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THE AGING OF ENGINES: AN OPERATOR'S PERSPECTIVE

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ABSTRACT

NATO countries are currently faced with the need to operate fleets of mature gas turbine engines built many years ago. Because of diminishing resources for new equipment, the prospects of replacing these engines with new ones are not good at present. How long such engines can be kept in service safely, without replacing a significant portion of their aging structural components has become a growing concern to engine life-cycle managers, due to uncertainties in residual lives. Another concern is the high maintenance cost associated with the replacement of durability-critical components, such as blades and vanes. The need to balance risk and escalating maintenance costs explains the growing interest in the application of life extension technologies for safely extracting maximum usage out of life-limited parts. In the case of aero-engines, maintaining airworthiness while ensuring affordability is of prime concern to both life-cycle managers and regulatory authorities. This lecture describes the modes of deterioration of engine components and discusses their effects on the performance, operating costs, reliability and operational safety of engines. It also identifies component life extension strategies that engine life-cycle managers may adopt to cost-effectively manage their engines, while ensuring reliability and safety. A qualification methodology for component life extension, developed and implemented for Canadian Forces engines, is presented. The methodology incorporates an Engine Repair Structural Integrity Program (ERSIP) that was conceived to establish structural performance requirements and identify tests for development and qualification of life extension technologies, to ensure structural integrity and performance throughout the extended life. Examples of life extension technologies applied to gas path components and critical rotating parts are described, including the use of protective coatings and repairs to increase component durability. The application of damage tolerance concepts to allow use of safety-critical parts beyond their conventional safe-life limits is also illustrated.

GLOSSARY OF ABBREVIATIONS

AET	Accelerated Endurance Tests
AMT	Accelerated Mission Test
ASMET	Accelerated Simulated Mission Endurance Test
CG	Columnar grain
CIP	Component improvement program
DFM	Deterministic fracture mechanics
DS	Directionally solidified
EBW	Electron beam welding
ENSIP	Engine Structural Integrity Program
ERSIP	Engine Repair Structural Integrity program
FCGR	Fatigue crack growth rate
FEM	Finite element modeling
FM	Fracture mechanics
FMECA	Failure Modes and Effects Criticality Analysis
FOD	Foreign object damage
HCF	High cycle fatigue
LEFM	Linear-elastic fracture mechanics
LCF	Low cycle fatigue
LCM	Life cycle management
LCMM	Life cycle materiel manager
LPI	Liquid penetrant inspection
NDI	Non-destructive inspection
OEM	Original equipment manufacturer
PFM	Probabilistic fracture mechanics
POD	Probability of detection
R&O	Repair and overhaul
SC	Single crystal
SIF	Stress intensity factor
SII	Safe inspection interval
TBC	Thermal barrier coating
TCP	Topologically closed packed
TF	Thermal fatigue
TIT	Turbine inlet temperature

1. INTRODUCTION

Turbine engine components accumulate damage in service as a result of their demanding operating conditions. The damage may take many forms depending on the component, engine type and operating environment [1]. Service-induced damage limits the usable life of many engine components. When damage becomes excessive as revealed by inspection or when component design lives are reached, the components are replaced with new ones. Replacing service-damaged parts is costly and is a significant contributing factor in the overall life cycle cost of engines.

Annual expenditures on replacement parts for modern engines can be well in excess of $50,000 per engine [2].

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Occasionally, that cost may grow significantly when components such as compressor wheels or turbine discs need replacing because the specified safe-life for the parts has been reached. Thus, at a time of tight budgets and diminishing resources for new equipment acquisition or equipment maintenance, it is not surprising that many NATO engine fleet managers have been exploring every possible way to extend the usable lives of expensive components. In the case of old engines, in particular those that were designed more than 40 years ago, the fleet managers may have to face additional challenges due to the scarcity of spare parts or spares not being available. In the absence of spare parts, uncertainties in component residual lives may be cause for concern.

For engine life cycle managers to ask the right questions and make the right decisions, they need to clearly understand how engine components deteriorate in service and how this deterioration impacts on the performance, operating cost, reliability and safety of engines. They also need to know how to take advantage of the latest developments in component life extension technologies to cost-effectively manage their engines. In this respect, it is important for fleet managers to be fully cognizant of the requirements to qualify the life extension technologies for ensuring the engines remain reliable and safe to operate upon return to service.

The objective of this paper is to provide engine managers with guidance and strategies for the development and implementation of new or existing life extension technologies. It describes the modes of deterioration of engine components and how the damage affects structural integrity. It reviews procedures applied to manage aging components and some of the latest developments in component life extension. It explains the Qualification Methodology developed jointly by the NRC Institute for Aerospace Research (IAR) and Orenda Aerospace Corporation (OAC) for the Canadian Department of National Defence (DND). Finally, it provides examples of substantiation test programs that have been used to qualify some recently developed life extension schemes for CF engines.

2. MODES OF DETERIORATION OF ENGINE COMPONENTS

Damage incurred by engine components during service may be external, affecting dimensions and surface finish as a result of fretting, wear, erosion, corrosion or oxidation. It may also be internal, affecting the microstructure of highly stressed and hot parts, as a result of aging of the material microstructure, creep or fatigue. External damage impacts significantly on functionality of the parts, including the aerodynamic performance of gas path components, but may also reduce their load bearing capacity. Surface damage in the form of low cycle fatigue (LCF) cracks, scars or dents, fretting-wear or foreign object damage (FOD), respectively, may lead to high cycle fatigue (HCF) failures [3]. Internal damage may reduce component strength and lead to component distortion. Its accumulation also causes the initiation of flaws, which may ultimately lead to cracking and component failure. Examples of external and internal forms of damage and component distortion and cracking are presented below.

2.1 Surface Damage

2.1.1 *Fretting* is caused by oscillatory motion of very small amplitude between two contacting surfaces. It occurs in engines, for instance, where blades come in contact with discs. The rubbing motion between the components generates debris that is trapped between the two contacting surfaces and may lead to scoring. The resulting surface scar may act as a stress raiser and give rise to fretting-fatigue cracks. Both fan and compressor sections are susceptible to this form of damage. The susceptibility to fretting of blades and discs is governed by many geometric and materials factors. The latter include the difference in hardness between the blade and the disc materials, the magnitude of surface residual stresses and the lubrication of the contact surfaces. Depending on the hardness ratio either one of the components may incur the damage. A close up view of fretting damage incurred by the dovetail of a compressor blade is shown in Fig. 1.

Fretting tends to reduce HCF life, but may also impact on LCF life of the parts. Fretting may also affect splines, couplings, clutches, spindles and seals. It may occur in joints that are bolted, keyed, press fitted, shrunk or riveted. Fretting can be minimized trough use of anti-fretting compounds, including soft coatings, and surface modification treatments [4].

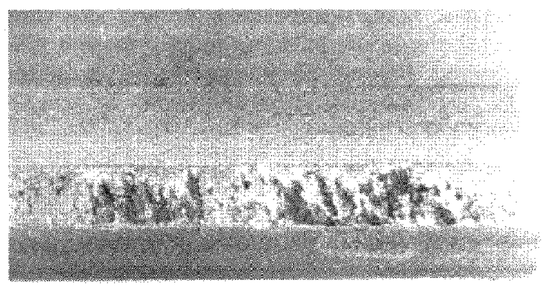


Figure 1. Dovetail of a compressor blade from a CF engine subject to fretting damage, showing deep fretting scar. The surface damage may lead to HCF or LCF failures.

2.1.2 *Erosion* of compressor airfoils is caused by ingestion of hard particulate matter through the engine air intake. The impacting action of gas entrained hard particles on gas path components leads to gradual removal of material, which given sufficient service time can significantly alter component shape and surface finish. The change in geometry can be quite severe as evidenced by the eroded airfoils of the compressor vane segment shown in Fig. 2. Change in leading edge radius and surface roughening can significantly reduce compressor efficiency and engine performance. In extreme cases, geometry changes may alter the modal response of the airfoils, to the point where they may be excited at their natural frequencies under normal operating conditions. Such resonant vibration may lead to HCF failures of blades. Erosion of compressor gas path components can be minimized through the use of inlet particle separators or by enhancing the erosion resistance of airfoils through use of protective coatings or surface modification treatments [5].

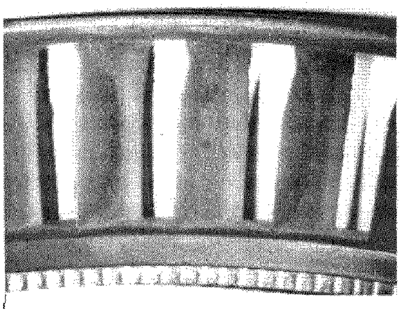


Figure 2. Erosion damaged CF T56 compressor vane segment after approximately 5000 hrs of service.

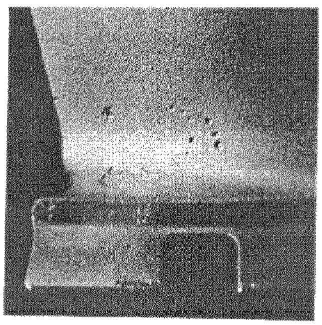


Figure 3. Corrosion pits close to the root of a compressor blade exceeding damage allowable limits.

2.1.3 Pitting corrosion may occur in the presence of environments containing a high concentration of chloride ions, for instance in a marine environment. Both stainless steel and titanium alloys are particularly susceptible to this form of damage. Corrosion pits can have a dramatic effect on the structural integrity of compressor parts. Depending on the type of component and the location of the corrosion pits, they may act as crack initiation sites for HCF or LCF cracks. Corrosion pits of the type shown near the root of the compressor blade, Fig. 3, would normally be cause for blade rejection [6]. If left unchecked, this form of damage may lead to catastrophic component failure. Corrosion control may be achieved through use of protective coatings[7].

2.1.4 Oxidation damage is cause for rejection of a significant fraction of turbine blades and vanes during scheduled engine maintenance. Oxidation reduces the thickness of airfoils and their load bearing capacity. It may also give rise to sites for HCF or LCF crack initiation. In cooled blades, oxidation of normally uncoated internal cooling passages may combine with external surface oxidation to reduce component wall thickness and, thus, the load bearing capacity of the blade [8]. The coatings that are applied to most of these parts, over their airfoil portions, provide some protection against high temperature oxidation. The coatings are aluminide intermetallic compounds designed to form a protective oxide scale that prevents ingress by oxygen and further oxidation. However, coating life is limited due to spalling of the protective oxide scales and gradual consumption of the coating elements. Thus, ultimately, even the coated parts need replacing.

Components made of stronger superalloys are often subjected to higher operating temperatures than their weaker counterparts and, therefore, they are exposed to more aggressive oxidizing environments during service. Thus component made of directionally solidified (DS) alloys with either columnar grained (CG) or single crystal (SC) microstructures are often life limited due to surface oxidation. The problem can be quite severe at turbine blade tips, because any protective coating wears off rapidly at this location, due to tip rub. In the case of CG blades, tip oxidation may lead to thermal fatigue cracking of the longitudinal grain boundaries that are embrittled as a result of boundary oxidation, Fig. 4 [9].

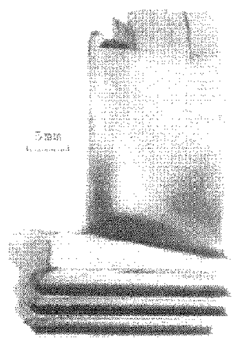


Figure 4. Tip cracking of a DS-CG blade caused by thermal fatigue of longitudinal grain boundaries embrittled as a result of grain boundary oxidation.

Blades and vanes may be recoated in conjunction with other refurbishment or repairs performed during engine overhaul. Tip repairs are an option for CG and SC blades [10].

2.1.5 Hot corrosion is a form of accelerated high temperature oxidation/sulphidation that may destroy hot gas path components when they are exposed to deleterious mixtures of reactive contaminants originating from the fuel and the ingested air. At temperatures in the range from 850 to 950°C, residual sulfur from the fuel combines with sodium chloride picked up from a marine environment to form sodium sulfate. This corrosive compounds fluxes protective oxide scales, which destroys coatings, thereby exposing the substrate material to the environment. Once a hot corrosion reaction starts, it will quickly consume the component as shown in Fig. 5 [8]. Substituting a protective coating for another that has more resistance to hot corrosion may be a cost-effective option. The addition of a coating to the internal cooling passages of blades and vanes is also an option when hot corrosion occurs within the cooling passage [11].

2.2 Internal Damage

Another form of deterioration of engine components results from the accumulation of internal microstructural damage. This is an insidious form of damage because, in contrast to surface damage, it cannot be readily detected by non-destructive inspection (NDI) techniques. Its rate of accumulation is strongly influenced by service stresses and temperature and, therefore, to a great extent by user practice. Because there are uncertainties in the temporal variation of service stresses and temperatures, internal microstructural damage cannot be easily predicted.

Internal microstructural damage is the result of plastic strain accumulation and metallurgical aging reactions [1]. Plastic strain accumulation is the product of fatigue and/or creep, which affects highly stressed and hot parts, such as compressor and turbine discs, as well as turbine blades and vanes. Metallurgical aging reactions affect mostly hot gas path components. They may also occur in the rim of discs in small turbines, which often operate at higher operational speeds and temperatures than do larger turbines [9].

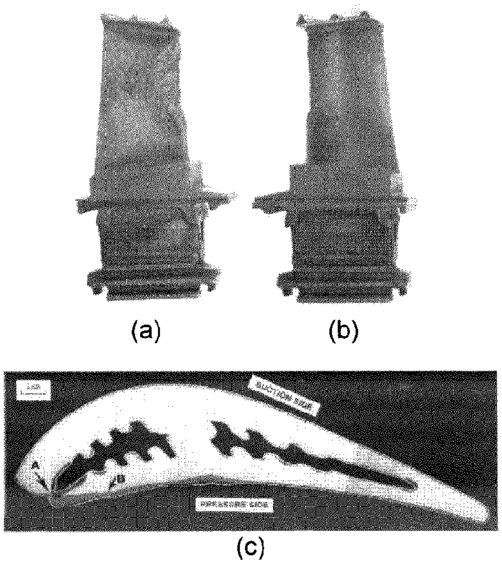


Figure 5. Hot corrosion damage incurred by a Mar-M246 blade in the engine of a CF maritime patrol aircraft (a) pressure side view; (b) suction side view; (c) metallographic section taken halfway across the airfoil through the hot corroded region [8].

2.2.1. Plastic strain accumulation manifests itself in the form of dislocation substructures, including persistent slip bands and wavy slip, which develop under cyclic thermo-mechanical loads at stress concentration sites such as disc bores and bolt holes or the rim region of rotors, prior to crack initiation. The dislocation substructure shown in Fig. 6(b) is indicative of high temperature LCF damage accumulation in turbine discs [12]. Such form of micro-damage accumulation is mostly dependent on number of applied load cycles. Plastic strain accumulation also manifests itself as creep deformation leading to formation of creep cavities and internal cracks, Fig. 6(d) [1].

2.2.2 Metallurgical aging reactions include coarsening, agglomeration and rafting of gamma prime precipitates in nickel base superalloys, carbide coarsening in cobalt base superalloys, degeneration of primary carbides into continuous carbide films along the grain boundaries of polycrystalline alloys, and precipitation of topologically closed packed (TCP) phases, such as sigma and/or mu phases within the interdendritic regions of the alloys [1]. Such reactions are mostly time dependent, although to some degree some can be stress assisted as well. For instance, in DS CG or SG alloys, the stress field within components dictates the orientation of gamma prime precipitate rafting.

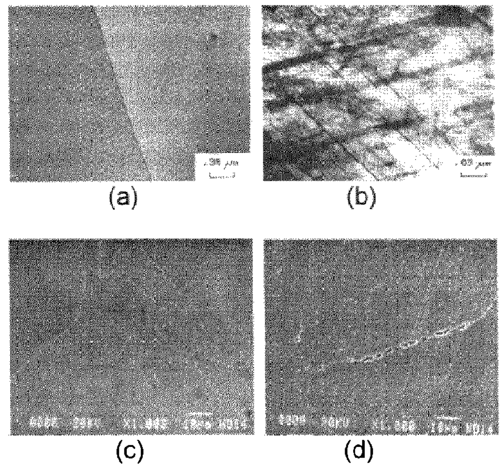


Figure 6. Effects of service exposure on microstructure (a) Virgin disc; (b) Service exposed disc showing evidence of dislocation activity (PSB) indicative of LCF damage accumulation (c) Virgin disc spacer; (d) Service exposed disc spacer showing evidence of creep voids along grain boundaries; [1,12].

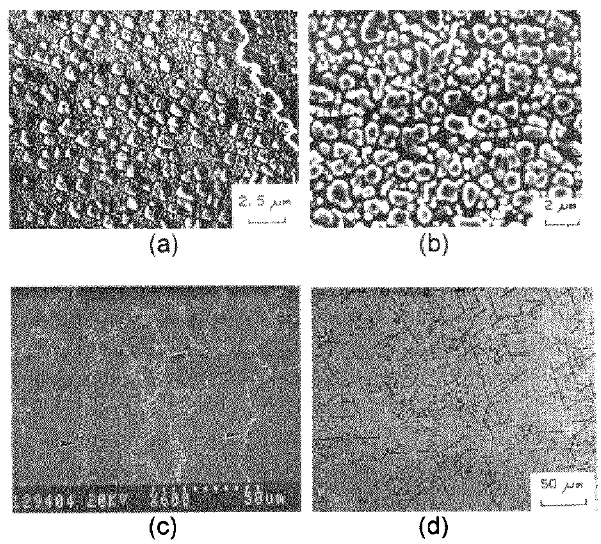
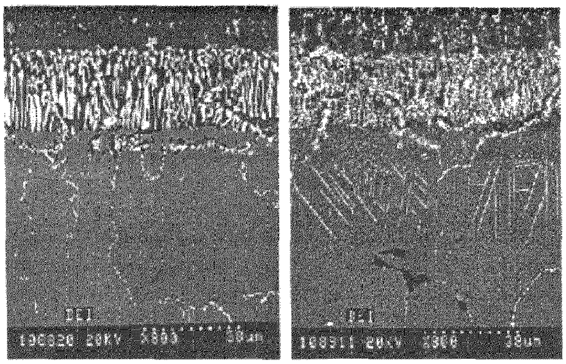


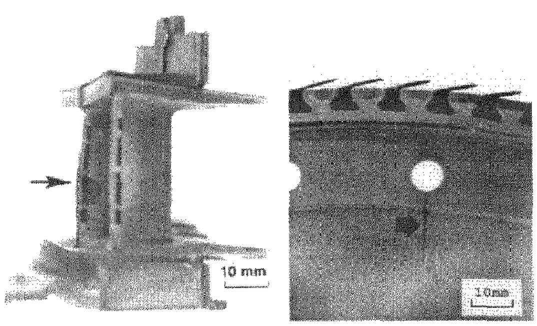
Figure 7. Internal microstructural damage in high time blades: (a) bimodal distribution of gamma prime precipitates in new blade; (b) coarsening of gamma prime precipitates and elimination of secondary gamma prime in service exposed IN738 turbine blade; (c) precipitation of carbides (arrows) along grain boundaries in IN738 turbine blade; (d) precipitation of sigma phase in IN 713C blade [13,14].

Examples of gamma prime coarsening, carbine reactions and TCP phase precipitation in nickel base superalloys are shown in Fig. 7 [13,14]. Precipitate coarsening and TCP phase formation reduce material strength, whereas continuous carbide film formation and TCP precipitates embrittle the material, therefore, making components notch sensitive. The precipitation of sigma phase immediately below the protective coatings, Fig. 8, compounds the notch sensitivity problem during service.



(a) (b)

Figure 8. Service-induced precipitation of sigma phase immediately below the protective coating in a Mar-M246 turbine blade from a CF transport aircraft engine; (a) new blade; (b) service-exposed blade.



(a) (b)

Figure 9. (a) Airfoil bowing, loss of coating and cracking of a CF T56-A15 first stage nozzle guide vane; (b) Crack initiating from bolt hole in CF j85 CAN40/15 compressor disc [15,16].

2.3 Distortion and Cracking

Under creep loading conditions, the loss of strength caused by service-induced metallurgical ageing leads to component distortion. Vane airfoil bowing or lengthening and untwist of turbine blade airfoils is commonly observed in aero engines Fig. 9(a) [15]. Extreme airfoil distortion may lead to HCF failures. Cracking due to cavity link-up, or oxidation-assisted creep cracking along the grain boundaries, usually follows microstructural damage accumulation, in conjunction with component distortion. Microstructural damage accumulation under LCF loading conditions is also followed by crack initiation and growth in highly stressed components such as discs, Fig 9(b) [16]. This type of cracking may lead to catastrophic failures.

3. MANAGING THE DETERIORATION OF ENGINE COMPONENTS

For the purpose of life cycle management, engine components may be classified as either durability-critical or safety-critical [17].

(a) Durability-critical components, includes those whose deterioration affects mainly engine performance and fuel efficiency and may result in a significant maintenance burden but will not normally impair flight safety. These parts include cold and hot gas path components such as blades and vanes. The life of these parts is limited by one of a variety of possible modes of damage accumulation, which vary with the type of component and its operating environment.

(b) Safety-Critical components, includes those whose fracture may result in the loss of the aircraft if the fracture is not contained. These components include most of the large rotating compressor and turbine components, such as wheels, discs, spacers and shafts. The life of these parts is usually limited due to LCF damage accumulation.

For *durability-critical parts*, life limits are rarely specified. An *"On-Condition"* maintenance approach is normally used, where parts are removed from service when measurable damage limits are reached [18]. Damage limits are specified in R&O Manuals, including for instance limits on the extent of corrosion pitting, elongation or untwist of turbine blades or cracking of turbine vanes. No "hard time" life is set for these parts, although a minimum life expectancy may be guaranteed by the original equipment manufacturer (OEM) for turbine blades and vanes. However, the difficulties associated with predicting service behaviour of metallurgically complex blade and vane materials, under conditions that can vary widely with user practice, make it difficult to reliably establish life expectancies. These difficulties are made worse by complex coating-substrate interactions that are not usually well understood . Consequently, it is not unusual for shortfalls to be experienced in life expectancies of turbine blades or vanes.

For *Safety-Critical parts*, whose lives are limited by LCF, two life cycle management approaches may be used [18]:

(a) The *"Safe-Life"* approach, for which all components are retired before a first crack is detectable, and

(b) The *"Damage Tolerance"* approach, for which component lives are established on the basis of fracture mechanics (FM) principles and retirement of parts is based on a number of life cycle management (LCM) options, as explained below.

3.1 The Safe Life Approach

The Safe-Life approach for LCM of safety-critical parts assumes that when a crack appears, the component has failed, and it further ensures that all similar components are retired at an equivalent life, before the first crack appears. To that end, the approach follows a "cycles to crack initiation" criterion, with a minimum safe life (also called hard time life) capability established statistically through extensive mechanical testing of test coupons and components under simulated service conditions. The statistical minimum is based on the probability that only 1 in 1000 component (-3σ) will have developed a detectable crack, typically 0.8 mm in surface length, at retirement, Fig. 10. This approach is known in the UK as the life-to-first-crack (LTFC) method, for which the detectable crack radius is chosen to be 0.38mm.

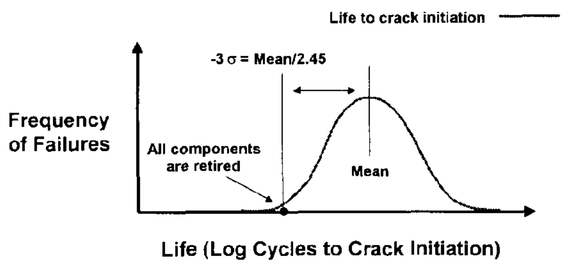


Figure 10. The Safe-Life approach.

The advantages of the Safe Life approach are that the maintenance requirements are kept to a minimum, while the time in service of components without inspection is maximized. The main disadvantage of the Safe-Life approach is that it may be overly conservative, because components are retired with a significant amount of useful residual life (by default, 999 out of 1000 components are retired with significant life remaining) [17,18]. Furthermore, the method is costly since all parts may need replacing nominally all at the same time. Finally, the availability of spares may be a problem for old engines.

3.2 The Damage Tolerance Approach

The Damage Tolerance approach assumes that fracture critical areas of components contain crack-like manufacturing or service-induced defects giving rise to the propagation of cracks during service. It further assumes that components are capable of continued safe operation as the cracks grow under thermal and mechanical stresses. Cracks are assumed to grow in a manner that can be predicted from linear-elastic fracture mechanics (LEFM), or other acceptable methods. Cracks are also assumed to grow sufficiently slowly to allow their detection through regularly scheduled inspections. Finally, the approach follows an inspection schedule established by analysis to ensure that cracks will not grow beyond a dysfunction limit, in between inspections.

The time interval between the scheduled inspections, or *safe inspection interval (SII)*, is calculated as the time it takes a hypothetical crack to grow from a size immediately below the detection limit, to a dysfunction crack size, beyond which the risk of rapid or unstable crack growth becomes too high. The dysfunction crack size is obtained for an assumed crack geometry from the fracture toughness of the material and the crack tip stress intensity factor, using appropriate safety factors. In practice, the SII is assumed to be one half the safety limit. The approach is described schematically in Fig. 11.

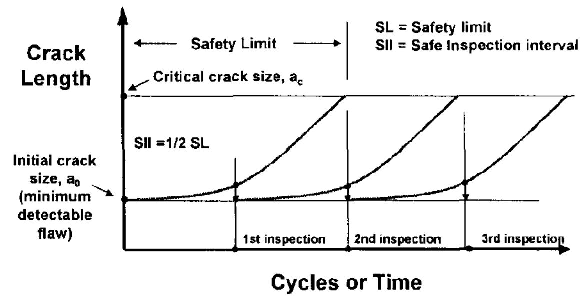


Figure 11. The Damage Tolerance approach.

For damage tolerance based LCM of safety-critical parts, two implementation methods may be used, one covered under MIL-STD 1783 (ENSIP) and the other known as Retirement for Cause (RFC) [18]. A third option known as 2/3 Dysfunction Life may also be practiced [19]. Each approach is explained below.

3.2.1. ENSIP (MIL-STD-1783)

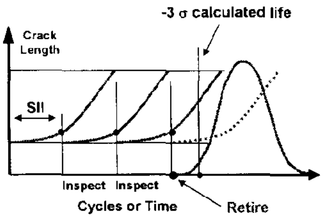
The Engine Structural Integrity Program (ENSIP) is an organized and disciplined methodology introduced in 1984 by the USAF for the structural design, analysis, development, production and life management of engines. ENSIP embraces a damage tolerance approach to set *safe inspection intervals* for life management of safety-critical parts. However, conventional structural design criteria are also used to minimize risks of failure due to vibration, LCF, HCF and creep. Individual parts (e.g. compressor and turbine discs) are retired once their safe-life (LCF life to crack initiation) is reached, Fig 12(a) [18].

3.2.2 Retirement for Cause

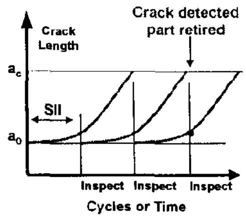
The Retirement for Cause concept is a damage tolerance based method for managing the life of safety critical parts. The method relies on fracture mechanics to set a Safe Inspection Interval (SII) as practiced under ENSIP. Retirement life is based on periodic inspections until a crack is detected, at which point the part is retired. Individual parts, (e.g. compressor and turbine discs) are retired once they are found to contain a crack, Fig 12(b) [18].

The advantages of a damage tolerance based approach for LCM of safety critical parts are twofold. Firstly, the approach ensures that cracks emanating from manufacturing defects (or service induced cracks) in any one of the components will not grow beyond dysfunction size. Secondly, it allows life extension beyond LCF safe-life limits if and when needed, through use of the RFC technique [17,18].

However, there are also some disadvantages. The damage tolerance approach is more costly to implement than the safe life approach. It requires an elaborate NDI infrastructure to support increased inspection requirements. Also, the handling of components is increased. Finally, in the context of new engine designs it may add weight to components that are conceived for damage tolerance. Such components may require thicker sections to better tolerate the presence of cracks or to ensure low crack growth rates [17,18].



(a) ENSIP



(b) RFC

Figure 12. Schematic representations of the damage tolerance approaches based on (a) MIL-STD 1783 (ENSIP); (b) Retirement for Cause (RFC).

3.2.3 The 2/3 Dysfunction Life Approach

With this approach, developed in the UK, an acceptable but variable margin of safety is given by replacing the detectable "engineering crack" of the Safe-Life approach by a set fraction of "2/3 of the total life to dysfunction", that is the life before the onset of rapid crack growth. The value of 2/3 is chosen because experience has shown that the crack size at "2/3 failure" is approximately equal to 0.38 mm, as defined in the LTFC method. For damage tolerant designs, in which the 2/3-dysfunction life exceeds the LTFC, this approach provides opportunity for life extension. For such cases, FM crack propagation calculations may be used to determine the available service life beyond "first crack" [19].

Whatever is selected as the LCM approach, safe and reliable prediction of component cyclic lives or safe inspection intervals requires an accurate knowledge of the stresses and temperature in the parts, as well as their variation with time. A detailed knowledge of the response of structural materials to external forces and the environment, especially in terms of active modes of damage accumulation and their interactions is also important. Experience shows that there are significant uncertainties associated with all these external variables, as well as with the accuracy of structural analysis methods and damage modeling methods employed for predicting lives.

4. MANAGING NATO'S AGING ENGINE FLEETS

Several of the NATO countries are faced with having to operate old engines for which the extent of damage incurred by fracture-critical parts, and therefore the residual lives of the parts, are not accurately known. Owing to diminishing resources for new equipment, it is not likely that these engines will be replaced soon. How long old engines can be safely kept in service without having to change a significant portion of their structural components has been a growing concern amongst life cycle materiel managers (LCMM). Another concern with old engines is the high rejection rates usually experienced for durability-critical parts, such as gas path components. The high replacement cost of these parts is a significant life cycle cost driver in mature engine fleets and a concern to fleet managers from an affordability point of view. The need to balance risk and affordability, has provided incentives for LCMMs to identify and implement strategies for extracting maximum usage of expensive to replace components, while ensuring the engines remain safe to operate and are reliable in service.

4.1 Engine Usage Monitoring as a Tool for Component Life Extension

One strategy for extending component lives, while ensuring safety, is to equip mature engines with modern health and usage monitoring systems (HUMS). In principle, the temporal variation of any of the engine parameters responsible for component damage can be measured with on-board monitoring systems. By applying suitable transfer functions to predict the residual lives of parts from knowledge of their operating conditions, life extension may be possible when the assumed mission severity is overly conservative.

A more precise estimate of parts life consumption obtained through the use of HUMS, in combination with an engine parts life tracking system (EPLTS), allows operators to achieve more optimal use of parts life potentials and may help them avoid in-service premature failures. In addition, engine inspection schedules and parts removal can be optimized to achieve a more cost-effective maintenance program, while ensuring safety.

However, predicting parts life consumption from knowledge of their operating conditions requires accurate models to predict rates of damage accumulation. Such models are often empirical and not very reliable. Because of related uncertainties, large safety factors are normally imposed on LCF and other life predictions. The use of more reliable damage accumulation models would preclude the need for conservatism. In this context, defining the role of environment on creep-fatigue interactions and the impact of microstructural ageing reactions of mechanical properties of engine materials is most important [1]

4.2 Other Strategies for Component Life Extension

Other component life extension options are available to engine fleet managers.

For durability-critical parts, the options include returning service-damaged engine parts to functional serviceability through use of repairs, such as welding, brazing, rebuilding, re-contouring and rejuvenation heat treatments, or delaying the rates of damage accumulation through the addition of protective coatings or surface modifications treatments. A change of material to a more durable alternative is also an option.

For safety-critical parts, the option is to implement a damage tolerance based Retirement for Cause approach for life cycle management of the components.

4.3 Implementation Requirements

The decision to replace component or extend its life by means of one of the above options, must consider the operational consequences of component failure, the cost effectiveness of the proposed life extension technique and the substantiation testing that will ensure the parts remain airworthy, safe and reliable through the life extension.

4.3.1 Safety Considerations

The associated risks depend on whether the component is functionally or structurally significant. This can be best addressed through a reliability analysis tool known as a Failure Modes and Effects Criticality Analysis (FMECA). A FMECA is a powerful tool, usually computerized, that leads fleet managers through modes of failures that can occur on components, identifies the probability of those failures and the possible consequences of failure. It provides data on component failure rates obtained from various sources (e.g. field experience, R&O, OEM, CIP, FMS), on the basis of which Fault Tree Analysis (FTA) and Reliability Block Diagram (RBD) models can be developed as management tools. A FMECA provides a basis for maintenance logistics analysis [2,18].

4.3.2 Cost Considerations

Operators must also decide on cost-effectiveness of the proposed life extension scheme for the targeted application. A cost-benefit analysis (CBA) requires that consideration be given to rates of component rejection, the cost of new parts, the cost of developing and qualifying the proposed life extension technology and the cost of applying the technology in a service environment [21,22]. A sufficient return on investment is normally required to justify implementation of any life extension scheme, unless circumstances dictate otherwise. This may happen, for instance, when no spare parts are available due to a supply shortage, or because spare parts are not produced anymore for a very old engine.

4.3.3 Technical Considerations

Implementation of any component life extension scheme outside the scope of an OEM maintenance manual, or its application beyond OEM specified damage limits, requires that the process be carried out to the same standard the original product was qualified to, or to an equivalent standard. For military aero-engines, the applicable standards are MIL-E-5007D, MIL-E-8593E (AS), MIL-STD-1783 (ENSIP) or MIL-STD-1529, although other standards may apply depending on the engine type and country of origin [17, 23]].

MIL-STD-1529 (Vendor substantiation for aerospace products) describes procedures to qualify additional/alternate vendor and fabrication sources other than those qualified for the original product. This standard is therefore particularly relevant to qualification of life extension schemes [23]. However, it does not specify technical requirements. It simply states that "substantiation tests include but are not limited to those tests required by applicable Government or engine manufacturers' specifications". The technical requirements are contained in these specifications.

Judging from the Critical Parts Qualification Procedures for T56 Engines [24], substantiation tests for qualifying parts subjected to life extension might consist of the following generic elements, depending on type of components and their criticality:

(1) Dimensional inspection to ensure that the parts conform to drawing requirements;

(2) Metallurgical verification to ensure that the materials conform to engineering specifications;

(3) Structural tests to ensure that the relevant mechanical properties (Creep, LCF, TF,...) are equivalent to or better than properties of original parts as dictated by engineering specifications;

(4) Functional tests to ensure parts functionality is not impaired by the life extension process (e.g. cooling flow rates for internally cooled parts are identical to flows in original equipment); and

(5) Rig and engine tests to verify that the parts after testing still meet the serviceable limits specified in the applicable R&O Manual and that their general condition is comparable to that of approved parts subjected to an identical test.

Two types of engine test are specified in MIL-E-008593E (AS), the standard used to qualify the various models of T56 engines. These tests are:

1. *Accelerated Endurance Tests (AET)*

These are over-temperature tests, typically of 150 - 300 hour duration, designed to assess the durability of hot section components, wear in mechanical parts and impact of vibration of engine components. The tests are normally performed at a turbine inlet temperature (TIT) at least 8°C above maximum allowable steady state TIT for power settings at or above the maximum continuous level. The tests may be preceded and followed by a 25 hour stair-step test schedule involving rotational speed and power transients. Once the test is completed, engine performance (power, specific consumption) is compared with that of specified rating limits. The engine is then disassembled and its parts are inspected for qualification purposes

2. *Accelerated Simulated Mission Endurance Tests (ASMET)*

These are tests designed to simulate the effects of thermal exposure, LCF and thermal fatigue on the durability of compressor and turbine components. The tests are typically of 600 - 2000 hour duration, during which the engine is subjected to accelerated mission oriented cycles that are developed from the expected mission profile of the aircraft.

5. THE CANADIAN EXPERIENCE

The Canadian Forces (CF) have recently adopted a qualification methodology developed jointly by IAR and OAC for component life extension, which is being used to implement advanced repairs and other life extension processes for a variety of CF aero engine parts, including components considered to be flight critical. A knowledge base and testing infrastructure has been established to support these technologically challenging tasks, with financial assistance from DND [21].

Development of the methodology started by a careful review of civil and military regulatory agency requirements applicable to the design and life cycle management of aero engines [23]. The reviewed standards included the United States Federal Air Regulations (FAR 33), the Canadian Airworthiness Regulations CAR Part V (Engine Standards - 533), the European Joint Airworthiness Regulations (JAR-E) as well as the US military standards MIL-E-5007E, MIL-E-8593A, MIL-STD-1783 (ENSIP), MIL-STD 1529 and the UK military standard Def Stan 00-971. From this review, it was apparent that the military specifications and standards are generally more stringent than civil requirements. Some standards rely more heavily on testing, including full-scale rig and engine testing. The methodology developed for the Canadian Forces is largely derived from this review. It incorporates Transport Canada requirements for Design Approval applications, which allows commercial operators to take advantage of the methodology.

The Canadian Forces Methodology for LCM of Aging Engines is an organized engineering approach consisting of:

1) a *Failure Mode, Effects and Criticality Analysis* (FMECA) to establish the criticality of damage found in components and its effects on aircraft performance and operation;
2) a *Repairability and Cost Benefit Analysis* (RCBA) to establish whether damaged components should be replaced or repaired; and
3) an *Engine Repair Structural Integrity Program* (ERSIP) to ensure that the parts to which a life extension process is applied will remain airworthy, safe and reliable through the life extension.

The FMECA package developed for the CF is a stand-alone reliability analysis tool designed to assist the CF LCMMs with engine maintenance. The CF FMECA identifies potential failure modes of hardware and the effects of failure on system performance, reliability and safety. It has been designed to assist CF LCMMs with identification of root failure causes and the development of corrective actions. It also identifies durability/safety critical components requiring special management approaches [2].

The Canadian ERSIP was conceived to establish structural performance requirements and identify tests for the development and qualification of life extension technologies [25]. ERSIP modifies and extends the limits of MIL-STD-1783 (ENSIP) to satisfy operator's needs for the management of components subjected to life extension processes. It incorporates the damage tolerance approach implied by ENSIP and is used by the CF to establish structural performance, process development and verification requirements that will ensure structural integrity of components subjected to life extension. The ultimate goal of ERSIP is to ensure structural safety, durability, reduced life cycle cost and increased service readiness of engines.

The application of ERSIP involves five generic tasks, the purposes of which are summarized below [25].

Task I: Original Design and Service History Information
Calls for alignment of the proposed life extension scheme with the original structural design criteria to ensure operational needs and requirements are satisfied (may require that some design data be provided by the OEM).

Task II: Process Development and Design Modifications
Calls for coupon level qualification and quality control testing plans to develop the life extension process and any design modifications, if applicable.

Task III: Component Tests
Calls for assessing the strength, damage tolerance, durability and dynamic response characteristics of components processed for life extension.

Task IV: Engine Tests
Calls for verifying component performance in test engines either under simulated service conditions via an ASMET, or an Accelerated Mission Test (AMT) designed to simulate the life limiting damage modes of interest [2], or through field testing.

Task V: Life Management
Provides a plan for the management of repaired engines based on a life extension interval (LEI) greater or equal to the time between overhaul (TBO).

The overriding goal of ERSIP is to ensure that parts will remain airworthy, safe and reliable through the life extension. The technologies that are being addressed through ERSIP include:

- Restoration of damaged components to serviceable conditions (e.g. by repair or rework) in conformity with the original requirements, or those of ERSIP;
- Modifications intended to improve the structural durability or damage tolerance of an engine component (e.g. a material change, the addition of a coating or use of a surface treatment), in conformity with the requirements of ERSIP;
- Reuse of components under a damage tolerance-based life cycle management scheme to achieve life extension, in conformity with the requirements of ERSIP.

6. LIFE EXTENSION THROUGH RESTORATION (REPAIRS AND REWORKS)

R&O manuals contain information on some basic repairs that may be applied at overhaul to allow damaged components to be returned to service. As engines mature and service experience accumulates, it is not unusual to find out that some components are incurring damage that was not anticipated by the OEM and for which no repair procedure has been developed. Revisions to service bulletins and manuals for military engines are issued from time to time by OEMs to introduce new repairs or changes in design and materials with a view to addressing component durability shortcomings. This is most often done under a Component Improvement Program (CIP). However, the introduction of new repair or rework procedures by the OEMs often falls behind advancements in technology. The latter provide opportunities to create new repairs or implement new design or material changes, such that components experiencing unforeseen deterioration can be returned to service, instead of being scrapped [26]. New technologies may also allow parts to be repaired beyond the damage limits envisaged and set by the OEM as given in the standard R&O Manuals.

Much work has been done in the commercial world to develop repairs and reworks for civilian aircraft engines. Repair vendors have been competing quite successfully with OEMs in these developments. The delegation of authority by National Aviation Authorities to R&O Organizations has encouraged such initiatives. The technology developed for civilian products is for the most part applicable to military platforms and can be adopted by military organizations to achieve cost-effective management of their engines [2,26]. The following listing identifies types of repair/rework technologies that CF engine fleet managers have considered in order to achieve component life extension:

- New welding materials and techniques for cracked and worn parts;
- Advanced brazing or diffusion bonding techniques for cracked components;
- Rejuvenation of properties in creep or fatigue damaged parts by HIPing and/or heat treatment; and
- Surface rebuild or re-contouring for eroded, corroded or worn parts.

6.1 Advanced Welding Repairs

Welding technologies have long been used to repair cracked gas path components such as vanes and blades and combustor liners. Procedures are well established and detailed in the engine R&O Manuals. However, the emergence of new techniques makes it possible to repair components that are not covered by the standard manuals or beyond the limits set by the OEM [1].

For instance, OAC has recently developed a patch repair for eliminating FOD in F404 fan and compressor blades using electron beam welding (EBW) [27]. The damaged area is replaced by a patch made of the same material as the blade, which is joined to the airfoil by EBW. The repaired assembly is then finished-machined to meet dimensional specifications, **Fig 13**. These blades have been shown to perform as well as new blades in qualification tests and will shortly be field-tested in a CF engine. A similar process is used at Pratt & Whitney Canada (PWC) to repair airfoils of integrally machined titanium alloy impellers and blisks [9].

The substantiation tests used to develop and qualify the weld repaired fan blades included evaluating the microstructure, tensile strength and fatigue properties of welded test coupons as well as comparing properties of new original equipment with properties of repaired parts. The comparison included vibration characteristics, fatigue properties and impact resistance of the components. The instrumented ballistic test rig developed to compare the resistance to impact by a high velocity projectile of new and weld repaired fan blades, is shown in Fig 14. The weld-repaired components were finally subjected to testing in an engine, as part of a CF F404 accelerated mission test (AMT) performed on behalf of the CF at IAR. This AMT was used to qualify the fan blade repair as well as several other life extension schemes developed by OAC and IAR.

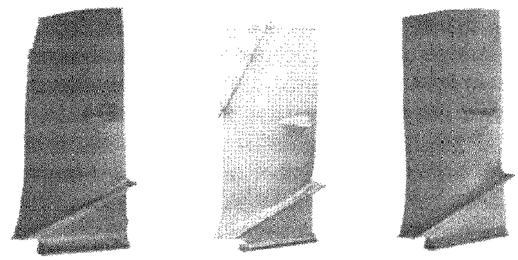


Figure 13. Repair of FODed F404 fan blade by electron beam welding a corner patch to replace damaged portion of blade.

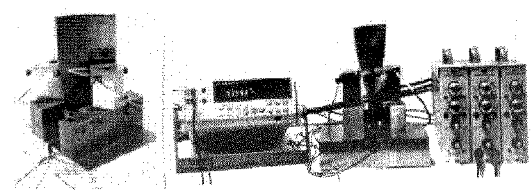


Figure 14 Instrumented ballistic test rig for assessing resistance to impact by high velocity projectile of EBW repaired fan blades.

Techniques such as Dabber™ welding [28], or powder metallurgy welding techniques [29], as well as pulsed laser or plasma welding techniques are making it possible to weld high strength superalloy components that were previously difficult to weld. Automated welding systems, in combination with laser or micro-plasma power sources, have allowed the application of weld metal consistently, using extremely low heat input to minimize effects on the base material [10].

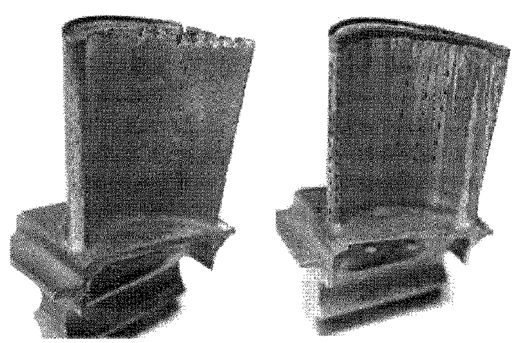


Figure 15. Weld tip repair of DS René 80 first stage turbine blade form CF F404 engine (Courtesy Liburdi Engineering, Hamilton, Ontario)

The tip repair of a F404 directionally solidified first stage high-pressure turbine (HPT) blade is illustrated in Fig. 15. Here, the tip oxidation/cracking damage was first removed by grinding. The blade tip was then rebuilt by vision assisted automated plasma arc welding, using weld material chosen to provide enhanced oxidation resistance [10].

The repair of worn seal teeth from rotating air seal components is another economically desirable refurbishment procedure. The knife-like circumferential seal teeth on the outer surface of air seal components that intrude into a surrounding abradable shroud, are susceptible to wear during service. The teeth also get damaged during assembly and disassembly at overhaul. The tips of the damaged teeth are first ground down and then rebuilt with over-lays of similar alloys, using welding techniques such as Dabber™ [28] welding or pulsed laser or plasma torch welding [10]. An example of refurbished seal teeth prior to final machining to the desired shape is shown in Fig. 16.

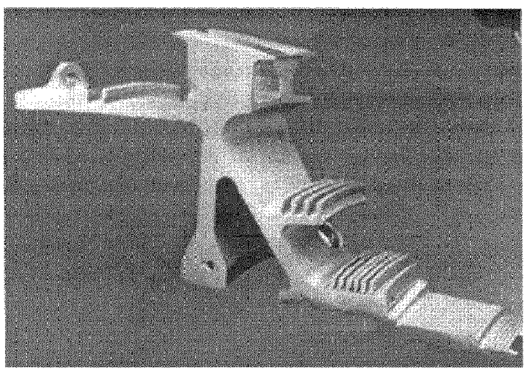
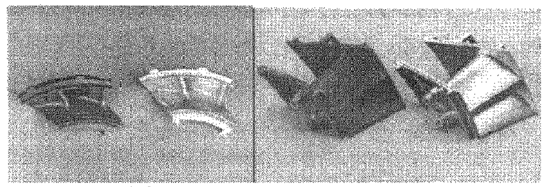


Figure 16. Example of seal teeth repair using the Dabber™ technique (Courtesy of Liburdi Engineering, Hamilton, Ontario).

Substantiation tests and analysis for weld repair of turbine hot section components may be quite elaborate. For the seal teeth repair for instance, it is necessary to assess the risk of cracks initiating at imperfection in the built up weld or at the weld metal/parent metal interface. The presence of undetected cracks at these locations may cause a significant debit in LCF properties of the component. Test results reported by Domas [28], obtained with simulated seal teeth specimens, in combination with fracture mechanics analysis, provided an insight into the likely performance of repaired components, and contributed to the establishment of process capability and margins in relation to specification limits.

6.2 Advanced Brazing Repairs

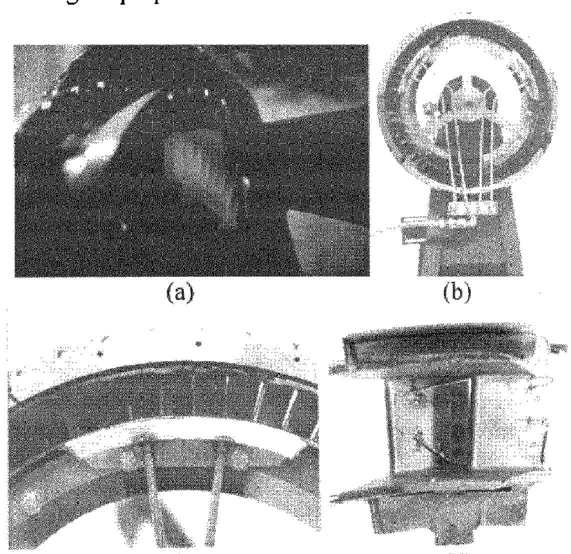
The development of hydrogen fluoride (HF) based cleaning processes such as the AFOR-DBR of Vac-Aero International or the University of Dayton Research Institute (UDRI) oxide reduction process, in combination with diffusion brazing technology, has made it possible to repair previously unbrazable nickel base superalloys. These processes rely on gaseous HF to remove thermodynamically stable oxide from crack faces, thus promoting wetting and infiltration of the braze alloy by capillary forces and, thereby, creating a structurally sound and durable joint, Fig 17 [10, 30].



(a) (b)

Figure 17. (a) INCO 792 compressor turbine test pieces in the as received condition and after AFOR/DBR processing (Courtesy of Vac-Aero International); (b) HF cleaned and braze repaired René 80 LPT nozzle (Courtesy of Orenda Aerospace Corporation).

The Liburdi Engineering LPM™ joining/cladding process [10] was developed as a hybrid wide gap brazing technique that has proven successful for the repair of both blades and vanes. The process enables a wide range of alloys to be used for crack repair and surface buid-up. The damage is first removed by grinding and a powder metallurgy putty of matching or custom composition is applied and diffused into the surface to complete the repair and achieve the desired mechanical and metallurgical properties.



(a) (b)

(b) (d)

Figure 18. Hardware for thermal fatigue testing of NGVs in burner rig: (a) and (b) specimen fixture and cooling arrangement; (b) close-up of specimens mounted in fixture; (c) fine wire thermocouples spot welded on air foils for temperature calibration

Substantiation tests for repairs of turbine hot section components would typically include an evaluation of the microstructure and the mechanical properties (stress rupture, LCF) of welded or brazed joints, using test coupons designed to simulate joint geometry and loading conditions applicable to the repaired part. Rig tests would also be performed on components to ensure for instance that the flow rates through internal cooling passages are identical for new and repaired parts. The effective flow area of a nozzle guide vane is another parameter that would have to be checked for compliance to specifications. Other types of rig tests may be considered to compare the durability and damage tolerance of new and repaired parts under simulated service conditions [31]. Finally, the repaired parts would be subjected to engine testing and/or field evaluation [2].

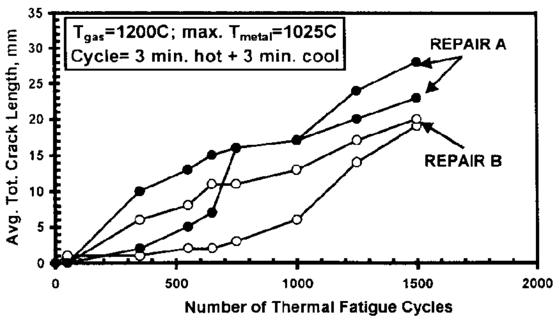


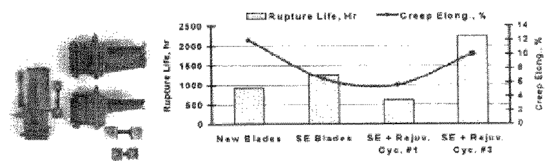
Figure 19. Crack growth rate data during burner rig testing of NGVs repaired by two different processes [31].

At NRC-IAR, engine specific test hardware has been developed for use in a high velocity burner rig to compare the thermal fatigue response of repaired NGVs when subjected to thermal cycling in high velocity gas jets (Mach 0.3 to 0.8). Elements of the hardware used to assess the durability and damage tolerance of repaired T56-A15/A14 first stage NGVs are shown in Fig 18. The burner rig is a Becon LCS-4B combustor fitted with an exhaust nozzle that simulates the geometry of the actual nozzle of the engine combustor. The parts to be tested are mounted in front of this exhaust nozzle, in a wheel assembly made up of an entire set of NGVs, Fig 18(a). During testing, the assembly is rotated back and forth by a pneumatic actuator to alternately expose two sets of tests components to the rig hot gases and a cooling air jet at room temperature. Internal cooling of the test components can be provided, if desired. For T56 engines, three NGV segments are tested simultaneously. While one set of test components is being heated in the hot gas jet, the other set is being cooled in the air jet. The rotation of the wheel assembly occurs at predetermined time intervals to ensure that steady state temperature is achieved during both the heating and cooling portions of the cycles. Hot gas temperature and rate of cooling airflow are adjusted to achieve component surface temperatures that are close to the operating temperatures of the parts. The test is interrupted after predetermined numbers of thermal cycles to inspect the NGVs for the presence and size of cracks. Crack growth rate data obtained for two sets of T56-A15/A14 first stage NGVs repaired by different vendors are shown **in Fig 19** [31].

6.3 Rejuvenation by Heat Treatments

Heat treatments and hot isostatic pressing (HIPing) have been used for over two decades to eliminate service induced microstructural damage in turbine blades and vanes to restore creep properties [1]. HIPing eliminates creep voids that may have formed during service. The re-coating heat treatment completes the rejuvenation. HIPing also eliminates casting porosity, which improves component reliability by reducing scatter in material properties. HIP rejuvenation cycles have been developed at IAR for alloy 713C and IN 738, both of which are used in T56 engines [32,33], . HIPing can also be used in conjunction with other forms of repairs, for instance to eliminate shrinkage porosity within braze joints [30].

The qualification of HIP rejuvenation treatments requires that mechanical properties of rejuvenated components be established from tests performed on specimens machined from components. Miniature creep specimens machined from T56 turbine blades are shown in Fig. 20(a). The rupture life and creep elongation data shown in Fig. 20(b) for new, service-exposed and rejuvenated IN 738 blades indicate that loss of creep ductility induced by service can be fully recovered by HIP rejuvenation, while time to rupture relative to new blades can actually be improved, providing post HIP heat treatments are carefully designed [33].



(a) (b)

Figure 20 (a) Miniature creep specimens machined from the airfoils of different T56 turbine blades; (b) Comparison of rupture life and creep elongation of new, service-exposed and HIP rejuvenated blades showing influence of HIP cycle and improvements of properties achieved with cycle #3 [13,31]

6.4 Re-contouring of Compressor Airfoils

Processes such as Sermatech's RD-305 or Lufthansa/University of Aachen's ARP can be used to recover the loss in aerodynamic performance of eroded compressor airfoils. With these processes, the leading edge of eroded airfoils is re-machined by grinding to a new optimized contour geometry designed to meet the flow condition characteristics of the original blading. The design criteria are chord length, profile thickness and leading edge angle. Re-contouring is limited to blades for which the chord length is greater that the OEM specified minimum allowable, and therefore the components are deemed reusable. Re-contouring translates into increased compressor efficiency, reduced fuel consumption, extended on-wing times and reduced spare parts costs [34].

7. LIFE EXTENSION THROUGH MATERIALS MODIFICATIONS

The options available to enhance component durability through materials modifications include:

- Substituting a component material for another, such as replacing a blade alloy from a conventionally cast form to a DS product (CG or SC) to increase the component creep strength or its thermal fatigue (TF) resistance;

- Replacing or adding a protective coating to improve resistance to wear, fretting, erosion, oxidation or corrosion , such as hard coatings to protect compressor airfoils against erosion or a TBC for hot parts; and
- Applying a surface modification treatment such as shot peening, ion implantation or laser surface treatments to improve resistance to various modes of surface degradation or fatigue.

7.1 Retrofitting with New Materials

The decision to change a component material is normally made during a component improvement program led by the engine manufacturer but supported by different user services. There are examples of material changes involving both blades and discs. For CF F404 engines, the first stage HPT blade material was changed from CC René 125 to DS-CG René 80 and, recently, a change to SC alloy N4 has been approved. This was done to improve the creep strength and thermal fatigue resistance of the blades. However, with changes of this type, the expected improvements are not always met, because a different mode of damage that was not anticipated may prove to be life limiting. Disc alloys are more rarely changed. Any change would be made to correct a durability shortcoming. For example, one of the CF J85 CAN40/15 engine compressor disc was changed from AM355 martensitic stainless steel Fig. 9(b) to a DA718 nickel base superalloy to eliminate premature bolt hole cracking.

7.2 Replacing or Adding a Protective Coating

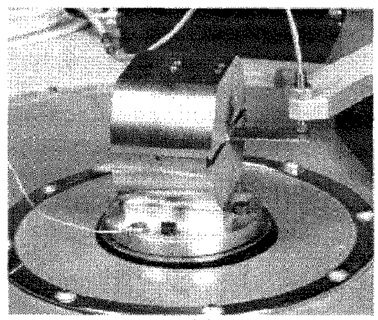
Both the substitution and the addition of coatings provide opportunities for life extension of cold end as well as hot end components. The qualification testing would typically involve documentation of the physical and mechanical properties of the new coating as well as evaluation of relevant durability and damage tolerance properties. More importantly, coating-substrate interactions and their impact on component properties would have to be carefully assessed.

7.2.1 Hard Coatings for Compressor Airfoils

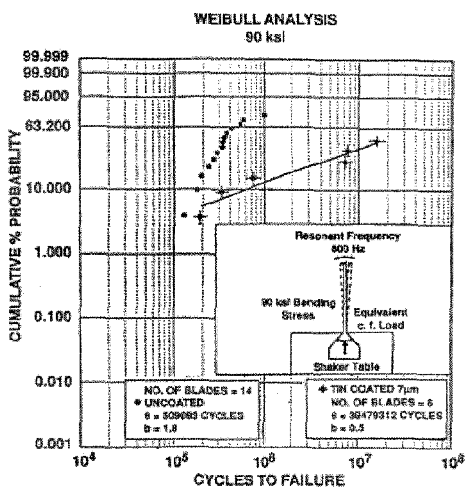
Titanium nitride (TiN) has been qualified as a coating for the protection of compressor airfoils against damage due to erosion and corrosion. In particular, the RIC™ PVD TiN coating from Liburdi Engineering is a bill-of material option for T56/K501 RR(Allison) engines [10]. Titanium nitride is a hard ceramic compound, which provides excellent protection against erosion and some corrosion protection, although pitting may be a problem with some commercial TiN coatings. The latest generation of coatings are designed to improve both erosion and corrosion resistance. These new coatings are based on alloying TiN with carbon or aluminum (TiCN, TiAlN). Coatings based on chromium nitride are also being considered for corrosion protection [5].

Substantiation testing for compressor coatings will typically involve measurements of coating thickness, uniformity and surface finish, assessment of coating durability (adhesion, hardness, ductility and toughness, erosion resistance, corrosion resistance). The effects of coating on substrate microstructure and mechanical properties (strength, fatigue), on vibration characteristics and fatigue properties of coated components, and on compressor efficiency and engine performance would also be considered [5,31]. A sand ingestion test may be conducted inside a test cell. However, because such testing may not accurately simulate real operating conditions, a field evaluation is usually preferred.

Coatings can significantly alter the natural frequencies of compressor airfoils. The change may be sufficient to cause the airfoils to be excited at resonant frequencies during service and such vibration may lead to high-cycle fatigue (HCF) failure. The effect of coatings on the fatigue life of the component is another important consideration of the qualification process. Substantiation test hardware developed at IAR to assess the effects of coating on fatigue properties of T56 6th stage compressor blades is shown in Fig. 21(a) [31].



(a)



(b)

Figure 21. (a) Apparatus for measuring natural frequencies of fatigue properties in first bending mode vibration of 6th stage T56 compressor blades; (b) Weibull distribution plot of fatigue life of the coated and uncoated blades measured with the apparatus [5].

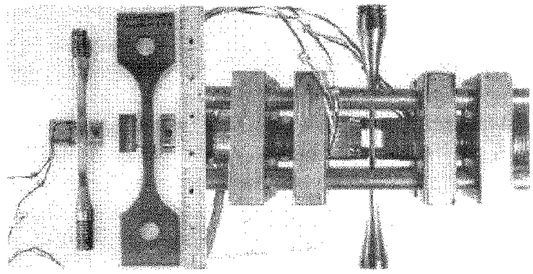
The blade is attached by its dovetail to a fixture mounted on an electromagnetic shaker and made to vibrate at its natural frequency in the first bending mode, until failure occurs. The number of cycles to failure of coated blades is compared with that for bare blades at identical root stress levels. Weibull analysis of the fatigue test results for RIC coated blades indicates that the cumulative probability of failure for the coated blades is significantly smaller at 90 ksi than for the uncoated blades, Fig. 21(b) [31]. The S/N data obtained by rotating bending fatigue also indicate that RIC coated test coupons of the blade material (17-4PH stainless steel) have fatigue lives comparable or better than those of bare coupons [5]. The increase in fatigue life may be due to changes in the blade microstructure induced by the coating process or to the residual compressive stresses induced by the coating on the surface of the blades.

7.2.2 Coatings for Turbine Hot Section Components

It is common practice to re-coat turbine blades and vanes at overhaul. This provides an opportunity to change the coating for one that is better suited for the operating environment. For instance, a Pt-aluminide will provide some durability enhancement over conventional aluminides where hot corrosion is found to be life-limiting. Also, the addition of a thermal barrier coating (TBC) to some critical areas of an airfoil, such as the tip region of turbine blades or leading edge of an NGV, will lower component metal temperature and minimize the rate of material consumption due to oxidation at these locations [2]. TBCs will also reduce the sensitivity of these components to thermal fatigue cracking by decreasing the transient stresses induced by thermal cycling during service [2].

Coatings can also be added to components that have been designed to operate without a protective coating. This is the case for the F404 turbine nozzle. Rapid oxidation of the airfoil surface of this component gives rise to loss of material through spallation and thermal fatigue cracking near cooling holes. OAC has developed a braze repair for the cracks and a repair-compatible coating for the airfoil that has been shown to greatly enhance component durability in accelerated burner rig tests [2,35].

Substantiation testing to qualify new protective coatings for turbine hot gas path components will typically involve a metallographic evaluation of coating microstructure to evaluate quality of the coating and its impact on microstructure and phase stability of the substrate material. An assessment of the effects of coating on the mechanical properties of coated material test coupons (Tensile and creep strength, HCF LCF, thermal mechanical fatigue (TMF), ductile-brittle transition temperature (DBTT), etc.) is another important requirement. Rig tests are required to ensure that the flow rates across cooling passages of internally cooled parts are identical for original and modified parts. Rig tests are also required to compare the durability and damage tolerance of original and modified parts, under simulated service conditions. Engine testing as part of an accelerated mission test (AMT) will complete the qualification. The IAR burner rig is used to assess the durability of modified components as well as to screen candidate coatings applied to test coupons in the form of pins or plates, depending on the type of coating and intended applications [31].



(a) (b)

Figure 22. (a) Tensile coupons and (b) IAR fretting fatigue test rig for assessment of fretting fatigue resistance of fan engine materials [31].

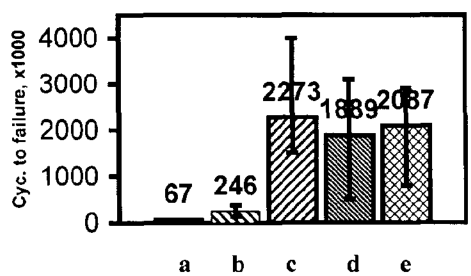


Figure 23. The fretting fatigue life of Ti-6Al-4V: a = base metal; b = CuNiIn + MoS_2; c = shot peened; d = shot peened + CuNiIn + MoS_2; e = shot peened + MoS_2.

7.3 Surface Modification Treatments

Surface treatments such as ion implantation and chemical surface treatments have been explored for use in conjunction with shot peening and soft coatings to alleviate the fretting fatigue problems in the dovetails areas of titanium alloy fan and compressor blades. Shot peening in combination with chemical treatments has been found to be quite effective in laboratory tests but other surface treatments, including soft coatings and lubricants, appear to also have the potential to improve fretting fatigue resistance of titanium alloys. The effects of these treatments on fretting fatigue life of two titanium alloys (Ti64 and Ti17) have been evaluated using an apparatus developed at IAR, Fig. 22. Some of the results of this work are presented in Fig. 23 (36,37).

8. LIFE EXTENSION OF SAFETY-CRITITCAL PARTS THROUGH IMPLEMENTATION OF NEW LCM METHODOLOGIES

The main factors considered in the choice of materials for compressor and turbine discs in engines developed in the 50's and 60's were tensile properties in the bore and the rim of the discs and creep properties in the rim of turbine discs. This was done essentially to provide an overspeed margin without disc burst. For engines of that generation, fatigue lives were not provided for safety-critical parts [38]. On occasion, LCF lives of turbine rotors have been established by OEMs at a late stage in the life of an aging engine, usually under a component improvement program (CIP).

For a number of such undertakings, the actual service lives of a significant fraction of components from lead-the-fleet engines have been found to be greater than the calculated LCF safe lives. This reinforces the view that components retired at their design safe-life limits may have significant fractions of usable life remaining. This is the case for an engine operated by the Canadian Forces. In another situation affecting the Canadian Forces, the engine was old enough that replacement discs and impellers were not available. In situations of this type, the implementation of a damage tolerance approach can assist fleet managers minimize risk if they must keep the engines in service.

Table 1 CF engine components targeted for life management based on RFC concepts and status of the analyses.

Engine	Component	Status
Nene X	Impeller	Complete
	Turbine disc	Complete
	Turbine shaft	In progress
J85	Compressor Disc	Complete
	Turbine disc assembly	In progress
T56	Turbine spacer	In progress

Implementation of a damage tolerance based LCM approach for safety critical parts is best undertaken by the OEM, for instance under CIP sponsorship. However, there may be situations where this is not possible, either because CIP sponsorship is not an option or the OEM is not prepared to undertake the required work.. In this case, there may be no alternative but for the user to undertake the implementation.

In Canada, components from three CF engines have been targeted for implementation of a damage tolerance/RFC based LCM approach. The components are from Nene X, T56 and J85 engines and they are identified in Table 1, along with the status of the undertakings.

Implementation of a damage tolerance based LCM methodology for aging components requires information on (i) initial crack length, (ii) changes in stress intensity factor (SIF) with crack extension, (iii) fatigue crack growth rates (FCGR) at the fracture critical location in the component, (iv) dysfunction crack size (DCS) and (v) cyclic and/or steady state service usage of the engine. This information is needed to predict a fracture mechanics based safe inspection interval under typical service conditions. There are seven steps that would normally be followed to obtain or generate the required information and to conduct and validate the analyses.

The seven steps include:

- Determination of stress and temperature data for the component of interest;
- Identification of the fracture critical location in the component;
- Determination of the stress intensity factor (SIF) of cracks at the fracture critical location and its dependence on crack size and estimation of the dysfunction crack size (DCS);
- Generation of fracture mechanics data for safe inspection interval (SII) calculations
- Generation of POD data for the NDI technique used at depot level to inspect the component (to provide estimate for initial crack size);
- Calculations of the safe inspection interval (SII); and
- Full scale testing in a spin rig to validate the SIIs.

Examples taken from CF experience for each of the steps are presented below.

Step 1: Determination of Engine Stress and Temperature Data

Mechanical stresses can be calculated from rotational speed, geometry and mass of the component. However, predicting thermal stresses requires detailed information on temperature transients and temperature maxima, which are more difficult to obtain. This information is strongly influenced by service usage and may not be available from the engine manufacturer. For the J85 turbine disc assembly, work is underway at IAR to generate the information under the auspices of a TTCP project. A turbine assembly has been instrumented to determine the temperature distribution across the components. An engine has also been equipped with a slip ring to transmit the information collected from sensors to instrumentation located outside the engine. A photograph of the modified engine is shown in Fig. 24. The results of this study will be used to set boundary conditions in FEM models to establish temperature and stress distribution within the turbine assembly.

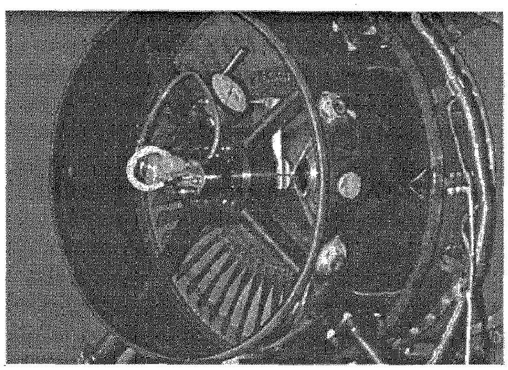


Figure 24. Back end of J85 CAN40 engine with slip rig installed to transmit sensor information to an outside data acquisition system.

Step 2: Identification of the Fracture Critical Location

This is obtained by analysis using the finite element method (FEM). For Nene X components, the thermal-mechanical loading information required as boundary conditions was discussed with the engine manufacturer. Both 2-D and 3-D analyses were performed for the turbine disc. The bottom serration of the firtree slots in the disc was identified as the fracture critical location because of a combination of high thermal and mechanical stresses operative at this location [39]. The distribution in von Misses stress between two of the disc slots is shown in Fig. 25 for the 2-D analysis case.

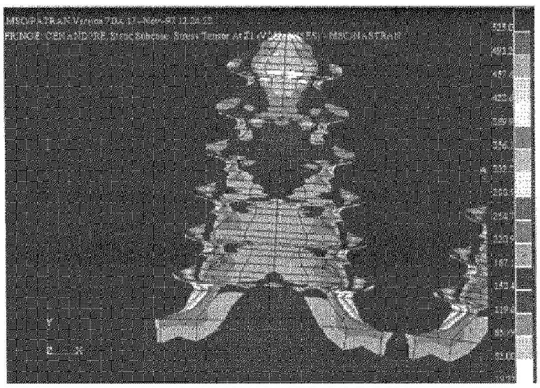


Figure 25. Details of the 2D stress distribution for the rim region of the Nene X turbine disc in an uncracked condition

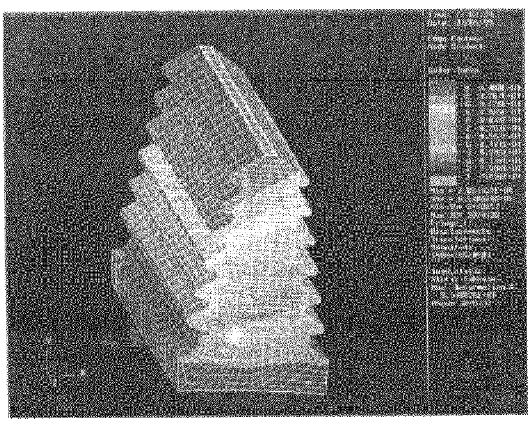


Figure 26. Details of the 3D strain distribution for the rim region of the disc with a modeled crack in the bottom serration of the firtree slot, which is the primary fracture critical location in the disc [39].

Step 3: Fracture Mechanics Analysis for SIF Calculations

This is also obtained analytically using the FEM. For the case of the Nene X turbine disc, a crack was assumed to be present in the bottom serration region of the firtree slot, as shown in Fig. 26. The fracture mechanics analysis was performed for cracks with surface length to depth ratios (2c/a) of 3:1 and 4.5:1. A 2c/a ratio of 3:1 is often observed in aero engine discs. A 2c/a ratio of 4:1 is considered to represent a worst case situation, whereas a 2c/a ratio of 4.5:1 represents a multiple crack situation [39]. Variation of the stress intensity factor (SIF) with crack depth for a crack with a 2c/a ratio of 4.5:1 is shown in Fig. 27 [40]. For a crack 8 mm deep, the SIF value ahead of the crack tip is 64 MPA $m^{1/2}$, which is a fraction of the fracture toughness (K_{1c}) of the material. That crack depth magnitude of 8 mm was used in all subsequent calculations to arrive at the SII for this component.

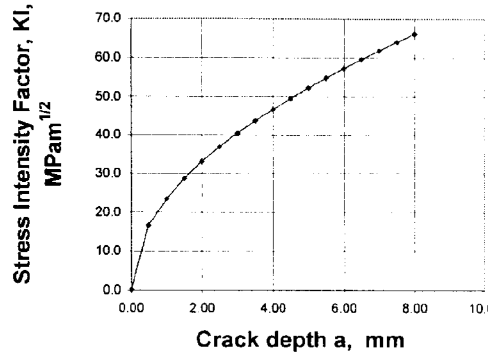


Figure 27. Stress intensity factor (SIF), as a function of crack depth for a crack with a 2c/a ratio of 4.5:1 located in the bottom serration of the firtree slot in the Nene X disc [40].

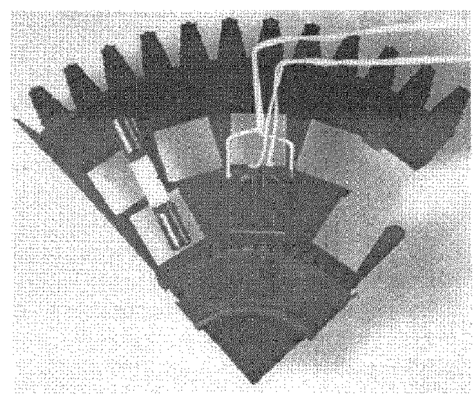


Figure 28. Tensile and compact tension specimens machined from the Nene X disc. The eletrical leads are part of the automated DC-PD technique used to monitor crack growth during testing.

Step 4: Generation of Fracture Mechanics Data

The fatigue crack growth rate (FCGR) data versus stress intensity range, ΔK, is preferably generated using compact tension specimens machined from a high time component as close as possible to the fracture-critical location, Fig. 28. A high time component is selected to account for any effects of service-induced microstructural degradation on crack growth. The test temperature is chosen on the basis of service temperature estimates for the fracture critical location. The specimen geometry should conform to ASTM E-647 specifications to ensure that plane strain conditions prevail ahead of the crack tip during testing. The FCGR data reproduced in Fig. 29 reveals that material machined from a high time IN 718 disc has significantly higher FCGRs than material machined from a new or a medium time disc (41). Such variations in FCGR data would significantly influence SII calculations. The worst case scenario would normally be adopted for the calculations [42,43].

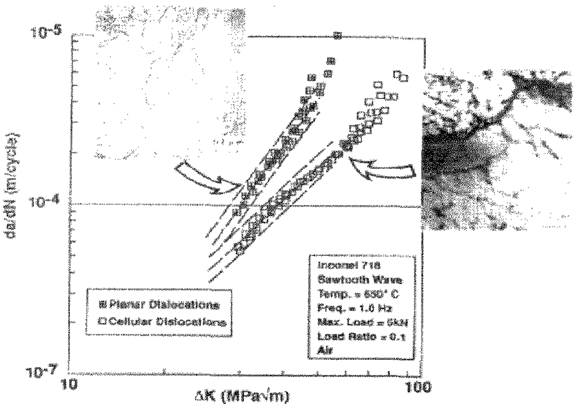


Figure 29. Comparison of fatigure crack growth rates (FCGR) in new and service exposed alloy 718 turbine discs; high time disc shows significantly higher FCGR as compared to new and medium time disc [41].

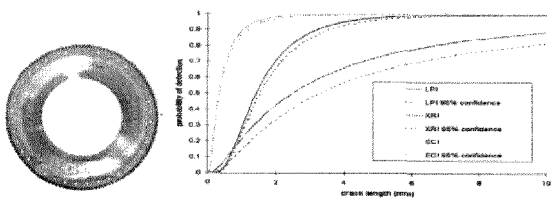


Figure 30. Probability of detection of natural cracks in Fe-Ni-Cr alloy turbine discs using liquid penetrant inspection (LPI), eddy current inspection (ECI) and x-ray inspection (XRI) [44,45].

Step 5: Generation of POD Data for the NDI Method of Interest

The size of cracks likely to be missed during component inspection needs to be estimated to predict SIIs. Ideally, enough data should be generated to obtain probability of detection (POD) curves specifically for the component of interest and the method used for its inspection. This may not always be possible. The initial crack size (a_i in Fig. 11) for components targeted by the CF have been selected from a demonstration program carried out in Canada on a series of J85 life expired compressor discs containing natural fatigue cracks originating from bolt holes [16]. For this program, several discs were inspected by a variety of NDI methods and by different inspectors. The NDI results were verified by prying open each bolt hole inspected and examining the fracture surface for evidence of service induced cracking. Actual flaw sizes were established from the microscopic examinations. Finally, POD data for each of the NDI methods considered was generated through standard POD analysis. This program was first introduced to the AGARD community in 1988 as part of a Workshop on Non Destructive Evaluation [44]. A follow-up AGARD sponsored activity broadened the scope of the study for the benefit of the NATO community [45]. The substantial data on POD of NDI methods generated through this effort are available from the USAF supported Nondestructive Testing Information Analysis Center (NTIAC) in Austin, Texas. The POD curves for three common inspection techniques, including liquid penetrant inspection (LPI), are compared in Fig. 30.

For the Nene X turbine disc, the initial flaw size was assumed to correspond to either the maximum crack length missed (a_{max}) or the crack length value at 90% POD with 95% confidence (90/95 POD) obtained for LPI under the J85 disc program. This is justified since LPI is used to inspect Nene X components and the two discs are made of stainless steel [40].

Step 6: Calculation of SII

The SII is set as a fraction of the time it takes to grow a crack of assumed geometry from the initial crack size to the dysfunction crack size. Either deterministic or probabilistic fracture mechanics (DFM or PFM) analyses can be used to compute the SII. A DFM analysis is based on worst case situations for all variables in the life calculations. Many have claimed DFM to be overly conservative and this explains the growing interest in PFM based analyses. A PFM analysis aims primarily to simulate the consequences of missing a crack during inspection (42,43). It also helps account for scatter in FCGR data.

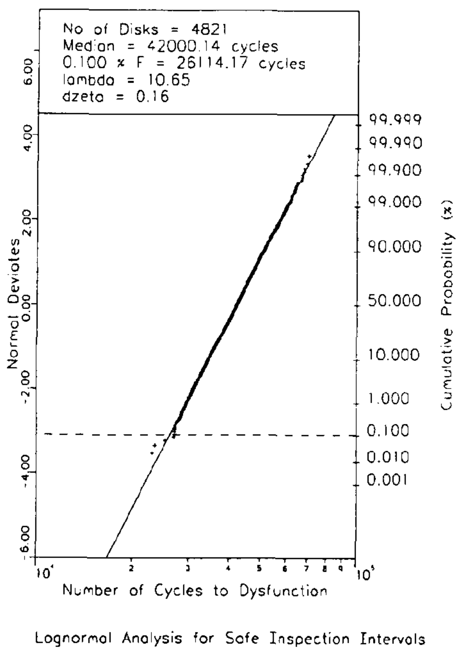


Figure 31. Lognormal analysis of PFM generated data for Nene X turbine discs simulating the effect of worst possible scatter in FCGR data and uncertainties associated with the LPI technique [40].

A lognormal analysis of PFM data generated for the Nene X turbine disc, which simulates the effect of worst possible scatter in FCGR and uncertainties associated with the LPI technique is presented in Fig 31. The data reveal that the 0.1% probability of failure or 1 in 1000 chance that a missed crack will reach dysfunction size is of the order of 26,000 cycles. Assuming the worst case scenario of 5 cycles per hour of engine usage, the 0.1% probability of failure translates into 5000 hours. Since the engine is currently inspected every 1800 hours, it is not surprising that no components have ever failed catastrophically in service. Interpretation of PFM analysis simulating experimentally observed scatter in FCGR data indicates that the overall safety factor inherent in the predicted SII for the fleet size is 4.7 [40].

Final decisions about setting SIIs for the Nene X disc were made on the basis of a DFM analysis. A worst case crack propagation curve (depth versus cycles) for a crack 1 mm deep originating from the fracture critical location and growing to a DSC depth of 8 mm is shown in **Fig. 32**. The data clearly demonstrate the importance of the sensitivity of the NDI technique on the SII. For instance, if the initial crack depth is 1 mm the crack propagation interval (CPI) is about 30,000 cycle, whereas it reduces to about 8,000 cycles if the detectable crack depth is only is 4 mm. The SII recommended for the Nene X disc is 1/3 of the CPI, or 10,000 cycles. For the worst case scenario of 5 cycles per hour of engine usage, the DFM based SII expressed in flying hours is 2000 hours. Therefore, there is high probability that any crack would be detected during routine overhaul inspection, since the routine inspection for this engine is conducted every 1800 hours [40].

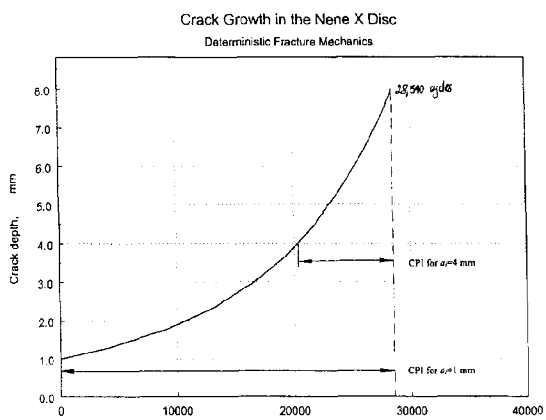


Figure 32. Damage tolerance based life cycle management curve for Nene X turbine discs using 90/95 POD value of the LPI tehnique as the detection limit of the technique [40]

Step 7: Validation of SIIs Through Spin Rig Testing
Validation of the SII calculations requires spin rig testing of components under conditions that simulate service. Testing of J85 and Nene X components is underway at IAR to validate component lives and safe inspection interval calculations using the spin rig facility shown in **Fig 33.** To date 9,000 mission cycles have been accumulated on the J85 turbine disc assembly at a test temperature of 450°C. Both LPI and eddy current inspection techniques are being used to inspect the components as the cycles accumulate.

All above predictions are based on the assumption that cyclic usage was the primary driver for crack initiation and growth processes in the Nene X disc. This assumption was made on the basis of discussions with the OEM, from which it was also concluded that time dependent creep mechanisms did not contribute to the crack growth process to any significant degree during service. Based on past experience with the fatigue response of turbine disc materials, it was also assumed that LEFM could be used to describe crack growth within the component. Had creep contributed to the accumulation of damage in any significant way, creep-fatigue interactions would have had to be considered.

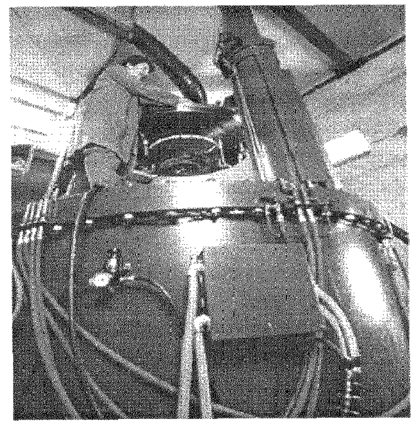


Figure 33. J85 turbine disc assemble being readied for testing in IAR spin rig.

CONCLUDING REMARKS
Aging in engines is a process involving gradual deterioration of components, which begins when an engine enters service. This aging process cannot be avoided and must be managed. Managing the aging of engine components to extract maximum usage of expensive to replace components requires a good understanding of deterioration modes and their effects on engine performance, reliability and safety. From an operator's perspective, a number of options are available to extend component lives beyond book limits. The options include taking advantage of technology developments to repair damaged components or enhance their durability through material modifications, including the substitution or addition of protective coatings. The implementation of a damage tolerance based life cycle management methodology for safety-critical parts may also be a practical option in some situations. The Canadian experience suggests that significant cost savings may accrue from such undertakings [21].

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