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RECENT DEVELOPMENTS IN DAMAGE TOLERANCE ANALYSIS FOR HELICOPTERS

by

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1.0 INTRODUCTION

This paper is written to present recent accomplishments of Georgia Tech and Sikorsky Aircraft in the area of damage tolerance analysis (DTA) for helicopter applications. Over the past several years, both organizations have supported structural integrity analysis for U.S. Air Force (USAF) helicopters. It has been found that while rotorcraft fuselage structural integrity is similar to that for fixed wing aircraft as defined in MIL-STD-1530A (Reference 1), many issues are peculiar to rotorcraft applications particularly with regard to dynamic components. This makes the effort much more complicated.

The U.S. Army Aviation Applied Technology Directorate began to formulate plans for an official Helicopter Structural Integrity Program (HSIP) Document known as MIL-STD-XXXX in the mid 1980s (Reference 2). However, due to funding constraints, the program was not carried out in an overall effort, but instead focused on requirements for specific emerging technologies such as safe-life reliability, flight recorders, and damage tolerance analysis of composites. A structural integrity program plan is still a requirement imposed on all new U.S. Army helicopter systems, although not regulated by a specific military standard. The RAH-66 Comanche program incorporates an HSIP which generally follows the guidelines of MIL-STD-1530A.

Recently, a long term engineering effort to aid in force management concepts by rapidly and accurately defining inspection intervals and retirement times for aircraft components has been undertaken by the USAF. It attempts to combine related efforts of sister service organizations to improve and streamline logistical support of their rotorcraft fleets. Based on lessons learned from prior contractual efforts, this paper reflects the views of the Georgia Tech Research Institute (GTRI) for a proposed solution to provide an effective additional logistical tool for structural integrity. The efforts of the USAF's Warner Robins Air Logistics Center (WRALC) are duly recognized for their innovative initiative for damage tolerance of rotorcraft.

2.0 WRALC PROGRAM GOALS

Successful mature weapons systems employed by the military services are always called upon to fly faster, higher, stealthier, for longer ranges, with greater maneuverability, and usually with heavier payloads. Very often, missions involve frequent low level nap-of-the-earth or terrain following flight, even in adverse weather. In order to ensure the structural integrity of aging weapons systems under their ever-increasing mission severity, the USAF has spearheaded a developmental integrity and force management concept applied to helicopters based on the success of their fixed wing counterparts. The specific goals of the WRALC program are to:

a) enhance aircraft safety,
b) improve maintenance procedures,
c) increase mission capability, and
d) lower life cycle cost.

Traditional force management of helicopters is implemented by safe-life component removal and replacement, based upon accumulated fatigue damage. Rotating helicopter components, their fixed control system counterparts, as well as engine components, are subject to this approach. Some parts, particularly those that are restricted to depot repair, could be more cost effective to remove and replace in conjunction with other depot activities since the part may be readily accessible with available depot tooling.

The target helicopter for validating this program is the Sikorsky H-53, originally fielded in the mid 1960s. Damage tolerance criteria was never specified for the H-53 or any aircraft of that vintage. Trying to solely apply damage tolerance based fleet management in accordance with design criteria outlined MIL-A-87221 (Reference 3) to such vehicles does not appear practical. Some hybrid form of damage tolerance and safe-life procedures must therefore likely be adopted. In general, dynamic components are particularly difficult to manage using damage tolerance because of relatively rapid crack growth, and may therefore be better suited to safe-life or a hybrid program. The technology and analysis requirements of the Engine Structural Integrity Program are considered to be the most appropriate guideline for future dynamic component design (Reference 4).

The MIL-A-87221 damage tolerance force management approach is applied by non-destructive inspection (NDI) of a component at an analytically or empirically determined interval. Reworking or modifying the component to enhance the fatigue life is possible until an eventual part safety limit is reached. The determination of crack growth curves and confidence in them is driven by the crack growth modeling technology and material data bases available.

Sikorsky Aircraft performed initial damage tolerance studies on eighteen of the most fatigue sensitive H-53 parts, primarily from the rotor systems, investigating DTA techniques and performing extensive component finite element modeling. This investigation also defined the need for improved accuracy and reduced conservatism in usage data, stress intensity solutions, crack models, and materials data for small cracks. The results showed that reliable NDI for 0.010 to 0.030 inch cracks would be required before considering damage tolerance management of these sensitive parts.

Under a separate WRALC contract, Georgia Tech analyzed eight H-53 components from the rotor and airframe systems. Many aspects of the Georgia Tech analysis were different, but the bottom line results were the same. None of the parts analyzed had sufficient damage tolerance capability to depart from their recommended safe-life qualification. Analysis showed that crack growth thresholds are exceeded even for small flaws (.005 ins), especially in the rotor systems.

With recommended improvements as cited in the Sikorsky and Georgia Tech DTA studies, force management for helicopters based on damage tolerance analysis may be possible in a significant amount of helicopter structures. As noted, it is necessary to develop more realistic stress spectra, crack models, and material data (Reference 5). This can be achieved by using a combination of flight tests, ground tests, and analytical/computational procedures.

2.1 IMPLEMENTATION ISSUES

Figure 1 presents the overall framework for implementing an HSIP as postulated by WRALC. It incorporates existing software and procedures used for fixed wing aircraft logistics, with additional consideration of the USAF growing helicopter fleet. Individual aircraft usage and NDI results are assembled at the squadron level and submitted to WR-ALC using standard equipment. It is then incorporated into the Aircraft Information Retrieval System (AIRS) program which provides information by tail number on locations, hours, mission profiles, parts tracking, etc. The data is fed into the structural integrity computer program to perform fatigue analyses. The HSIP manager works with government engineering personnel and contractors to make force management decisions.

The USAF program for helicopter structural integrity also includes technology developments such as cycle counted flight loads data, full scale strain survey data, improved crack growth models, and usage data based on the Individual Helicopter Tracking Program (IHTP). The IHTP is an essential part of the proposed HSIP layout and is similar to existing IATPs (Individual Aircraft Tracking Programs) for fixed wing aircraft. The purpose of IHTP is to obtain usage data representing accurate spectra for the fleet, squadron, and individual helicopters. This option replaces the current assumption of only one universal average usage spectrum for the entire weapons system fleet.

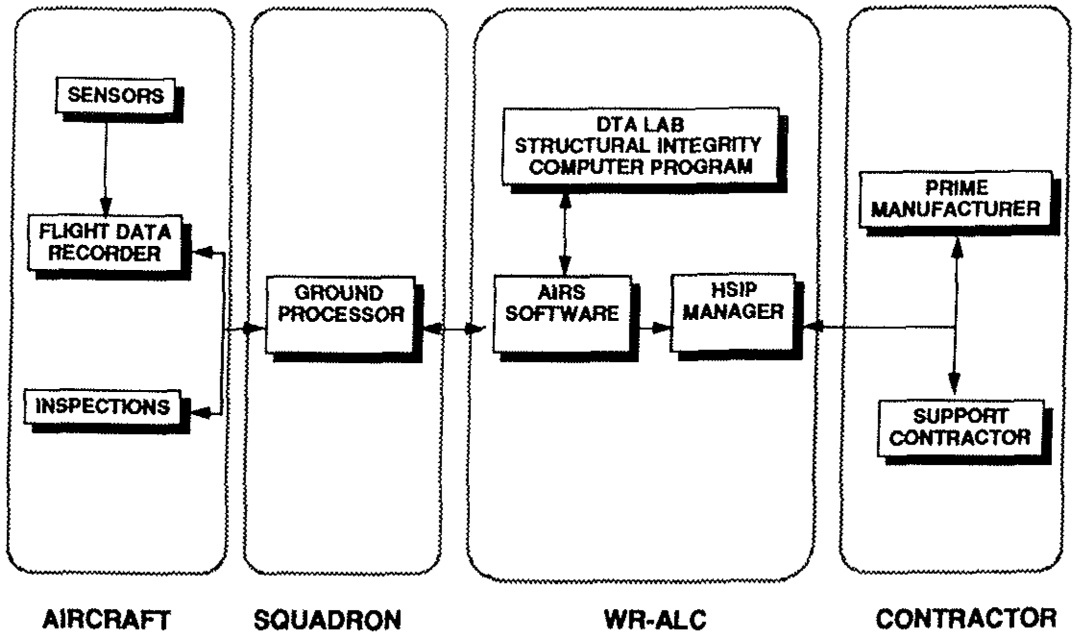

Figure 1. Proposed WR-ALC Helicopter Structural Integrity Program Framework.

The primary IHTP function tracks the occurences of specific flight and power-on ground manuevers. This differs from current fielded IATP systems where a relatively small number of parameters such as CG, altitude, airspeed, load factor, etc., are recorded in time history fashion and analyzed after the missions are flown. Time history recording is infeasible for a lightweight durable helicopter monitoring system because helicopters are capable of more maneuvers than fixed wing aircraft, and more parameters are necessary to describe any rotorcraft maneuver state. Instead, occurances of mission maneuvers are recorded in a histogram fashion requiring much less computer hardware capacity. This is based upon real time processing of the data using automatic regime recognition algorithms. Recent advances in regime recognition technology has made this feasible for rotorcraft.

On 5 to 10 percent of the helicopter fleet, the IHTP system is used to perform Loads and Environment Spectra Surveys (L/ESS). These aircraft are equipped with additional instrumentation to record parameters used for structural substantiation. Signals from strain gages on selected critical components are processed in real time to record cyclic information in histograms. These are checked periodically to ensure that aircraft are not being flown outside of their recommended envelopes and to check that minor aircraft modificaitons have not unexpectedly caused a major change in loads. The IHTP system could also be adapted to crash sequence monitoring if desired.

Obtaining actual usage data on an aircraft by aircraft basis will reduce the conservatism and error associated with a single assumed spectrum for the entire fleet. Continuous usage monitoring will provide the structural integrity computer program with up to date information to resubstantiate component retirement times and inspection intervals.

Several NDI methods are in widespread use today, and newer improved techniques are constantly being developed. These range from visual inspections which can reliably detect flaws on the order of .5 inches to laboratory based automated eddy current techniques with a capability of around .005 inches. This impact of crack detection capability upon force management decisions must be considered for each candidate part. It may happen that the flight hours available are too short for credible application of damage tolerance techniques to force management. Reducing the crack detection threshold significantly below current field capability may not be cost-effective. For a small fleet, it may be less costly to replace parts based on safe life or alternate damage tolerance evaluation than to improve reliable NDI crack detection levels. Removing parts for retrograde to depot level (or laboratory) NDI equipment is an option that must be weighed against cost, operational impacts, and the ability to track individual parts.

HSIP methodology must also consider the unique aspects of mature aircraft systems in order to manage them effectively. The degradation of structural material is not considered in standard analytical fatigue practice for new or mature aircraft, but obviously can induce fatigue through stress concentrations and microcracks. Because fewer new military aircraft systems will be developed and procured by the military services, high demands are placed on mature systems to continue operations, often beyond their design lifetime.

Because of these political and economic circumstances, recent emphasis has rightfully been placed on aging aircraft research. In December of 1992, the USAF and the FAA signed a Memorandum of Understanding which defines requirements and coordinates their mutual interests in maintaining aging aircraft. Common areas of research include: probabilistic approaches for widespread damage, health monitoring systems, improved flight loads surveys, corrosion prevention, composite repair patches, and emphasis on small commuter aircraft. A review of existing capabilities showed that much work is still needed in NDI and composite repair technology. Other advancements are desired in fracture mechanics for small cracks, crack growth in corroded structure, multi-site damage growth prediction, NDI of thick composite structures, and corrosion detection.

As a result, the Air Force Office of Scientific Research recently launched a large multidisciplinary basic research effort to investigate topics relative to the aging fleet. In April of 1993, a kickoff meeting and workshop on aging aircraft research was held at Georgia Tech with participants from academia and the government. Most academic participants are from universities with specialties in mechanics, chemistry, and physics. Scientific advances were presented which could fill the needs of the military and FAA for ensuring airworthiness.

2.2 STRUCTURAL INTEGRITY COMPUTER PROGRAM

A key element of the WRALC effort is a structural integrity computer program which will perform comprehensive fatigue analysis of airframe and rotor system components to determine inspection intervals and component retirement times on an ongoing basis. Ultimately, the computer program will serve as an essential module in the logistical support system of Figure 1. This will provide the user with the capability to rapidly determine fleet management decisions under changing weapon system mission requirements.

The U.S. Army's pursuit of HSIP is described in Reference 2. Defined is the need for a tool that will aid designers and logistics agencies in deciding which force management technique to apply: damage tolerance, safe-life (crack initiation), or a combination of both (total life). The purpose of the computer program is to provide a flexible total life tool, validated by selected full scale component testing, and able to examine potential usage, configuration, and parametric impacts on helicopters and aircraft structural integrity. Once in place, this tool can be used in any helicopter application, and even for fixed-wing aircraft, because the basic architecture and elements of the processor are the same for all applications, and only require data bases specific to a particular weapons system. The development of the computer program will use the H-53 as its subject helicopter to validate the concept because of the expertise accumulated by the contractors for this aircraft in past efforts.

Figure 2 illustrates how the computer program will function. All tasks are brought together graphically to provide a clear picture of their interactions. There are six basic steps to the processor, as labeled by the shaded boxes. First, usage spectrum generation is either based on a generalized spectrum for the entire fleet, or tailored for an individual aircraft through the use of an IHTP. For the selected components, accurate stress spectra will be developed by utilizing cycle counted flight test measurements and using state of the art stress analysis techniques. To permit the calculation of stress spectra for mission segments that differ from those covered in H-53 flight tests, a rotor/fuselage model will be generated for use in a validated rotor loads program. The loads effort not covered in this paper, but is discussed in detail in Reference 6. Validated stress intensity factors, including those for threaded parts, will be used for crack growth. Safe life and crack growth analyses will consider two-dimensional cracks, closure effects, mixed mode crack growth, sensitivity to applied stress levels, geometric effects, and variable amplitude loading. Crack growth analysis results for components will be validated by correlation with full scale testing results. The results will be used to determine appropriate inspection intervals and component replacement times.

The remainder of this paper discusses recent developments in these areas being pursued by Georgia Tech and Sikorsky Aircraft to supply the above described program with the appropriate technology modules and data bases. The long term objective is to demonstrate a total life fleet management program for helicopter structural components.

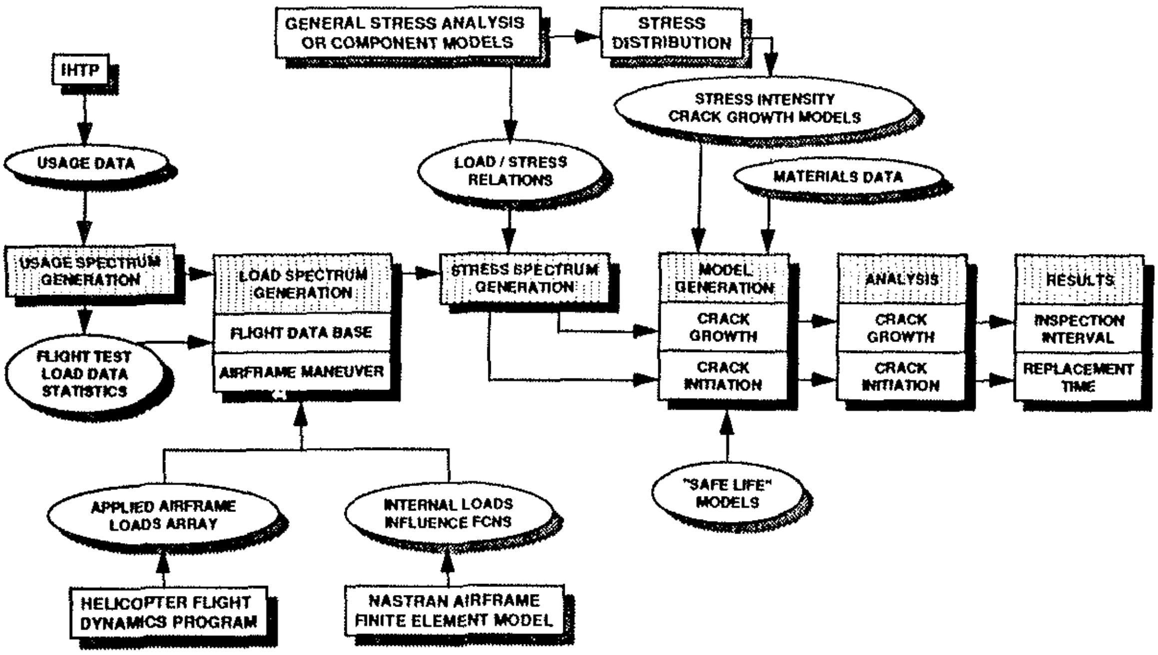

Figure 2. Structural Integrity Computer Program Architecture.

3.0 HELICOPTER FATIGUE SENSITIVE PARTS SCREENING

A comprehensive list of fatigue sensitive parts is desired to determine the most important components to include in the structural integrity computer program. In fixed wing aircraft, such parts include chordwise and spanwise wing splices, stringer-reinforced skin panels, and two bay wing panels. For helicopters, these components may include the rotor hub, blade attachment hardware, pushrods, swashplates, and airframe structure, usually at folding areas and primary fittings. In addition to a list of components, the potential crack locations for each part, and a priority ranking are also necessary. For initial development of the structural integrity computer program, a cross section of failures expected from aging aircraft and rotorcraft should be used.

Previous assessments done for the C-130 and C-141 have typically yielded a list of several hundred crack sites. Critical assemblies like wing members and rotor head components contain many potential crack sites due to the repercussions of a failure and high stresses associated with primary load paths in these members. In the case of fixed wing aircraft, Reference 7 presents a method for ranking crack locations to identify the most severe for further analysis. GTRI has adapted this method to rotary wing aircraft. The priority ranking for helicopters is based on ten sets of criteria that include design margin of safety, flight safety criticality, corrosion susceptiblity, inspectability, damage susceptibility, component retirement time, and crack growth time. Fatigue issues like surface treatments, residual strength requirements, and crack arrest features are emphasized. Parts are given composite scores up to a maximum of 100 with the most critical components having the highest scores.

4.0 STRESS INTENSITY FACTOR SOLUTIONS

Georgia Tech and Sikorsky are investigating the application of the most appropriate stress intensity factors to the structural integrity computer program. In previous contractual efforts it was recognized that standard stress intensity factors available in the literature were not applicable to many of the surface crack geometries in rotor parts. GTRI has conducted a survey of H-53 fatigue test reports and field failure data to determine the variety of cracks found in rotorcraft structures. Most noteworthy is the presence of many finite length thick walled cylindrical load bearing members in the rotor system. These are typically found in components housing stack bearings to permit rotational degrees of freedom of the rotor blades. The cylinders are not loaded in torsion and often experience axial edge cracks. Three dimension surface and corner cracks with curved fronts have also been found in complexly loaded components like rotor hubs. Cracks in fillet radii and thread roots are often encountered in components like push rods, spindles, and shafts. Other rotorcraft cracks are also found in lugs, loaded and unloaded holes in plates, many times with multi-site damage, typical of their fixed wing counterparts. Bolt failures under tension, bending, and torsion are commonplace.

The most current stress intensity factor solutions are found in Murakami's handbook (Reference 8), published in 1987. The handbook is very comprehensive, covering stress intensity factors in plates and shells for single and multiple cracks. Solutions are given for various loading conditions including multiaxial stress states, loaded and unloaded holes, wedge forces, pressures, thermal stresses, and electromagnetic forces.

Georgia Tech is cataloging recent developments since the time of Murakami's handbook. Because over 2000 references were found for stress intensity papers and books, some ground-rules were established to assist in paring down the amount of material. First, only linear elastic solutions were investigated which is consistent with current DTA practices for determining USAF inspection times. This eliminated fully plastic and elastic plastic solutions using finite element methods as the dominant technique. Next, only isotropic materials were selected, which eliminated much of the composites research recently published. Finally, crack geometries which were not suitable for helicopter failures were eliminated. Several papers have been presented publishing solutions for bolts under tension and bending, kinked cracks, axially cracked cylinders, solids under torsional loading, unsymmetrically cracked three dimensional finite bodies, edge cracks in round bars, and cracks adjacent to elliptic or spherical inclusions. For the most pertinent information, see References 9 through 20.

Other recent work has concentrated on new methods to compute stress intensity factors. Available standard methods include finite element models, boundary elements, compounding methods, weight function methods, mapping, asymptotic approximations, alternating methods, integral transforms, and numerical integral solutions. An approximate stress intensity solution for surface cracks in a general nonuniform stress field was developed at the USAF Aeronautical Systems Center, Structures Division, Flight Systems Engineering by Grimsley and incorporated into the MODGRO damage tolerance analysis computer program (Reference 21). Since that time, Newman and Raju have published solutions appropriate to some rotor problems (Reference 22), Sikorsky has modified the Reference 21 solutions to remove some of its conservatism, and United Technologies Research Center (UTRC) has developed a very accurate surface integral analysis technique to determine stress intensity solutions for surface cracks. The UTRC solution depends on having a finite element model for the part. The newly developed methodology is being investigated to define a consistent approach for use in determining inspection intervals.

5.0 SHORT CRACK GROWTH MODELS AND DATA

5.1 MATERIAL DATABASE

From previous studies, Sikorsky and Georgia Tech found that for a large number of helicopter structures it is necessary to accurately predict crack propagation rates for small cracks (.005 to .020 inches). This is a direct result of the helicopter high cycle fatigue phenomenon, where most of the crack growth time is spent in short crack propagation. Values of crack growth rate (da/dN) in the near threshold region are extremely sensitive to slight changes in stress intensity factor range (ΔK), and must be thoroughly examined before being consistently used in a validated crack growth analysis tool. Although excellent efforts have been made to enhance knowledge of short crack behavior in several materials (References 23 through 26), the database is presently sparse and nonstandard. Many issues regarding specimen design, crack initiation technique, statistical material properties, and measurement techniques must be explored. Sikorsky and Georgia Tech are both conducting investigations into these issues through independent experimental test programs.

The objective of the Sikorsky program is to obtain crack propagation rate data for surface cracks between .005 and .020 inches in depth for materials used on the H-53 helicopter. Emphasis is placed on near threshold data at growth rates from 10^{-9} to 10^{-6} inches/cycle. Specimens are machined from actual components and include Ti-6Al-4V beta STOA forgings, Ti-6Al-4V alpha-beta forgings, 7075-T73 aluminum forgings, and 4340 steel plate (150-180 ksi ultimate). Specimens used are modified Larsen flat tensile specimens concave in two dimensions at the gage section. They are manufactured using electrostatic discharge machining (EDM) and chemically milled to remove residual stresses. Cracks are initiated at a stress ratio of -1.0 from a small EDM starter notch resulting from polishing away a larger notch to reduce notch size.

Much testing has been completed for stress ratios of .1 and .5 for all of the materials. Resulting ΔK's are near the long crack thresholds and growth rates generally range from 10^{-8} to 10^{-6} inches/cycle. Final data analysis will be completed in early 1994. Currently plans are being made to conduct tests under spectrum loading. Two standard helicopter rotor spectra, HELIX and FELIX, were investigated but were not found to have desirable overload ratio characteristics. The overload ratio is defined as the ratio of excursion load to large blocked cycle segments of the spectrum. HELIX is the most benign with overload ratios of only 1.1 to 1.15, while FELIX was similar with overload ratios of 1.4 to 1.6. In order to get the best spectrum effects, an airframe spectrum derived from actual rainflow cycle counted H-53 tail pylon fold hinge flight test data will be used which contains overload ratios of 2.0 to 2.4.

While the objectives of the Georgia Tech test program are the same, some of the methods and equipment are quite different. Primarily, the focus of the Tech project is to create naturally occurring short corner cracks instead of notch assisted surface cracks. Specimens with a square gage section have been designed to initiate natural cracks under bending rather than tension and are currently being evaluated to optimize surface preparation and reproducible initiation sites. Natural corner cracks have been successfully initiated on several aluminum specimens. Once the specimen design has been finalized and cracks initiated, crack growth testing will proceed under tension, monitoring the corner crack dimensions using acetate replica and optical microscopy techniques. Crack growth testing is planned at stress ratios of .1 to .5, with the program finishing in late 1994.

5.2 CRACK MODELS

Further complicating the accuracy of current crack growth methods is the effect of overloads, resulting in crack retardation and acceleration. References 27 and 28 examine current technology for short cracks and variable amplitude loading and emphasize the need for accurate models and material databases. The latest research indicates that mechanisms for fatigue crack nucleation, short crack growth, and long crack growth are different, and attempts to extrapolate back from long crack growth behavior to the earlier regimes is not believed possible.

Conventional crack models may not be satisfactory for the small cracks of interest in helicopter damage tolerance assessments.

Retardation models based on residual stress effects alone cannot describe the transient crack behavior observed indicating that improved models are needed to quantify the effects of variable amplitude loading. These models should include both residual stress effects and closure obstruction effects to describe fatigue under spectrum loading.

Mixed mode growth also occurs in many typical airframe and dynamic components, including uniaxially loaded surface cracks in plates near shoulders or risers and symmetric cracks in plates under biaxial loading. In the case of microcracks, mixed mode conditions result from non-homogeneous grain structure, causing the crack front to curve and branch many times before a single mode becomes dominant.

Recent studies at Georgia Tech have (Reference 29) emphasized short elastic and plastic stress fields using finite element analysis. Proposed is a two parameter representation for the stress and strain field around a short crack involving the stress intensity factor, K, and a higher order T stress (or related biaxiality ratio, B), as introduced by Rice (Reference 30). So far, this has been investigated for finite width center cracked and single edge cracked plates. Other geometries are also being examined. Proven is the fact that short crack growth behavior should not be normalized with respect to specimen width (a/w) but rather with respect to a microstructural parameter such as grain size.

Given this, it is shown that the K singularity breaks down for crack sizes smaller than 20 grain sizes (.04 inches) in typical metals, requiring the need for an additional parameter, such as T or B. Finite element analysis shows a proportionately sizeable plastic strip in the wake of short cracks which cannot be ignored, and further studies are underway to determine the effects of stress re-distribution due to plastic yielding. Because a single parameter description of the stress singularity is not considered valid for short cracks, representing material da/dN data solely in terms of ΔK may also be inappropriate. Coordinated efforts with the above mentioned Georgia Tech short crack material test program will explore the validity of alternate representations.

6.0 FLIGHT AND GROUND TESTING

While flight tests provide an invaluable source of field information for structural integrity, they are expensive, tie up valuable assets for a significant time, and require significant planning. In order for the test to be an asset to the force management concepts of Figures 1 and 2, knowledge of critical areas and substantiation parameters must be applied early on. Prior knowledge gained from ground component strain surveys can be used to reduce the number of strain gages required or measure only the most important parameters. To further reduce instrumentation costs, the ground test assemblies can be placed on the flight test aircraft after the strain surveys have been conducted, provided minimal damage has been done to the gages.

Flight test conditions are also limited to a controlled test plan and a number of external factors such as scheduling and maintenance. Random variables such as pilot technique, turbulance levels, and mechanical wear are not reproducible test conditions. These factors require analytical investigation of loads to fill the holes in the test data. A well planned flight test will pay for itself many times over when the proper engineering data is gathered to easily perform supplementary analyses.

The flight test measured load and strain data used by most aircraft manufactures in fatigue assessments consists of either 95th percentile or maximum vibratory stresses. It is assumed that the maximum vibratory load acts continuously over each maneuver, leading to conservative crack propagation times. Flight loads data is generally processed at a specific dominant frequency (e.g. once per main rotor revolution), which filters out all higher frequency data. Processing of flight loads data can be improved by incorporating rainflow cycle counting procedures to retain adequate frequency data for DTA and safe-life fatigue calculations.

Sensitivity studies were performed by Sikorsky on rainflow cycle counted H-53 flight test data. It was found that data should be digitized at a frequency at least 10 times higher than the frequency of interest in order to adequately capture peak-valley characteristics of the waveform. Results showed that cycle counting reduces undue conservatism, and in some cases, dramatic improvement in crack growth time was noted. On the other hand, many components showed little difference. Because of the computational expense and extra bookkeeping associated with cycle counting, the procedure is recommended only for selected critical flight test parameters.

Controlled strain surveys also permit important validation of analytical models. In general, finite element, boundary element, and photoelastic models are very good in determining local stress concentrations provided that the correct loads, boundary conditions, and basic load paths within the structure are understood. The emphasis in the strain surveys is therefore to verify these parameters rather than evaluate very local strains. In rotor head assemblies, load paths are very complex and often nonlinear due to the presence rotating joints and bearings. Sikorsky has recently completed a survey for the H-53 elastomeric rotor head for WRALC and will use the results to validate a detailed finite element analytical model to be constructed in a future effort.

7.0 FUTURE IMPROVEMENTS

7.1 STRESS ANALYSIS

Flight test data and airframe finite element models normally provide data on structural loads. These structural loads are usually adequate for rotor component safe-life replacement times, since full scale fatigue test s-N data is obtained and presented in terms of these loads. However, for airframe structure where safe-life evaluations are based on small specimen fatigue data, and for all damage tolerance analyses it is necessary to know stress at the crack origin and the stress distribution along the crack path. In addition, for crack growth analysis the relationship between stresses and stress intensity must also be established.

Potential crack locations in existing helicopter structure often involve threaded parts such as main rotor control rods and dampers, and tail rotor control rods, sleeves, and spindles. The stress analysis of threaded parts can be performed with traditional h-method finite element models, but at tremendous computational expense to obtain accurate results. Other more efficient stress analyses have been developed using a combination of thread load analysis with a boundary element analysis to obtain local thread root stress distributions. The boundary element method has been shown to provide accurate thread root stresses once the thread load distribution is known. The accuracy of the thread load distribution, however, is not yet well established. United Technologies has recently expanded boundary element analysis to allow the solution of contact problems. This may allow efficient solution of both thread load distribution and thread root stresses with this approach alone. Studies must be performed to investigate friction effects in threaded parts, including type of contact and lubrication, and range of friction coefficients.

7.2 FULL SCALE COMPONENT TESTING

Application of the structural integrity computer program to force management of helicopters requires confident evaluation of fatigue behavior for component assemblies. This is accomplished by obtaining crack initiation/growth data for a few representative selected structures via laboratory full scale testing.

The full scale test procedure consists of analyzing all available flight stress data for critical flight regimes and determining an appropriate propagation test spectrum in terms of a critical flight stress parameter. The conditions are duplicated in the laboratory through proper application of loads in a test fixture. A fatigue crack can be initiated via normal cyclic loading or with the help of an induced artificial flaw. Testing proceeds subjected to constant amplitude and spectrum loading conditions until fracture or an appropriate inspection interval has been demonstrated. The propagation of the fatigue crack is monitored by the most appropriate inspection method, based upon accuracy, availability, and feasibility of conducting the inspection in the field. Crack length versus spectrum flight hours is recorded as engineering data.

Metallurgical specimen evaluations are employed to investigate crack origin analysis and striation counts of the fracture.

This procedure has been carried out by GTRI and Sikorsky for the H-53 standard oil lubricated main rotor head. Since service initiation, the hub has experienced failures on Marine, Navy, and USAF aircraft. A test program was conducted to gather crack growth data for inspection interval calculation and to assess NDI techniques for the failure site.

Initial studies showed crack growth times to be roughly 30% of actual failure times, a value too conservative for practical use. The MODGRO damage tolerance analysis computer program (Reference 21) has capability to make beta factor corrections in an arbitrary stress field defined by normalizing against crack origin stress and specifying gradients along both axes of a semi-elliptical crack. The program interpolates these values and uses them in a Vroman integral technique as the crack grows. By carefully measuring the stress field in the vicinity of the crack and incorporating the nonuniformity into the crack growth model, excellent correlation was achieved. Correlations were made with experimental crack propagation data taken under spectrum and constant amplitude loading. A comparison with results of several spectrum test specimens showed that appropriate trends were present in the analytical crack growth curve shape and magnitude.

The success of this technique suggests that these procedures should be adopted in future crack growth testing for validating the structural integrity computer program. Both constant amplitude and spectrum loading is recommended. The errors inherent in modeling the crack geometry and material data are not present during the full scale component test. Constant amplitude experimental testing is not complicated by overloads due to variable amplitude cycling which results in crack retardation and acceleration. With constant amplitude failure data, the effects of geometry and nonuniform stress fields can be isolated. Controlled spectrum load tests provide necessary empirical data for retardation models which can then be applied to analysis using a fully random variable amplitude usage spectrum.

8.0 SUMMARY

Due to the foresight of the Warner Robins Air Logistics Center, a major long term engineering effort has recently been launched to investigate aspects of damage tolerance based fleet management for rotorcraft. The specific goals of the USAF program are to enhance aircraft safety, improve maintenance procedures, increase mission capability, and lower life cycle cost by thoroughly investigaing the technology requirements of HSIP in an overall program sense. Individual tasks are funded as possible to advance the technology required in a structural integrity computer program (Figure 2) which meshes into an existing framework (Figure 1) for management of the fixed wing fleet.

This is an important achievement after the unfortunate demise of the U.S. Army's intended formal HSIP program due to budgetary constraints. It is noted that the Army requires a structural integrity program plan for new rotorcraft systems, although not regulated by a specific standard. As an example, the RAH-66 Comanche program incorporates an HSIP which generally follows the guidelines of MIL-STD-1530A USAF Aircraft Structural Integrity requirements.

Georgia Tech and Sikorsky Aircraft are pursuing the development of a state-of-the-art structural integrity computer program to serve as an automated tool for determining component retirement times and crack growth times of critical structure under changing mission requirements. This work builds upon the combined experiences of both organizations in the field of damage tolerance and force management. A new method adapted specifically to rotorcraft for evaluating and ranking the criticality of fatigue prone components is being applied to a variety of structural assemblies. The most severe components will be selected for further development of the computer program.

Required improvements in several technology areas are being researched to develop a valid tool for USAF use. New stress intensity factor solutions and methods are being explored for crack geometries typical of helicopter component failures, as well as assessing the application of standard solutions. Short crack growth models and material propagation rate data are being developed because of the sensitivity of rotor components to small flaws (.005 to .020 inches). Analytical and experimental short crack programs initiated in these areas by Georgia Tech and Sikorsky will continue through 1994.

Flight and ground test efforts are also in progress to augment the loads database for fatigue analysis. Recent experience has been gained in conducting strain surveys and crack growth tests of full assemblies to provide understanding of load paths in complex dynamic structures to validate analytical finite element and boundary element models.

Coupled with WRALC's HSIP framework, and IHTP usage monitors now under development at Sikorsky, the structural integrity computer program offers a capability to continually evaluate the management of the helicopter fleet to optimize safety, availability, and operational cost.

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