAP-TSC-PD-U-05-04, 2005.
Dempsey, Paula J.	"Integrating Oil Debris and Vibration Measurement for Intelligent Machine Health Monitoring", NASA/TM – 2003-211307, March 2003
Dempsey, P. J., G. Kreider, T. Fichter.	Tapered Roller Bearing Damage Detection Using Decision Fusion Analysis, NASA Technical Report: NASA/TM – 2006-214380, July 2006.
Forster, N.H., K.L. Thompson, and T.N. Baldwin.	"Spall Propagation Characteristics of SAE 52100 and AISI M50 Bearings", BINDT-CM, July 2010.
Garvey, R.	"Outstanding Return on Investment When Industrial Plant Lubrication Programs are Support by International Standards", JAI, Vol. 8, No. 6, JAI103526, 2011
Humphrey, G. R.	"Filter Debris Analysis by Energy Dispersive X-Ray Fluorescence Applied to J52P408 Engines*", Denver X-Ray Conference, August 2007, Denver, CO
Lastinger, W., R. Overman, and L. Yates.	"Finding Bearing Failures through Filter Debris Analysis", Reliability Information Analysis Center (RIAC) Journal, Vol. 14, No. 1, 1st qtr, p. 8-12, 2006.
	MetalSCAN Users Manual, C001570, Revision 7
Miller, J.L. and D. Kitaljevich.	*"In-line Oil Debris Monitor for Aircraft Engine Condition Assessment",* IEEE, 0-7803-5846-5, 2000.
Muir, D. and B. Howe.	"In-Line Oil Debris Monitor (ODM) for the Advanced Tactical Fighting Engine", SAE Aerospace Atlantic Congress, 961308, May 1996
Qiu, H., N. Eklund, H. Luo, M. Hirz, . Van Der Merwe, T. Rosenfeld, E. Hindle, F. Gruber.	"Fusion of Vibration and On-line Oil Debris Sensors for Aircraft Engine Bearing Prognosis", 51st AIAA/ASME/ASCE/AJS/ASC Structures, Structural Dynamics and Materials Conference, April

	2010, AIAA 2010-2858.
Toms, A. M.,	*"Detecting Bearing and Gear Failures through At-Line Wear Debris Analysis",* MFPT April 2010, Huntsville, AL
Toms, A. M., E. Jordan, and G.R. Humphrey.	"*The Success of Filter Debris Analysis for J52 Engine Condition Based Maintenance",* Proc. 41st AIAA, Tucson, AZ, July 2005
Toms, A. M., J.R. Powell and J. Dixon.	"The Utilization of FT-IR for Army Oil Condition Monitoring", presented at and published in Proc. JOAP International Condition Monitoring Conference, Humphrey, G. & R. Martin, ed., JOAP-TSC, Pensacola, FL (1998), pp. 170-176.
Toms, L and A. Toms.	"Machinery Oil Analysis - Methods, Automation & Benefits", 3rd Edition, STLE, Park Ridge, IL, 2008, ISBN: 978-0-9817512-0-7.

(Copies of these documents are available from sources as noted.)

M.3 DEFINITIONS

Acid Number (AN): A measure of the acidity of an oil sample expressed as the weight in milligrams of the amount of potassium hydroxide (KOH) required to neutralize one gram of the oil, as prescribed by the ASTM D664 (potentiometric) or ASTM D974 (colorimetric) test methods. Provides an indication of lubricant degradation.

American Society for Testing and Materials International (ASTM): An international standards development organization.

Army Oil Analysis Program (AOAP): A program that implements equipment oil analysis testing as a quality management tool to enhance safety, conserve resources and to extend the life of major assemblies and components. AOAP is mandatory for all Army aircraft and select non-aeronautical equipment.

Atomic Emission Spectroscopy (AES): A test used to determine the relative concentration of wear-metals and some oil additives in lubricating oil by measuring the intensity of the characteristic emission lines of atoms heated to a state of excitation by an electric arc or an induction heater. Provides the elemental content in an oil sample.

Alarm Limits: Set-point thresholds used to evaluate condition or performance data, which when exceeded, indicate a machinery or performance problem.

At-line Tests: Tests performed at the flight line rather than in a laboratory. However the tests are not performed on-aircraft.

Debris Sensor: A device that generates a signal proportional to the size and presence of wear-debris with respect to time. For example, a monitoring device that detects wear debris by fluctuation in a magnetic field.

Ferrography: The identification by optical microscopy of wear and contaminant particles commonly found in used lubricant and hydraulic oil samples that have been deposited on ferrograms.

Filter Debris Analysis (FDA): The removal and analysis of debris from machinery filters. The process extracts, counts and sizes debris then determines metallurgical composition of the debris, generally by x-ray fluorescence (XRF) spectroscopy.

Fourier Transform Infrared (FTIR) Spectrometry: A form of infrared spectrometry in which an interferrogram is obtained; this interferrogram is then subjected to a Fourier transform to obtain an amplitude-wavenumber (or wavelength) spectrum. The absorbance of specific wavelengths of infrared energy determines the organic structure of compounds in lubricating oils.

Inductively Coupled Plasma (ICP) Spectroscopy: An AES that utilizes a high temperature inductively heated carrier gas to excite wear metals and some oil additives in lubricating oil.

Inductive Sensor: A sensor utilizing electromagnetic fields as a medium to permit the detection and measurement of metallic (conducting) particles.

In-line Sensor: A sensor installed with the full-flow of the oil line being monitored passing through the device.

Karl Fischer Titration (KFT): Titration method that uses coulometric or volumetric titration to determine trace amounts of water in an oil sample.

LaserNet Fines (LNF): Laser imaging technique to identify size and shape features of wear debris.

Off-line Testing: Also referred to as Off Aircraft Testing. Tests performed off the aircraft. Testing may occur at the flight line or in a laboratory.

Oil-wetted Component (Oil-washed): Machinery components that are lubricated by oil in a bath or pressurized lubrication system. The component may have its own oil system such as the Apache nose gearbox or the oil system may supply multiple components such as the Chinook transmission.

On-aircraft Testing: Tests performed on the aircraft.

On-line Sensor: A sensing unit fitted on a machine. May be full-flow (in-line) or partial-flow.

Part Per Million (PPM): A unit of measure to describe small values (parts per million, 10^{-6}) of dimensionless quantity.

Polyol Ester Lubricant: A synthetic lubricant base stock formed by the reaction of a fatty acid and a glycol. These lubricants exhibit good oxidation stability and low volatility. Rotorcrafts typically utilize MIL-PRF-23699F for engines and DOD-PRF-85734A for gearboxes.

Rotary Disk Spectroscopy (Rotary Disk Electrode/ Rotrode): An AES that utilizes a rotating carbon disk to introduce the oil sample into an electric arc for excitation.

Sample Interval: The nominal time between successive samples. An optimum (standard) sample interval is derived from failure profile data. It is a fraction of the time between initiation of a critical failure mode and equipment failure. In general, sample intervals should be short enough to provide at least two samples prior to failure. The interval is established for the shortest critical failure mode.

Shear Mixed Layer: Refers to the load-bearing surface layer of a bearing or gear that has been polished to a smooth, ductile, low-wearing surface during component run-in.

Trend Analysis: Monitoring of the level and rate of change over operating time of measured parameters.

Viscosity: The measure of the resistance to flow of a fluid that is being deformed by either shear or tensile stress or the "thickness" of a fluid.

X-Ray Fluorescence (XRF) Spectroscopy: An instrument that utilizes a high-energy x-ray source to excite atoms in a material. The elemental makeup of the material is identified and measured from the spectrum of light emitted.

M.4 INTRODUCTION

Oil condition and debris monitoring is the analysis of a lubricant's properties, suspended contaminants, and wear debris. In-service oil analysis provides information on the lubricant and machine condition. Typical analyses performed include:

a. component wear (break-in, normal, and fault initiation/progression),

b. lubricant condition (base stock degradation and additive depletion), and

c. lubricant contaminants (dirt, water, fuel and incorrect fluid).

Fusion of this information with the physics of failure models, seeded fault testing , and vibration monitoring over the life of a component facilitates remaining useful life prediction. Additionally, new oil testing and periodic lubricant retesting [MIL-STD-3004] ensures a quality product prior to use.

Pioneering development work in machinery oil analysis began in the early 1940's by the Denver, Rio Grande and Western Railway (DRGW) on their new diesel engines. These early programs quickly developed methods for determining the causes for catastrophic failure due to oil related problems. As a result of this and other successes, in-service oil analysis became firmly established as a reliable engine monitoring technique.

The US Army instituted a similar oil analysis program for its aircraft in 1959. This program was followed by the addition of ground combat equipment in 1975 and remaining Army equipment in 1979. In the 21st century, the newest commercial and military aircraft utilize on-line wear debris sensors to provide diagnostic and prognostic capabilities such as the oil debris sensors on the F-35 JSF (F135 engine and STOVL LiftFan) and F-22 Raptor (F119 engine) military aircraft and the commercial geared turbofan PW1000G.

The Army Oil Analysis Program (AOAP) was implemented to enhance safety, conserve resources, and to extend the life of major assemblies and components. AR750-1 states that oil analysis is mandatory for all Army aircraft unless the Deputy Chief of Staff (DCS), G–4 approves the exception per AR 750-1 Material Maintenance Policy. For a list of equipment and components enrolled in the AOAP, the AOAP Web site: https://aoapserver.logsa.army.mil/aircraft_page_1.asp

In the past two decades there have been significant improvements to oil testing equipment. Automation alone has provided improved test repeatability and reproducibility by reducing operator influence. For example, a series of wet chemistry tests have been replaced by a single automated Fourier Transform Infrared (FTIR) spectroscopy test.[67] The industry movement is towards performing oil and debris testing on-line, or at a minimum at-line, with hand-held oil condition testing and filter debris analysis.[68]

Changes to component configuration and operation also impact oil analysis, such as:

a. Advanced lubricant formulations to meet specific problems such as MIL-PRF-23699F HTS for high thermal-oxidative stability and MIL-PRF-23699F C/I for improved corrosion inhibition.

b. Improved design and metal manufacturing to reduce stress induced cracking.

c. Finer filtration to remove wear debris that may cause secondary damage due to over-rolling.

d. Longer duration missions.

e. Operation in sandy conditions.

f. Utilizing remediated/refurbished components.

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[67] Toms, A. M., J.R. Powell and J. Dixon. "The Utilization of FT-IR for Army Oil Condition Monitoring", presented at and published in Proc. JOAP International Condition Monitoring Conference, Humphrey, G. & R. Martin, ed., JOAP-TSC, Pensacola, FL (1998), pp. 170-176.

[68] Garvey, R. "Outstanding Return on Investment When Industrial Plant Lubrication Programs are Support by International Standards", JAI, Vol. 8, No. 6, JAI103526, 2011.

Oil condition and wear debris monitoring should keep pace with these changes in order to provide reliable condition monitoring information. Monitoring the viability and usefulness of testing techniques for determining condition indicators should be an on-going process.

M.5 GENERAL GUIDANCE

The focus of oil condition and debris monitoring is to detect engine or gearbox behaviors indicative of degradation or an incipient fault that leads to component failure or performance changes in the oil-wetted components. Data from oil monitoring is trended over time to determine component performance or fault initiation. See Appendix G, section G.6.2, for further discussion. For accurate oil condition and debris trending, the part's usage should be monitored as appropriate for that platform. See Appendix A for further usage monitoring discussion. Time-on-oil is typically used for oil analysis diagnostics (rate trending).

M.5.1 Lubrication and Debris Monitoring: Lubrication is essential to the operation of many rotorcraft components. Lubrication carries the load and maintains wear surface separation; removes the heat of friction; provides oxidation stability and neutralizes acidic by-products formed; controls corrosion, rust and varnish deposits; and flushes away debris. Maintaining the proper lubricant film thickness through the operating regime of the equipment is essential for long life. This requires keeping the lubricant free of contaminants and degradation, otherwise, metal-on-metal contact may occur resulting in component wear and eventual failure. For instance, the high operating stresses in a gas turbine engine pose challenges to maintaining effective lubrication during extended operation. The operating conditions inside the engine as well as environmental factors (dust, humidity) can introduce contaminants into the oil.

M.5.1.1 Oil Condition and Contamination Monitoring: Oil condition and contamination testing includes monitoring the oil for base stock degradation, additive depletion and contaminants (water, fuel, dirt and incorrect fluids). The degradation and additive depletion may be trended over time providing oil change-out guidance. Contaminants generally have a finite allowable quantity (limit) and once reached, the recommendation is for an oil change.

M.5.1.2 Wear Debris Monitoring: Debris monitoring is performed to determine the presence, size and possible origin of both metallic and nonmetallic debris. Trending provides information on the component wearing and the rate of wear for remaining useful life (RUL) estimates. Wear generation is typically divided into break-in wear, normal rubbing wear and abnormal wear cycles.

a. Break-in wear is the polishing of the load-bearing surface during initial run-in while generating the shear mixed layer. This process removes build debris (micro swarf) left over from the machining process. Break-in wear generates a spike in measurable wear debris until the shear mixed layer is generated and normal rubbing wear begins.

b. Normal rubbing wear generates small metal particles due to the normal exfoliation of the shear mixed layer (wear of surface metals due to the constant working of the surfaces by a sliding or rotating load). Normal rubbing wear occurs throughout the life of the component until a fault is initiated.

c. Abnormal wear is a wear rate beyond the level of normal rubbing wear caused by overload, overspeed, over age, contamination, poor lubrication, etc. Once initiated, this abnormal state continues, eventually leading to component failure.

These wear cycles should be trended over time. Once a fault is initiated, it is critical to monitor not only the amount of debris but also a measure of the rate it is being generated since rate is the most useful in determining RUL estimates.

It should be noted that wear metal concentrations in oil are subject to variability.[69] The rate of wear debris release is not linear with time and for many fault mechanisms, wear occurs in bursts.[70] [71] Wear particle release is event driven. For instance, increased load or speed may result in increased wear events.[72] Filters remove the majority of debris particles greater than the filter pore size. Thus an oil sample only captures new wear and small, suspended, old wear. Wear metal analysis methods have particle size limitations that should be included in evaluations. For example, inductively coupled plasma (ICP) metal analyses are limited to those particles below nominally three microns. It should also be noted that gearbox geometry may trap particles in areas that are not in the oil flow path and thus prevent them from being detected.

M.5.1.3 Oil Filter Monitoring: Oil filters should be monitored to assess the level of filter blockage and prognosticate the need for filter replacement or additional mechanical system maintenance. With the move to finer filtration, Filter Debris Analysis (FDA) should be utilized for wear assessment.[73] FDA has been shown to provide earlier warning on abnormal wear conditions than sampled oil analysis due to the fact the filter captures approximately 95% of all debris, which enters the filter for the filter size rating.[74]

M.5.2 Monitoring Location Options: Oil condition and debris analysis can be performed on-aircraft or off-aircraft as shown in Figure M-1 and Table M-I with the oil obtained or monitored prior to entering a filter element.

M.5.2.1 Off-aircraft: In off-aircraft sampled oil analysis, a small quantity of oil is taken from the component lubrication system and analyzed off-aircraft (*off-line*). Testing may be performed in a laboratory, which is the current Army practice, or at the flight-line (*at-line*).

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[69] Toms, L and A. Toms., "Machinery Oil Analysis - Methods, Automation & Benefits", 3rd Edition, STLE, Park Ridge, IL, 2008, ISBN: 978-0-9817512-0-7.
[70] Muir, D. and Howe B., "In-Line Oil Debris Monitor (ODM) for the Advanced Tactical Fighting Engine", SAE Aerospace Atlantic Congress, 961308, May 1996
[71] Miller, J.L. and D. Kitaljevich. *In-line Oil Debris Monitor for Aircraft Engine Condition Assessment,* IEEE, 0-7803-5846-5, 2000.

[72] Forster, N.H., K.L. Thompson, and T.N. Baldwin. "Spall Propagation Characteristics of SAE 52100 and AISI M50 Bearings", BINDT-CM, July 2010.
[73] Toms, A. M., E. Jordan, and G.R. Humphrey. "*The Success of Filter Debris Analysis for J52 Engine Condition Based Maintenance",* Proc. 41st AIAA, Tucson, AZ, July 2005.
[74] Humphrey, G. R., "Filter Debris Analysis by Energy Dispersive X-Ray Fluorescence Applied to J52P408 Engines*",* Denver X-Ray Conference, August 2007, Denver, CO.

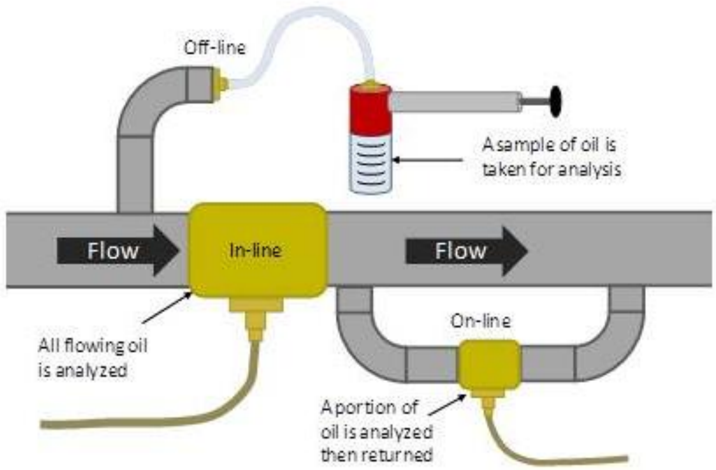

FIGURE M-1. Oil monitoring techniques

Table M-I. Oil monitoring techniques			
Oil Testing Locations			
		Typical Test Sample	**Typically Testing For**
Off-aircraft (off-line)	At-line	Oil or filter sample	Wear debris or fluid condition & contamination
	Laboratory	Oil sample	Wear debris and fluid condition & contamination
On-aircraft	In-line	Entire oil flow (full-flow)	Wear debris
	On-line	Partial oil flow	Fluid condition & contamination

M.5.2.2 On-aircraft: In on-aircraft debris and oil monitoring, the oil passing through the system is analyzed, providing near-real-time results with minimal outside influence. Testing may be *in-line*, which evaluates all the oil debris passing through the system. Wear debris on-aircraft applications are currently in use. A portion of the oil flow in direct connection to the lubrication system may also be monitored oil condition. However, on-aircraft applications for oil condition are not presently available. In-line and on-line applications are not mutually exclusive.

At least one of these methods (off-aircraft or on-aircraft) should be employed for both oil debris monitoring and oil condition monitoring. For near-real-time CBM results, on-line (oil condition) and in-line (wear debris) monitoring would offer the most beneficial solution.

M.5.3 Sampling Rate: The "optimum (standard) sample interval" is derived from failure profile data. It is a fraction of the time between initiation of a critical failure mode and equipment failure. A sampling interval (rate) should be established that is short enough to provide at least two samples prior to failure. This interval is established for the shortest critical failure mode based on Failure Modes Effects and Criticality Analysis (FMECA).[75]

[75] ASTM D7669-11, Standard Guide for Practical Condition Data Trend Analysis.

M.5.4 Limits: Maximum allowable level and trending limits for safe operation should be set for each parameter tested. Level limits provide the maximum value for the various parameters measured, for example, 1000 PPM of water in gearbox oil or 45 PPM of iron (Fe) in a gearbox. Trending limits provide the maximum trend (rate) at which a parameter should increase (or decrease). Limits are statistically derived from the large population data sets that essentially cover all oil-wetted faults. (See Appendix I, Sample Size) A basic statistical process control technique is used for evaluation of the data generating a normal frequency distribution. Level and trend limits should be tracked by component (i.e. sub-assembly level) serial number, not tail number.

M.5.4.1 Level Limits: Historically, AOAP warning and alarm level limits were statistically set at the mean plus 2 and 3 standard deviations, respectively, for oil condition and wear debris.[76] Analysis of historical data provides the limits utilized for sampled oil analysis as shown in Figure M-2. Alert level limits are set at two standard deviations and reportable level limits are set at three standard deviations. At-line instrument limits are statistically set in a similar fashion. On-line sensors tend to have pre-established level limit guidelines.[77]

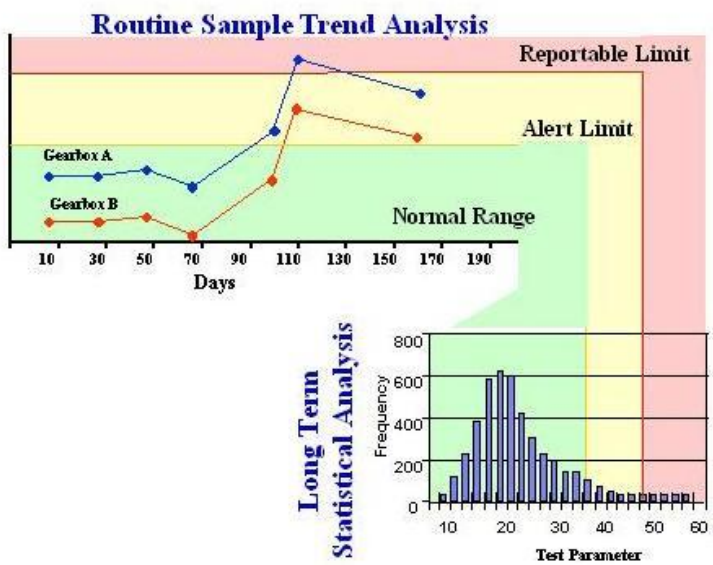

FIGURE M-2: Example standard bell shape curve. Long term statistical analysis supports routine sample analysis

M.5.4.2 Trending Limits: To diagnose and predict machinery and fluid condition, the rate of change should be trended. Level limits only state how much damage has occurred. The *predictive* or *forecasting* nature of condition monitoring is based on rate trending to determine the degree of damage and remaining useful life of the component or fluid. There are numerous

[76] ASTM D7720-11, Standard Guide for Statistically Evaluating Measure and Alarm Limits when Using Oil Analysis to Monitor Equipment and Oil for Fitness and Contamination.
[77] ASTM D7685-11, Standard Practice for In-Line, Full Flow, Inductive Sensor for Ferromagnetic and Non-ferromagnetic Wear Debris Determination and Diagnostics for Aero Derivative and Aircraft Gas Turbine Engine Bearings.

techniques to calculate rate trends from the very simple to the more complex. AOAP utilizes the rise-over-run trend. This rate trend takes the current sample minus the previous sample, divided by the usage metric, times the standard sample interval. For example moderate rate trends are typically set at 60% of alarm level and rapid trend rates are set at 90% of the alarm level. The usage metric and the standard sample interval metric must be the same units of measure, for example, hours. The rise-over-run trend calculation factors in equipment usage (Figure M-3) and is effective for continuous duty and intermittent duty machinery. However, samples should be taken at or near the optimum sample interval.[78] [79]

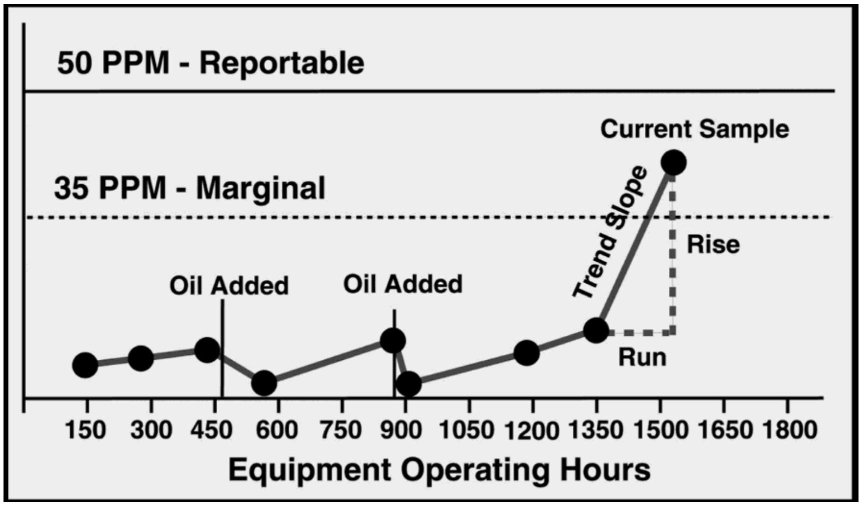

FIGURE M-3 Example trend plot demonstrating rise–over–run: PPM versus operating hours

On-line debris sensors typically use near-real-time cumulative trending (Figure M-4) to show the amount of mass removed and the rate of removal.[80]

M.5.4.3 Maintaining Limits: Level and trend limits should be maintained to ensure they continue to reflect the rotorcraft condition. A regular interval for review should be established, for example every three years. In addition, limits should be reviewed after a significant change in component configuration. For example, installation of finer filtration (less debris) or a new component with different metallurgy (different elements) will impact wear debris detection and analysis.

[78] ASTM D7669-11, Standard Guide for Practical Condition Data Trend Analysis.
[79] Toms, L and A. Toms., "Machinery Oil Analysis - Methods, Automation & Benefits", 3rd Edition, STLE, Park Ridge, IL, 2008, ISBN: 978-0-9817512-0-7.
[80] ASTM D7669-11, Standard Guide for Practical Condition Data Trend Analysis.

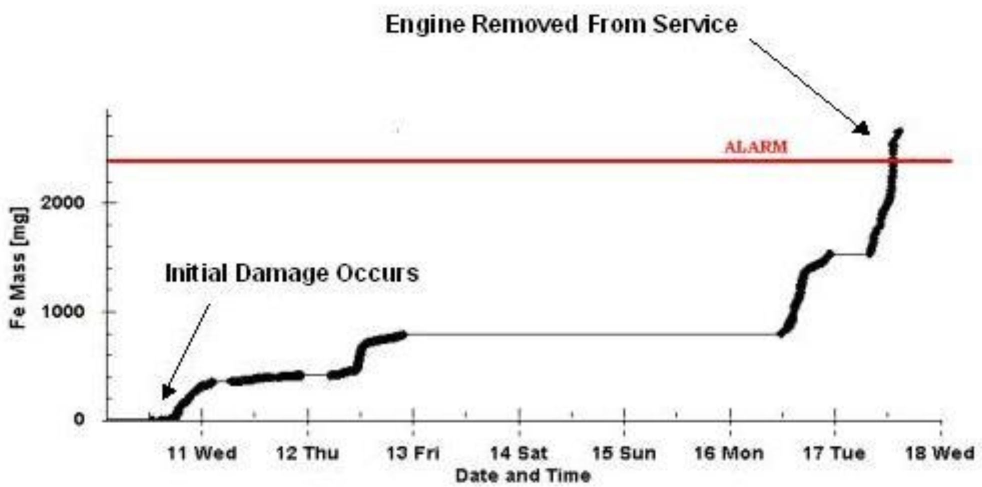

FIGURE M-4: Cumulative trend plot for on-line sensor data: iron (fe) mass versus date

M.6 SPECIFIC GUIDANCE

The implementation of oil condition and debris monitoring is a program approach rather than the application of individual test methods. Failure modes should be analyzed to determine the best method/technique for detecting the failure modes. These techniques should then be periodically assessed to determine the performance of the chosen method. The following subsections provide specific guidance for oil condition and debris monitoring in a condition based maintenance program. While traditionally oil condition and detailed wear debris analyses are ‘off-aircraft’ in legacy systems, state of the practice sensors are available for ‘on-aircraft,’ in-line measurements of oil debris and should be built into newer weapon systems, as previously cited, herein, for the F135, F119, and PW1000G engines.

M.6.1 Testing methodologies: There are a wide variety of testing techniques and technologies utilized for in-service oil testing. They are grouped into the broad categories of off-aircraft (laboratory and at-line) and on-aircraft (on-line and in-line) testing.

M.6.1.1 Off-aircraft Laboratory: AOAP provides in-service oil analysis services to the Army worldwide. Oil samples are taken from the recommended machine components and sent to an Army laboratory. [Army standards in references M.2.1.a-d and t-cc] A multitude of tests are performed as determined by the component failure modes. Examples are provided in the subsequent paragraphs of this section.

M.6.1.1.1 Off-aircraft Testing: The major advantage of off-aircraft oil testing is the wide range of tests available allowing for routine tests to be supplemented with additional, more detailed examinations, as needed. With off-aircraft testing (the current Army practice), substantial historical data and associated limits and trends are available. However, there are inherent problems associated with all off-aircraft sampling procedures such as contamination while sampling, improper sampling procedures (location, tools, consistency), poor sample representation (abnormal wear in particular), analysis time, inconvenience, manpower, and cost.

Additional problems inherent to off-aircraft *laboratory* testing include mixing up samples, transport to laboratory, and long lead time for results which may delay effective maintenance analysis and complicate logistic support of the aircraft. In addition, different instrument models or manufacturers *do not* correlate to one another. Consequently, these different data populations should not be compared to one another. Note also that to maintain quality data, precision laboratory instruments should be standardized daily and require periodic (generally yearly) calibrations.

M.6.1.1.2 Off-aircraft Sampling: Since improper or poor sampling techniques profoundly impact condition test data triggering a false trend alarm, sampling procedures should follow TB 43-0211 AOAP Guide for Leaders and Users and/or ASTM Sampling Practice D4057. Examples of poor sampling with off-line techniques are: stagnant sampling, sampling after component change out, sampling after oil or filter change or both, irregular sample intervals, and sampling without circulating the oil and bringing the equipment to operating temperatures. New oil should also be periodically sampled.

M.6.1.1.3 Off-aircraft Quality Laboratory and Testing Practices: The laboratory tools used to perform the condition monitoring tests influence the data.[81] Variations in analytical instrument configurations impact data reliability. Therefore, limits and trends should only be established based on results from the same make and model of test instrument. For example, trending atomic emission ICP results should be from ICPs with the same sample introduction configuration, same plasma energy, and preferably, the same manufacturer and model. Analytical instruments or test methods with poor measurement repeatability and reproducibility will result in correspondingly poor level and trend limits. Testing repeatability should be included with any limit and trend studies. ISO17025 provides general guidance for quality laboratory operation. If data from multiple laboratories and multiple instruments (same make and model) are to be compared and included in statistical analyses, a correlation (quality assurance) program should be mandatory to ensure all instruments are providing the same quality results. And finally, inappropriate analysis techniques may hide or distort interpretational conclusions. The condition monitoring tool chosen should provide evidence of the critical failure modes under review. To do so, the tools should provide results that are sensitive to the fault, unambiguous to the fault and statistically well behaved. See Appendix D, Minimal Guidance for Determining CIs/HIs for Propulsion Systems.

M.6.1.1.4: Off-aircraft Laboratory Rotorcraft Tests: The common laboratory tests for rotorcraft are: AES (Rotrode), FTIR, AN, viscosity and water. LNF and ferrography may also be used for cleanliness ratings and more detailed examinations of particulates. The following provides a short description on the various Army laboratory tests and some of the inherent advantages and disadvantages:

AES by rotating disk detects normal rubbing wear, some additives and abnormal wear.[82] It is easy to use; requires no sample preparation and is relatively easy to deploy. However

[81] Toms, L and A. Toms., "Machinery Oil Analysis - Methods, Automation & Benefits", 3rd Edition, STLE, Park Ridge, IL, 2008, ISBN: 978-0-9817512-0-7.
[82] ASTM D6595-00, Standard Test Method for Determination of Wear Metals and Contaminants in Used Lubricating Oils or Used Hydraulic Fluids by Rotating Disc Electrode Atomic Emission Spectrometry, 2011.

particles larger than 10 microns are not detected (not atomized) and the daily standardization with reference materials is often time consuming. A modification is available to the AES called rotrode filter spectroscopy (RFS), which increases the size range for particle detection up to at least 25 microns. This technique adds an additional step and time to the testing process; and requires different atomic emission limits to be established.

FTIR (ASTM E2412) detects fluid condition (degradation, loss of additives) and organic fluid contamination (water, fuel, incorrect fluid). It is easy to use; requires no sample preparation; incorporates a built-in reference standard; and is relatively easy to deploy. FTIR requires knowledge of the lubricant class (e.g., gas turbine versus diesel) and the use of a non-flammable solvent.

AN determines the change in relative acidity of a lubricant. An increase in acidity is important because it is generally caused by degradation of the lubricant. Note, AN requires the use of hazardous chemicals and generates a separate waste stream.[83]

Water by KFT determines the water content in oil detecting low, for example, 50 parts per million (PPM), concentrations of water.[84] The major disadvantage with KFT is it requires the use of hazardous chemicals and generates a separate waste stream. In addition, oxidation by-products are titrated as water, even with an oven attachment. FTIR, mentioned previously, is an easier technique to determine water contamination in polyol ester oils (MIL-PRF-23699F and DOD-PRF-85734A) and clearly differentiates water from degradation by-products.

Viscosity determines the “thickness” of the oil.[85] A change in viscosity indicates a change in oil chemistry. For instance, a decrease in viscosity may indicate contamination with a lower viscosity fluid such as fuel. However, a change in viscosity does not indicate what problem caused the change.

Ferrography determines the appearance, type, size and number of wear particles by microscopic examination.[86] This ability is beneficial since the morphology of larger particles may provide some indication of wear mode and metal source. In addition, ferrous and non-ferrous particles are differentiated. Ferrography is often very time consuming, it requires an expert microscopist, and does not determine the elemental or alloy composition of the particles.

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[83] ASTM D664-11A, Standard Test Method for Acid Number of Petroleum Products by Potentiometric Titration. Or ASTM D974-11, Standard Test Method for Acid and Base Number by Color-Indicator Titration.

[84] ASTM D6304-07, Standard Test Method for Determination of Water in Petroleum Products. Lubricating Oils and Additives by Coulometric Karl Fischer Titration.

[85] ASTM D445-11A, Standard Test Method for Kinematic Viscosity of Transparent and Opaque Liquids (and Calculation of Dynamic Viscosity).

[86] ASTM D7684-11, Standard Guide for Microscopic Characterization of Particles from In Service Lubricants. And ASTM D7690-11, Standard Practice for Microscopic Characterization of Particles from In Service Lubricants by Analytical Ferrography.

LNF determines particle size, count and aspect ratio to facilitate analysis of lubricant cleanliness and particle morphology of larger particles.[87] LNF is automated, relatively easy to use, and can determine the ISO particle count (cleanliness). However it is limited to particles less than 100-microns and particles are seen in silhouette, rather than color as in Ferrography.

The above tests represent the most common off-aircraft laboratory tests employed by the Army for rotorcraft. These test techniques are well documented and all have ASTM standards associated with them. The major advantage of laboratory testing is the wide variety of tests available.

M.6.1.2 Off-aircraft At-line: In the past decade, demand for more immediate machine condition information has resulted in more at-line testing instruments.[88] These instruments tend to be more rugged and often are easier to use than laboratory equipment. However, the instruments are generally designed for only one aspect of oil analysis (wear or fluid condition) and may require different types of samples than the laboratory oil sample, which is utilized for all the tests and instruments mentioned in Section 6.1.1.4. For instance, in filter debris analysis, a filter is the "sample", rather than the oil. Example at-line instruments are provided in the subsequent paragraphs of this section.

M.6.1.2.1 Off-aircraft At-line Testing: The major advantages of at-line testing are that the results are immediately accessible to the soldier and some of the logistical problems with laboratory testing are negated such as mixed up samples, transport to laboratory, and long lead time for results. Additionally, the instruments are typically easy to use. The disadvantage is the instruments do not necessarily correlate to laboratory instruments and some generate only a simple go/no go result. These tests still require a sample to be taken so the inherent problems associated with all off-line sampling procedures still exist. The Army currently does not routinely utilize any of these at-line oil analysis instruments. (See section 6.1.1.1).

M.6.1.2.2 Off-aircraft At-line Tests: Several at-line oil condition monitoring test available include:

Hand-held FTIR detects fluid condition (degradation, loss of additives) and organic fluid contamination (water, fuel, incorrect fluid). The device is easy to use and deploy; requires only a few drops of oil; does not require sample preparation or solvent; has a built-in reference standard; and provides results similar to laboratory FTIR instruments. As with a laboratory FTIR, knowledge of the lubricant class (e.g., gas turbine versus diesel) is required.

Filter debris analysis (FDA) counts and sizes particles by ferrous and non-ferrous, prepares a patch of debris for x-ray fluorescence (XRF) analysis, and performs XRF analysis to provide elemental composition of the debris. The filter back flushing and instrument standardization are often automated in an instrument and all particles sizes (1-1000+ microns) are analyzed by the XRF. FDA provides early fault warnings since filters contain all the wear

[87] ASTM D7596-10, Standard Test Method for Automatic Particle Counting and Particle Shape Classification of Oils Using a Direct Imaging Integrated Tester.

[88] Garvey, R. "Outstanding Return on Investment When Industrial Plant Lubrication Programs are Supported by International Standards", JAI, Vol. 8, No. 6, JAI103526, 2011.

debris (ferrous and non-ferrous) for their capture efficiency, and filters do not need to be analyzed as often as oil samples.[89] However, this test requires removing a filter. An instrument that performs these functions is utilized by the US Navy, Allied Forces and gas turbine OEMs.

Individual particle analysis determines the specific alloy for every particle (e.g., M50) rather than just elements (e.g., iron, molybdenum, chromium and vanadium). With this method, particle detection and location is performed by camera or scanning electron microscope (SEM) and alloy identification of each particle is performed by XRF. Particles may be obtained from a variety of sources such as magnetic chip detectors, filters, filter bowls, oil samples, etc. However, particles must be greater than 80 microns (abnormal wear size). For the SEM instrument, there is a significant cost and expertise to maintain the instrument and, although deployable, these instruments typically weigh 900+ lbs. To reduce the impact of this disadvantage to the military, lightweight versions utilizing the camera/XRF arrangement are being explored.[90]

The above tests represent the most applicable off-aircraft, at-line tests available for rotorcraft. The major advantage of at-line testing is the ability to provide immediate machine condition information.

M.6.1.3 On-aircraft: On-aircraft, oil debris and oil condition monitoring provide the advantage of continuous real-time or near-real-time monitoring of wear and lubrication problems. On-aircraft oil sensors have minimal impact on system flow, provide direct results, and have little interference from outside influence. The disadvantages to on-aircraft sensors are the sensors must be fitted to the equipment and often are limited to one test per sensor e.g., wear debris. For fluid condition and contamination, representative results are obtained by monitoring only a portion of the oil stream since it is the condition of the oil as a whole that is of interest. However, wear debris analysis may be misrepresented in partial flow scenarios due to flow dynamics of the wear particles. Consequently, wear debris should be measured by an in-line, full-flow sensor. Installation of an in-line monitoring sensor should not adversely influence the performance of a lubrication system: this can be a challenge in legacy rotorcraft. For new weapons systems, the sensor(s) should be built into the design. Debris monitors should be placed before any lubrication pump(s) or filter(s). Examples are provided in the subsequent paragraphs of this section.

M.6.1.3.1 On-aircraft Component Wear Monitoring: Several types of on-line and in-line sensors are available that detect metallic debris in the oil stream or portions of the oil stream and serve as indicators of component wear. Some sensors offer the earliest warning of bearing failures (magnetic coil) while others only provide last minute warning (chip detectors):

Electric Chip Detectors create a magnetic field that attracts ferromagnetic debris particles. The debris bridges a gap between two electrodes, which act as a switch closure for an

[89] Humphrey, G. R., "*Filter Debris Analysis by Energy Dispersive X-Ray Fluorescence Applied to J52P408 Engines",* Denver X-Ray Conference, August 2007, Denver, CO.

[90] Toms, A. M., *Detecting Bearing and Gear Failures through At-Line Wear Debris Analysis",* MFPT April 2010, Huntsville, AL.

alarm output. The device should be threaded directly into a lubrication system. These devices typically see between 30-60% of the oil flow, depending on design and location. Debris particles captured by the system should be removed for further inspection to determine the material type and possible wear mechanism. The switch nature of the Electric Chip Detector does not allow for debris trending and can cause excessive false positives by detecting insignificant debris build-up if not equipped with a metal fuzz burn-off mechanism. The Pulsed Electric Chip detector attempts to alleviate this problem. A low energy current pulse clears fine debris away to reduce false positives. The number of pulses the unit outputs should be programmed into its memory. The pulses clear away fine debris, but the larger debris particles are still held onto. As with the conventional unit, the debris should be retained for further inspection. According to ADS-50, electronic chip debris monitors should be utilized on all oil-lubricated systems. Chip detectors are in widespread use on aircraft.

Mesh Detector (screen) may also be used to provide a warning of ferrous and nonferrous conducting debris particles. Debris particles bridge the gap between strands to close a circuit. Oil flow passes through the screen with minimal pressure drop. These screens are only good at detecting conductive particles (but the particle does not have to be ferrous, e.g. aluminum) and the particle size must be comparable to the screen hole size in order to complete the circuit.

Induction Coil Sensors use a magnetic coil assembly to detect and categorize metallic particles by size and type (ferrous or non-ferrous). The minimum detectable particle size is determined by the bore size of the sensor. Currently, on-aircraft systems are designed to detect particles from 120 to 220 microns and up, depending on the diameter of the oil line. The sensor consists of three coils surrounding the inside bore. Two coils create a magnetic field, and the third coil detects any disturbances in the field. Depending on the type and magnitude of the disturbance, the control unit determines the type of particle and the particle size. This type of sensor should be installed directly in the lube oil line and can be configured as an “in-line” or “on-line” configuration although SAE AIR 1828 recommends the full-flow, in-line configuration. Distribution of the particle sizes along with particle frequencies (rate) should be monitored and trended. The control unit also reports the total mass of ferrous material that has passed through the sensor. This allows users to track the debris progression over time and to trend this information to determine the current state of the fluid wetted components. Installation of an in-line monitoring sensor should not adversely influence the performance of a lubrication system: this can be a challenge in legacy rotorcraft where a redesign of the lubrication lines may be required. Induction coil, full-flow sensors are available and should be built into newer weapons systems. They are currently utilized on the F35 JSF (F135 engine and STOVL LiftFan), F22 Raptor (F119 engine) and a wide variety of industrial gearbox and gas turbine applications.

M.6.1.3.2 <u>On-aircraft Fluid Condition and Fluid Contamination Monitoring</u>: Several approaches are in development and testing for on-line oil quality and contamination sensors, although none are in widespread use and none are available for on-aircraft applications at this time. Examples of these fluid sensors are dielectric (capacitance), conductivity (resistance), impedance and infrared, which are discussed below:

Sensors for Monitoring Electrical Properties of Oil: Several techniques are utilized to discern oil condition by monitoring the electrical properties of oil. Dielectric (capacitance) oil quality sensors work on the principle that the dielectric constant of oil will change as it degrades or becomes contaminated. Dielectric constant is a measurement of a substance's ability to resist the formation of an electric field within it. As the quality of oil deteriorates, the sensor measures its dielectric constant and outputs this information in the form of a voltage that is correlated to the quality of the oil. Dielectric sensors have been reported to be adept at detecting the presence of water in lubricating oils due to the large difference in dielectric constant between water and oil. The dielectric constant is highly dependent on temperature, which means that lubricant temperature must also be measured and the dielectric signal compensated to make up for the temperature changes. The sensor should provide real-time qualitative analysis of oil condition and some quantitative analysis of water content to the maintainer. Some oil quality sensors also contain built-in processors that monitor the dielectric constant and provide alerts when there are substantial changes. The conductivity (resistance) sensors measure the capacity of the oil to conduct electricity. These sensors are reported to monitor oil acidity and oil breakdown. Total electrical impedance (TEI) is the sum of resistance and capacitance. Sensors utilizing TEI are believed to provide greater discrimination of the individual oil degradation and contamination modes than either resistance or capacitance alone. Conductivity and TEI are also highly sensitive to temperature. There has not been any on-aircraft testing.

Infrared Sensors: Infrared sensors use a miniaturized infrared spectrometer to track pre-established infrared wavelength regions providing results similar to laboratory and at-line infrared instruments. These sensors are able to provide real-time data on oil condition (degradation and additive loss) and contaminants (water, fuel and incorrect oil) to the maintainer. Some applications of these sensors require extensive calibration with the desired oil characteristics. On-line applications have been tested on marine gas turbine generators but there has not been any on-aircraft testing.

In summary, on-aircraft sensors provide the ability for near-real-time debris and oil monitoring. While *oil condition* sensors are not available at this time for on-aircraft applications, *wear debris* sensors are available and in use for military and commercial applications.

M.6.1.4 Wear Debris Size and a Comparison of Methods used for Detection: A comparison of wear debris size during normal and abnormal wear and a comparison of wear debris detection techniques are provided in Figure M-5 and Table M-II. Figure M-5 presents a widely used diagram to describe the progress of metallic wear debris release from normal to catastrophic failure.[91] It must be pointed out that this figure summarizes metallic wear debris observations from all the different wear modes that can range from polishing, rubbing, abrasion, adhesion, grinding, scoring, pitting, spalling, etc.

[91] Toms, L and A. Toms., "Machinery Oil Analysis - Methods, Automation & Benefits", 3rd Edition, STLE, Park Ridge, IL, 2008, ISBN: 978-0-9817512-0-7.

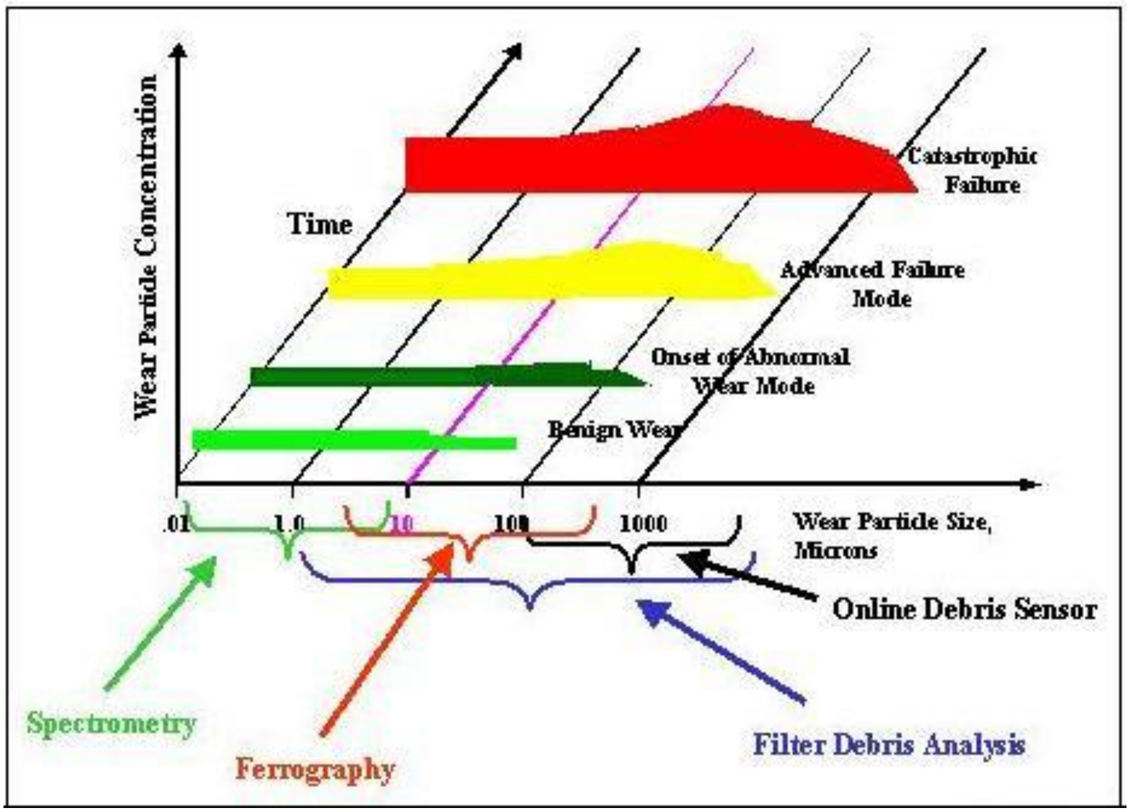

FIGURE M-5: **Wear modes, particle size ranges and detection methods**

M.6.2 Implementation of Oil Analysis: The following sections will discuss the implementation of various oil analysis processes

M.6.2.1 Off-aircraft Laboratory Data: The implementation of laboratory general sampling procedures, sampling intervals and general laboratory tests performed are documented in TB 43-0211 (sampling) and DA Form 5991-E and DA Form 2026 as described in DA PAM 750 accompany the sample to the laboratory. Procedures and tests are outlined in TM 38-301-1 and 2. Equipment limits and trends are provided in TM 38-301-3. Army results are uploaded to the Logistics Information Warehouse (LIW) and can be retrieved from it. Abnormal results are annotated on DA Form 3254R and 2407 or SAMS-E DA Form 5990E. However, processes for implementing sampling on new components, addition/subtraction of laboratory tests, and establishing/changing limits are not well documented. As mentioned in Section M.5.4.3, level and trend limits should be maintained on a regular basis to ensure that they are still applicable.

M.6.2.2 Off-aircraft At-Line: Any implementation of at-line testing should interoperate with the ground-based CBM information systems such as the ground data station and provide actionable maintenance recommendations to the operator of the device as approved by the authority for continued airworthiness.

M.6.2.3 On-aircraft: Implementation and communication of on-aircraft oil debris sensor varies depending on the type of sensor. All should be placed before any lubrication pump(s) or filters(s) and provide cockpit indications of abnormal debris generation, rates or sizes.[92]

Electric Chip Detectors should be removable without draining the lubricant and for lubrication systems with remote components and/or accessories; they should isolate the component or accessory generating the debris. For pressurized oil systems, chip detectors should be located such that all lubricant passes through the debris monitor. For non-pressurized lubrication systems, the debris monitor should be located such that wear debris from any internal components is likely to migrate quickly to the chip detector

Mesh Detectors (Screens) are removable and cleanable and should be downstream of the debris sensor. A remote method of ensuring sensor circuit continuity should be provided.

Induction Coil Sensors should be in line and include the capability to automatically interface with on-board data collectors, such as HUMS, for use by the soldier and engineering.

The decision as to whether to implement oil analysis and, if so, which approach or combination of approaches should be implemented should be determined through the TCM process, should be determined by fault modes, the best operational approach, cost effectiveness, and the least burden to the soldier. A tiered defense may be the most effective, utilizing all oil analysis approaches, on-aircraft for faults with rapid failures (bearing and gears), off-aircraft, at-line for quick fluid and debris checks and off-aircraft laboratory for long-term fluid condition. As new at-line and on-line techniques are implemented, historical methods should be re-assessed to determine whether there is a continued need or a more cost effective replacement technology. For instance, if on-line, near-real-time oil debris sensors are implemented and the bearing is composed of only one alloy (e.g., M50 with silver plating), it may or may not be necessary to continue to analyze an oil sample to determine that the elements detected in the oil are iron, chromium, molybdenum and vanadium (the composition of M50) and silver. Also, if fidelity of newer oil condition/debris monitoring technology far surpasses that of older applications, a Reliability Centered Maintenance (RCM) analysis may support replacement of past oil analysis methods.

M.6.3 Data Management: Data should be carefully managed to ensure integrity throughout the lifecycle from collection to destruction. Data from debris and oil condition monitoring tools should be tracked by component serial number, not tail number. Actionable information obtained from the data should be immediately and readily accessible to the soldier and engineering. Data from oil analysis should be integrated with other systems for a more comprehensive condition assessment. Any design should consider long term archival of captured data. This broader approach to retaining data will allow for later data mining to uncover long-term trends in reliability, availability and performance, which should be used for future improvements to CBM algorithms. See Appendix L for detailed information on data management.

[92] ADS-50-PRF, Rotorcraft Propulsion Performance and Qualification Requirements and Guidelines, 1996.

TABLE M-II: Wear debris detection methods

Instrument	Size Range	Sample Media	Advantages	Disadvantages	Military Users
AES-Rotrode	0.1 to 10	Oil sample	Provides elements, fast, easy, deployable	Off-aircraft (lab)	All
AES RFS	0.1 to 25*	Oil sample	Provides elements, analyzes slightly larger particles than Rotrode	Off-aircraft (lab), variable data	A few labs in test
AES-ICP	0.1 to 3	Oil sample	Provides elements, fastest, automated	Off-aircraft (lab), requires gas (Argon)	(Commercial labs)
XRF-SEM	80 to 1000+	Chips – from any source	Provides counts, size and alloy	Off-aircraft (at-line), expensive, most difficult to operate	USAF, some Allied
XRF-chips	80 to 1000+	Chips – from any source	Easy, provides counts, size and alloy	Off-aircraft (at-line)	In test USAF
XRF-filter	All	Filter	Provides particle count, size, elements & interprets alloy composition	Off-aircraft (at-line)	US Navy, Allied Forces
Magnetic coil sensors	120 to 1000+	N/A	On-aircraft, provides counts, size, mass, ferrous and non-ferrous	Retrofit or design on-aircraft	USAF & Navy (& commercial)
Magnetic chip detectors	Ferrous**	N/A	On-aircraft	Retrofit or design on-aircraft, only sees partial oil flow, ferrous only	All

*Size not provided in literature
**Enough particles to bridge gap

M.6.3.1 Data Validity: The validity of condition data is dependent on multiple factors. One of the primary factors is proper tracking of the data to the component serial number. Other factors include maintenance actions and operational and environmental conditions, which should be monitored and documented. The data validity of the condition monitoring tools should also be monitored. A few examples are provided:

M.6.3.1.1 Off-aircraft: For laboratory and at-line instruments, standardizing daily ensures the validity of data. These standardization checks may be internal to the instrument or may utilize actual samples similar to the oils being tested. If the daily standardization fails, the instrument should not be used until the cause of failure is determined, remedied and standardization passes. See Section M.6.1.1.3.

M.6.3.1.2 On-aircraft: For on-line and in-line sensors, the data acquisition system should have a built-in test capability to check the validity of the incoming sensor data.[93] The system should notify maintenance personnel if the sensor is suspected of being faulty. The integrity of a monitoring system should be ensured by having the ability to detect and flag faulty sensor data. When faulty data are detected, a process should be in place to account for the lost information. Linear interpolation between good values is typically used.[94]

M.6.3.2. AOAP: AOAP data is automatically uploaded to Oil Analysis Standard Interservice System (OASIS) for most instruments and OASIS is automatically updated to LIW for easy access by all units.

M.6.3.2.1Chip Detectors and Mesh Screens: There is generally a light on the maintenance panel indicating when these devices detect debris. Any visual debris observed during phase inspections is manually annotated in a logbook.

The introduction of any new system should be at least as good as the current process. At-line instruments should aim to have their data uploaded to the ground station and on-line sensors should aim to have their data automatically uploaded to the on-board portion of the HUMS.

M.6.3.3 Data Development: Statistical process control (SPC) and cumulative distribution techniques are generally used to statistically evaluate alarm limit values for oil analysis. The data set utilized should represent a normal distribution where the majority of samples are expected to fall within two standard deviations of the mean or represent about 94% of all samples taken. Note some oil measurements are bound by zero and the distribution will not appear normal. While a mean and standard deviation can still be calculated, the user should verify that alarm limits based on these statistics are descriptive of the actual distribution. For

[93] ASTM D7685-11, Standard Practice for In-Line, Full Flow, Inductive Sensor for Ferromagnetic and Non-ferromagnetic Wear Debris Determination and Diagnostics for Aero Derivative and Aircraft Gas Turbine Engine Bearings.

[94] ASTM D7669-11, Standard Guide for Practical Condition Data Trend Analysis.

example, only about 5% of the values should fall above the mean plus two standard deviations.[95] Appendix I discusses statistical processes, albeit for vibratory CBM algorithms.

For off-aircraft limits, tentative alarm limits may be set with as few as 30 samples although the quality of the limits improve with larger populations. For laboratory data, 100’s to 1000’s of samples in a data set are generally available for like components.

M.6.3.4 Exceedance Recording: Any exceedances should be recorded when each event occurs and flagged to the ground crew.

M.6.4 Life Usage Monitoring: Section G.7.3 in Appendix G encompasses life usage monitoring and applies to oil analysis utilizing laboratory, at-line testing, on-line sensors and manual observations.

M.6.5 Anomalies and Faults: Fault detection methods discussed in Appendix G.6.2 apply to oil condition monitoring. Oil debris and condition data are analyzed to determine normal versus anomalous behavior. Condition indicators (CI) and health indicators (HI) should be developed to allow for sufficient time to schedule maintenance and prevent catastrophic failure. There are various diagnostic levels:

a. In electric chip detectors, a chip light is indicated when sufficient debris has collected to bridge a gap between two electrodes.

b. AOAP limits, based on statistical analysis of large databases, indicate an abnormal increase in wear debris.

c. Oil debris sensor (magnetic coil) limit algorithms are based on bearing faults from test rigs and component teardowns.

An example algorithm is provided for rolling element bearings using an oil debris magnetic coil sensor. Note algorithms are also available for gears. A spall is essentially a rectangular area of damage with some average thickness for the missing material where the width of the spall is proportional to bearing roller width and the length of the spall is a function of the bearing mean diameter and the angle of spall. In the case of a cylindrical rolling element, the rolling element width is the width of the roller. In the case of a spherical rolling element, the rolling element width is the diameter of the roller. Thus, formulas may be derived to estimate bearing damage severity in terms of accumulated metallic wear debris counts or mass as functions of bearing geometry that include bearing pitch diameter, rolling element width, and number of rolling elements.[96]

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[95] ASTM E2412-10, Standard Practice for Condition Monitoring of In-Service Lubricants by Trend Analysis Using Fourier Transform Infrared (FT-IR) Spectrometry.

[96] ASTM D7685-11, Standard Practice for In-Line, Full Flow, Inductive Sensor for Ferromagnetic and Non-ferromagnetic Wear Debris Determination and Diagnostics for Aero Derivative and Aircraft Gas Turbine Engine Bearings.

The alarm limit is a damage severity level where it is recommended that the machine be shut down for inspection and servicing because continued operation may result in secondary damage to the machine. In order to quantify bearing degradation severity in terms of a suitable alarm limit, it is necessary to represent severity in terms of an equivalent angle of spall. An angle of spall of concern is considered to be the point where the supported shaft begins to experience some loss of position when two rolling elements have begun to simultaneously roll over the spalled area. This is equivalent to a spall angle of approximately 360 degrees divided by the number of rolling elements as shown in Figure M-6. This criterion for setting the alarm limit has been found to be a conservative limit. The formulas account for bearing size in calculation of the limits. Formulas define alarm limits for rolling element bearings in terms of accumulated mass or accumulated counts of metallic wear debris. The formulas correspond to a bearing spall wear scar size equivalent to the length between two rolling elements (Figure M-6).

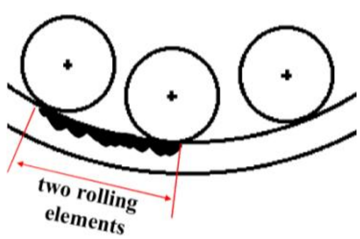

FIGURE M-6 Bearing wear scar equal to two rolling elements spall length

$$M_{ALARM} = Km\ (360/N)\ D\ w \qquad \textbf{(1)}$$

where:

M = Mass detected by sensor (mg)

Km = Calibration constant relating sensor detected debris mass for a specific bore size sensor to bearing spall geometry characteristics (mg/deg mm^2)

N = Number of rolling elements

D = Bearing pitch diameter (mm)

w = Rolling element width (mm)

$$C_{ALARM} = Kc\ (360/N)\ D\ w \qquad \textbf{(2)}$$

where:

C = Counts detected by sensor (counts)

Kc = Calibration constant relating sensor detected debris counts for a specific bore size sensor to bearing spall geometry characteristics (counts/deg mm^2)

N = Number of rolling elements

D = Bearing pitch diameter (mm)

w = Rolling element width (mm)

Warning limits should be set to alert an operator that a problem (e.g., bearing spall) has developed on the machine being monitored and that sensor parameters should now be monitored more closely as damage progresses towards the alarm limit. Its primary purpose is as an indicator of early damage to give an organization sufficient lead time to consider planning a scheduled maintenance at some future date. For instance, a warning limit set at a level that is 10% of the alarm limit is an indication that the bearing damage is in the early stages of damage progression.[97][98] If monitoring several oil wetted components and only one sensor is in the oil stream, the most conservative limit is generally chosen. Appendix D, Minimal Guidance for Determining CIs/HIs for Propulsion Systems, provides guidance on prioritizing fault modes.

$$M_{WARNING} = 0.10\ M_{ALARM} \quad \textit{(for debris mass)} \qquad (3)$$

$$C_{WARNING} = 0.10\ C_{ALARM} \quad \textit{(for debris counts)} \qquad (4)$$

M.6.6 Recommended Minimum Technical Requirements: To be considered a condition based maintenance tool, the ability to track performance metrics should be available. Any new monitoring tool introduced should at least provide the same fault detection or greater detection than the existing system. All existing monitoring systems and any potentially new systems should strive to meet the SAE standard for a Condition Monitoring tool. In other words, detect 90% of faults in progress with no more than 10% false positive indications based on subsequent tear down analysis. As an example, the US Navy states that FDA meets all the criteria established by the SAE standard for a condition monitoring task.[99]

M.6.7 Integration of Diagnostic Tools: Fault detection capabilities vary with the condition monitoring tool and the specific fault mechanism. Some tools detect a specific fault earlier than another tool, while for other faults the earliest detection is with the second tool. Therefore, it is desirable to combine the strengths of each method to improve detection accuracy and robustness [Qui]. Fusing vibration and on-line oil debris sensing/oil debris monitoring (ODM) data has been demonstrated to provide more reliable indications of machine condition. The data from these two trending methods can augment, verify, and validate each other.

Multisensor data fusion is a process to integrate oil debris and vibration based bearing damage detection techniques. Information fusion is defined as "the theory, techniques and tools conceived and employed for exploiting the synergy in the information acquired from multiple sources such that the resulting decision or action is in some sense better than that would be possible if any of these sources were used individually". [100] Data fusion methodology is the logical choice for integrating vibration and oil based measurement technologies for intelligent machine health monitoring.

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[97] ASTM D7685-11, Standard Practice for In-Line, Full Flow, Inductive Sensor for Ferromagnetic and Non-ferromagnetic Wear Debris Determination and Diagnostics for Aero Derivative and Aircraft Gas Turbine Engine Bearings.

[98]SAE JA1012. A Guide to the Reliability-Centered Maintenance (RCM) Standard. 22 Aug 2011

[99] Lastinger, W., R. Overman, and L. Yates, *"Finding Bearing Failures through Filter Debris Analysis*", Reliability Information Analysis Center (RIAC) Journal, Vol. 14, No. 1, 1st qtr, p. 8-12, 2006.

[100] Dempsey, Paula J. "Integrating Oil Debris and Vibration Measurement for Intelligent Machine Health Monitoring", NASA/TM – 2003-211307, March 2003.

There are several benefits of using sensor fusion instead of single sensor limits, including: a) more robust performance; b) extended spatial/temporal coverage since one sensor may contribute information while others are unavailable or lack coverage of the event; and, c) increased confidence because more than one sensor may confirm the same event which increases assurance of its detection.

Sensor data may be fused at the raw data level, feature level, or decision level. Direct fusion of raw sensor data requires sensors of the same/similar type, with similar output formats and sampling rates. Feature level fusion requires the raw data be first processed into features, and then these features are fused into a single combined parameter. Observing changes in the signature of this parameter then identifies faults. Feature level fusion is best applied to the same types of measurement technologies. Decision level fusion processes each sensor to achieve decisions, and then combines the decisions. For the example in Figure M-7, decision level fusion was chosen because this does not limit the fusion process to a specific feature. New features can be added to the system or different features can be used without changing the entire analysis. This allows the most flexibility when applying this process to condition based systems since, in most cases; different sensors and post-processing methods are used.

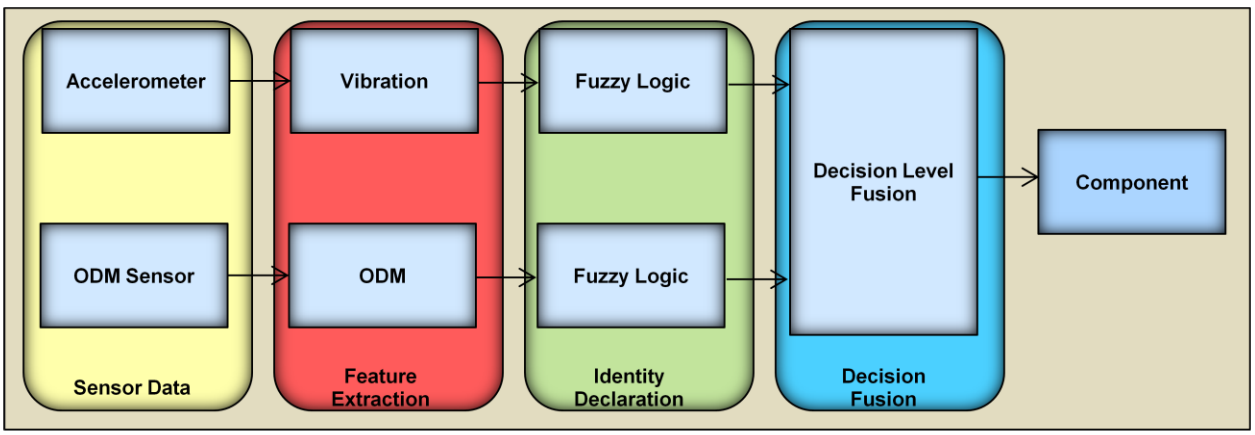

FIGURE M-7: Vibration data and ODM data fusion model [101]

As shown in the matrix in Table M-III, the combined output from the vibration and ODM sensors provides a more comprehensive state of the bearing and suggested maintenance action. Integration of condition indicators, as shown in Figure. M-7, should be practiced, where possible.

TABLE M-III: Combine ODM and vibration output

	Wear Debris Status		
Vibration Status			
	Low	**Medium**	**High**
Low	OK	OK	INSPECT
High	INSPECT	INSPECT	REPAIR/REPLACE

[101]Dempsey, P. J., G. Kreider, T. Fichter, Tapered Roller Bearing Damage Detection Using Decision Fusion Analysis, NASA Technical Report: NASA/TM – 2006-214380, July 2006.