DAMAGE TOLERANCE ISSUES AS RELATED TO METALLIC ROTORCRAFT DYNAMIC COMPONENTS

R. A. Everett, Jr.
Materials Research Engineer
U.S. Army Vehicle Technology Center
Hampton, Virginia

W. Elber
Director
U.S. Army Vehicle Technology Center
Hampton, Virginia

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ABSTRACT

In this paper issues related to the use of damage tolerance in life managing rotorcraft dynamic components are reviewed. In the past, rotorcraft fatigue design has combined constant amplitude tests of full-scale parts with flight loads and usage data in a conservative manner to provide "safe life" component replacement times. In contrast to the safe life approach over the past twenty years the United States Air Force and several other NATO nations have used damage tolerance design philosophies for fixed wing aircraft to improve safety and reliability. The reliability of the safe life approach being used in rotorcraft started to be questioned shortly after presentations at an American Helicopter Society's specialist meeting in 1980 showed predicted fatigue lives for a hypothetical pitch-link problem to vary from a low of 9 hours to a high in excess of 2594 hours. This presented serious cost, weight, and reliability implications. Somewhat after the U.S. Army introduced its six nines reliability on fatigue life, attention shifted towards using a possible damage tolerance approach to the life management of rotorcraft dynamic components. The use of damage tolerance in life management of dynamic rotorcraft parts will be the subject of this paper. This review will start with past studies on using damage tolerance life management with existing helicopter parts that were safe life designed. Also covered will be a successful attempt at certifying a tail rotor pitch rod using damage tolerance, which was designed using the safe life approach. The FAA review of rotorcraft fatigue design and their recommendations along with some on-going U.S. industry research in damage tolerance on rotorcraft will be reviewed. Finally, possible problems and future needs for research will be highlighted.

INTRODUCTION

Up to the 1990's rotorcraft, RC, have been designed almost exclusively using the safe-life philosophy with the Palmgren/Miner [1,2] nominal stress rule being used to establish retirement times for the dynamic components. Since the early 1980's several events have occurred that questioned the reliability of the safe life philosophy. Following several U.S. Army initiatives, under the umbrella of the American Helicopter Society (AHS) a round-robin in 1980 was centered around a hypothetical pitch link problem [3] in which both U.S. and European participants were asked to determine the retirement life of a pitch link. The results of this round-robin showed variations in predicted fatigue life from 9 to 2,594 hours. Even though all of these analysis used a methodology where most of the several parts of the analysis used conservative assumptions (top of scatter loads, various strength reductions from the mean S/N curve, etc.) , none of them could quantify the overall risk associated with a component's fatigue life. Because of several shortcomings in these conservative safe-life methods as well as several U.S. Army helicopter components which did not perform up to expectations, the U.S. Army established a new criteria in the 1980's which aimed to operate and design helicopter dynamic components with a risk of failure of roughly one in a million, or a reliability of "six nines" (0.999999) over their design life and usage. The statistical parameters needed to employ a "six nines" reliability requirement were explored in depth in a round robin instituted by the AHS subcommittee on fatigue and damage tolerance with participation by all of the major U.S. helicopter manufacturers [4]. Further, in the 1988 AHS technical specialists meeting on advanced rotorcraft structures a paper entitled "U.S. Army Requirements For Fatigue Integrity" [5] outlined several key points concerning a reliability analysis as incorporated in fatigue life analysis. Recommendations were also made in this paper that suggested the continuation of exploring the use of damage tolerance concepts in rotorcraft design. This 1988 AHS paper made reference to studies back in the late 1970's which addressed using damage tolerance for helicopters [6]. In this 1988 AHS paper which centered on the importance of nondestructive testing (NDT) in applying the damage tolerance methodology to rotorcraft, the reliability of the NDT technique was examined and cost tradeoffs between safe-life design and "on-condition" damage tolerance design were discussed. It was also concluded that through the use of damage-tolerant design and close control of NDT reliability it would be possible to minimize the dangers in aging helicopters and extend their useful life. Since the Aloha airlines accident in 1988, a significant effort in the U.S. involving the FAA, NASA, the fixed-wing major airframers, and others addressed the several concerns with aging aircraft and sought to develop the tools necessary to lower the chance of fatigue failures in aging fixed-wing aircraft [7]. Some of the analytical methods developed in this program might help overcome some of the difficulties in applying damage tolerant design to rotorcraft.

This paper will start by reviewing a U.S. Air Force study on applying damage tolerance to existing rotorcraft. Also reviewed will be a successful attempt to design a dynamic component, a pitch rod, using damage tolerance. The FAA review of possible aging rotorcraft problems will be discussed along with their recommendation to use damage tolerance design methods in rotorcraft. As a result of this FAA review into aging rotorcraft problems, U.S. industry has instituted a number of research projects on damage tolerance and rotorcraft. These projects will be discussed. Finally, possible problems and future needs for research will be highlighted.

EXISTING ROTORCRAFT AND DAMAGE TOLERANCE

The damage tolerance methodology evolved out of the U.S. Air Force aircraft structural integrity program(ASIP) which was initiated about 1958 after unexpected fatigue failures occurred in the B-47. In the beginning of ASIP, the safe life approach was used incorporating reliability concepts. Because of a number of unanticipated fatigue failures on such aircraft as the B-52, KC-135, F-5 and the F-111, the use of the safe life approach started to be questioned. The threat to safety from manufacturing damage when using brittle materials at relatively high operating stresses was not precluded by the safe life approach. From seeking to find a way to solve these problems, the damage tolerance approach was conceived. The use of damage tolerance design considerations was formally introduced in the U.S. military in 1972 when MIL-STD-1530 was published. This military specification used the analysis techniques defined in linear elastic fracture mechanics to describe crack growth in metallic materials. This along with inspections for cracks helped the U.S. Air Force evaluate the losses of the F-111 aircraft due to fatigue failures in the wing pivot fitting. The U.S. Air Force (USAF) proceeded to use these concepts to manage the fixed-wing fleet of aircraft in their inventory. At first this was only used on the airframes.

In 1978, the USAF first applied damage tolerance to structures which had similar characteristics to rotorcraft when they used this approach on the F-100 engine used on the F-15 and F-16 aircraft. The major differences between fixed-wing structures and engines were the high frequency of loadings experienced by engines, high thermal loads, and the occurrence of tearing and shearing modes of fracture. The success of the engine ASIP with the F-100 engine was estimated to provide a cost savings of $64 million [8]. These facts encouraged the USAF to explore the use of damage tolerance in rotorcraft. Consequently, in 1983 Sikorsky Aircraft was contracted by the USAF to conduct a damage tolerance assessment (DTA) of selected HH-53 rotorcraft structure.

This project had three major goals. First, develop a computer code that would integrate with an existing USAF management system producing a damage tolerance assessment of selected rotorcraft structure. Second, selected rotor and airframe structure crack growth characteristics were to be investigated by analysis. Third, technical issues and possible difficulties related to a DTA management system were to be identified.

Usage and Loads

The computer code developed several aspects needed in a damage tolerance assessment. Among these were a usage spectrum, structural loads, load/stress relationships, and crack growth analysis. In the process of developing the part of the computer code that would define usage statistics it was concluded that the available usage information was not adequate to develop accurate usage statistics. Due to this shortcoming in developing good usage information, Sikorsky was given a contract to develop usage monitors for the HH-53. This has currently evolved to where the new S-92 rotorcraft by Sikorsky will come provided with health and usage monitors (HUMS) on both the commercial and military versions of this helicopter.

For the DTA study of the HH-53 the conservative approach used in the rotorcraft safe life method was employed where it is assumed that for each maneuver the 95% or maximum vibratory load and its associated steady load is assumed to occur over the entire time of the maneuver. It was concluded at the end of this study that this conservatism needed to be further evaluated [9] and probably should not be used in a damage tolerance analysis. It has been expressed by several rotorcraft engineers that load distributions on a component from a specific maneuver are known better than the different types of maneuvers that make up a mission profile. However, it has also been stated that variations in loads due to pilot technique can cause a factor of two difference in loads measured for the same maneuver. This variation in loads with pilot technique and other possible sources that can effect load estimates (weather, etc.) should be investigated in a well planned flight loads survey.

Sequencing (or ordering) of flight regimes and load interaction effects resulting from helicopter loads should also be investigated. The USAF has stated that if a random sequence of maneuvers is used in the usage spectrum, as currently done by the USAF for fixed wing analysis, then sequence effects should not be significant in helicopter spectra [9]. A similar viewpoint has been expressed where such issues in usage definition as regime severity, altitude, CG, GW, and rotor RPM are more important than sequencing of flight regimes. A further complexity in loads determination is how phasing and varying rotor RPM will effect the loads [8].

Since very little usage data exist for helicopters, it may take further study of usage data before these variables can be considered with any degree of confidence.

Stress Analysis

As was the case for the fixed wing aircraft, the lack of detailed stress analysis was obvious when Sikorsky undertook this project. During the development phase of the HH-53, strength of material stress analyses were used. Again, as for fixed wing aircraft this was not sufficient for a damage tolerance assessment. To assess the critical crack initiation areas a detailed stress analysis was required such as finite element analyses. Two analysis tools were used by Sikorsky. Both NASTRAN finite element models and boundary element (BE) models were used to assess the stress states for the selected DTA components.

For some of the selected components where cracks would initiate in areas that contained threads, modeling with NASTRAN was found to produce finite element models with massive element meshes. The stress analysis in threaded areas was treated with a two step procedure that involved a thread analysis scheme that related external loads to thread load distributions and then employed these thread load distributions in a localized BE model of the thread crack origin to obtain crack path stress gradients [8]. Comparison of these thread stresses with finite element models showed good agreement if the thread load distribution was correct. One of the recommendations of this study was to conduct a more thorough evaluation of these thread load distribution models.

One of the difficulties in performing finite element analyses on rotor component structures is specifying the proper boundary conditions. A static load strain survey would be very useful in developing confidence in the assumptions made in choosing the boundary conditions.

Damage Tolerance Assessment

The DTA can be broken into two somewhat separate but related technical issues. In the DTA on the HH-53 done by Sikorsky, these issues were the crack growth model and the inspection for cracks. In the crack growth, model such issues as the determination of stress intensity factors and establishing material crack growth data were the principal considerations. In the crack inspection area the main concern was the smallest crack size that could be reliably detected.

One of the main ingredients in the crack growth model is the determination of stress intensities for the various structural geometries from which cracks will grow. In this study for the components selected for the DTA, surface flaws and corner cracks in non-uniform stress fields were the prime flaw types considered. Stress intensity solutions were needed in stress gradients found adjacent to lugs, holes, fillet radii, and threads. During this study, solutions for lugs and holes in uniform stress fields were readily available. However, for other geometries approximate conservative solutions were formulated. This involved the modification of part through crack solutions in a non-uniform stress field. The equation for this solution took the general form of

$$K = \sigma_o \sqrt{\pi a} \beta \beta_s$$

where σ_o is the crack origin stress, a is the depth dimension of the part through crack, β the geometry factor of Newman and Raju [10], and β_s is the non-uniform stress field correction factor which was developed by the Wright Labs of the USAF [8]. For moderate stress fields this solution was found to be adequate. However, for stress fields with steep gradients like threads, this solution was too conservative. It was concluded that further development was needed for stress intensity solutions in steep stress gradients.

The material property data for crack growth rates were taken mostly from the USAF Damage Tolerant Design Handbook [11]. Some crack growth data were taken from a few constant amplitude tests conducted on compact tension specimens. The test results for the steel (4340) and titanium alloy (Ti-6-4) specimens agreed fairly well with data from the literature. However, the aluminum alloy (7075) test results were somewhat suspect. It was concluded that future tests done on the aluminum alloy should be conducted on center-crack plate specimens.

Some spectrum testing was conducted under a typical helicopter spectrum on compact tension specimens. These tests were done in the mid-range of the crack growth rate curve and showed very little retardation effects in the aluminum alloy and moderate effects in the steel and titanium alloy. Spectrum tests were also conducted with smaller initial flaw sizes. In these tests, more retardation effects were noticed.

Some spectrum analytical studies done by Sikorsky using the generalized Willenborg model showed that spectrum effects may be important. Some crack growth estimates in the mid-range of the crack growth rate curve using the Willenborg model showed a 50% increase in crack growth times due to retardation. Further, similar studies on smaller initial cracks showed component lives could be increased by as much as a factor of ten.

Since no verification of crack growth times existed for these complex structures from full scale parts test, some uncertainty exists in the calculated crack growth times. This is especially true for crack growth times in threaded structures. Further uncertainties existed at this time in small-crack growth times near threshold, because very little test data existed during this study.

HH-53 Investigation Results

Some of the results of this Sikorsky DTA study are shown in Table 1. These crack growth times are mean values without retardation effects. It should be noted that doing the screening evaluation to select the components for this DTA study, most of the components shown in these tables were among the most crack growth sensitive rotor and airframe components except for the transmission support frame. A separate study done at the Georgia Tech Research Institute (GTRI) showed similar results [12].

Results in these tables indicate that a few of these critical rotor components could be managed by damage tolerance if reliable detection of 0.005 and 0.010 inch deep flaws were possible. Others would need to be redesigned. This study also found that some entire rotor components would require reliable detection of 0.030 to 0.050 inch deep cracks for life management by damage tolerance. If some of the conservatisms could be removed (i.e., use of 95% loads, overly conservative K solutions, not using retardation effects) possibly larger crack sizes may be calculated. It was also noted in this study that in the late 1980's automated eddy current techniques used by engine manufacturers had indicated reliable detection of 0.005 to 0.01 inch deep cracks. Of course, you have to know where to look for these critical flaws. Thus the need for highly accurate stress analysis.

Table 1. Crack Growth Results

Structure - Crack Location	Component Location	Material	Crack Propagation Time (Flight Hours) From Initial Crack Depth of : 0.005 in.	0.010 in.	0.030 in.
Upper Hub Plate - Lifting Hole	Main Rotor	Ti – 6Al – 4V	380	50	16
Horizontal Hinge - Damper Radius	Main Rotor	4340 Steel	212	14	-----
Spindle	Main Rotor	Ti – 6Al – 4V			
- Lug			----	> 2000	270
- Shank Radius			----	> 2000	80
Blade Spar	Main Rotor	6061 – T6	----	> 2000	271
Upper Left Pylon Fold Hinge Ftg. - Rivet Hole	Airframe	7075 – T73	410	170	45
Stabilizer Support Fitting - Hole	Airframe	7075 – T73	40	21	6
Transmission Support Frame - Holes	Airframe	7075 – T73	----	----	> 2000

For the airframe, similar results were noted. Most of the airframe components presented in Table 1 would have to be redesigned to be damage tolerance managed. However, as stated in this study, the transmission support frame is similar to most airframe structure and should be able to be damage tolerance managed. It is also noted in this study that the above mentioned conservatisms should be evaluated before any structure is redesigned.

This study, along with comments on this study in a USAF authored paper [9], concludes that damage tolerant management of rotorcraft is viable. In the time frame when this study was done improvements in several areas of technology were recommended. Some of these have been stated above. In the conclusions of this paper they will be highlighted again and put in perspective with today's technology.

A DAMAGE TOLERANT PITCH ROD

In the 1980's a failure of a main rotor pitch rod in the U.S. Army's AH-1 helicopter brought about the design of a pitch rod that was certified because it was shown that the new design met a damage tolerance criteria. The original pitch rod was made of an aluminum alloy and had an infinite fatigue life as determined from safe life concepts. After the failure of the aluminum alloy pitch rod, a new design was presented to the U.S. Army that had an infinite "safe life" but was now made of steel. This new steel pitch rod design was first rejected by the Army. The manufacturer then sought to enhance the U.S. Army's confidence in the new design by seeking to certify the steel pitch rod through damage tolerance analysis and testing.

In order to avoid expensive and time consuming research to adequately address complex helicopter structures and loadings, a simplified engineering approach was investigated. In this approach the actual component was replaced with an "equivalent structure" with built-in conservatisms that would produce conservative crack growth predictions. In this procedure, the complex structure, loading, and crack path geometry is replaced with a simplified counterpart for which a crack growth model had been proven. What made this approach acceptable to the U.S. Army was a verification testing program on the newly designed steel pitch rod that validated the equivalent structures method.

Approach

A series of conservative assumptions along with established material property data set the stage for this analysis. First, the stress intensity factor, K, evaluated for the equivalent structure must be larger than the actual structures:

$$K(a)_{equivalent} > K(a)_{actual}$$

Then based on a stress concentration method the stress fields in the component are evaluated and the critical cross sections are chosen where cracks would probably be initiated. For the cross section of this pitch rod, the critical sections were the rod end bearing banjo housing and the last engaged thread in the bearing shank, which are shown in Figure 1 as sections A-A and B-B, respectively. Finally, the crack growth analysis was simplified by neglecting load interaction effects on crack growth and neglecting the crack growth threshold. To arrive at a stress intensity factor solution for the "equivalent structure" an attempt was made to link the somewhat complex helicopter structures to the classical structures previously analysised in the fixed-wing damage tolerance designs. The approach used was to define a geometric parameter which characterizes the cracked structure and describes its failure. From a practical perspective, this study kept the predominant dimension in the equivalent structure equal to this same dimension in the actual structure. For the rod end bearing banjo housing, the critical cross section was characterized by the width of the cross section, b_{actual}, and was assumed to be the same as for the equivalent cross section of the actual pitch rod (see Figure 2). The equivalent cross sectional thickness was determined from both axial and bending stiffness considerations with the equivalent structure stiffness being less than or equal to the actual structures stiffness.

The next step in defining the K solution was to define the crack shape geometry and location. At the critical cross section the inner corner, Point M in Figure 2, was determined to have the highest stresses and thus was chosen as the location of the initial crack. Hence, the initial crack in the equivalent structure was defined as a corner crack in a finite plate [14]. Finally, a crack growth model is used to predict the crack growth time to failure. For this pitch rod analyses a software package called CRKGRO was used [15]. In CRKGRO a modified Walker equation is used for the load cycles that produce positive stress ratios (ratio of minimum to maximum stress, R) and the Chang equation is used for the negative stress ratio load cycles. Two cases of loadings were addressed. One considered the situation with an undamaged elastomer and the other

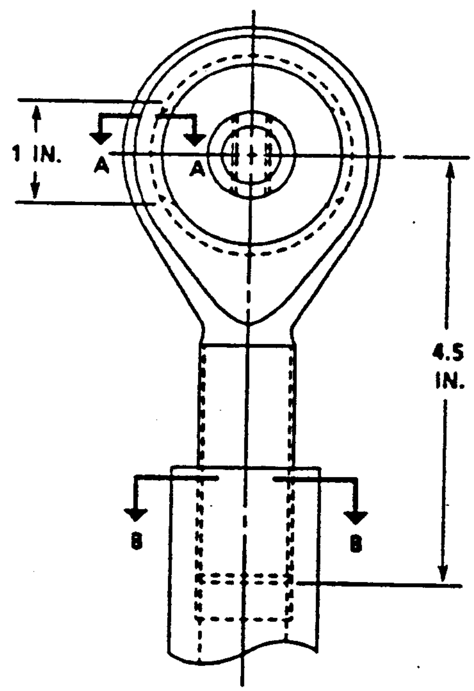

Figure 1. Rod – end bearing.

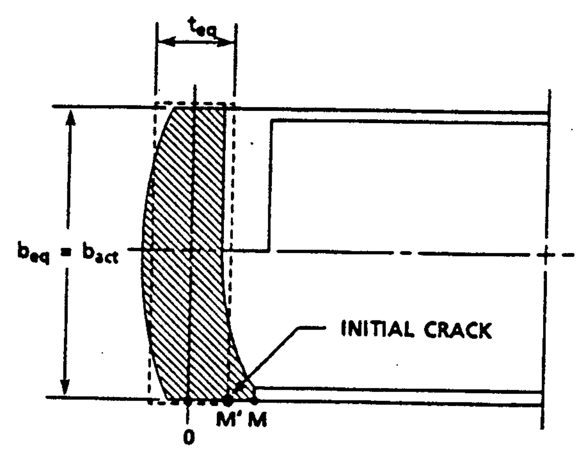

Figure 2. Rod – end critical cross section.

was with a damaged elastomer. For the undamaged elastomer the axial load is reacted by the elastomer, whereas, for the damaged elastomer condition the total axial load is carried by the critical cross section.

The crack growth predictions from the CRKGRO analysis gave a crack growth time from a 0.050 inch initial corner crack of 10,769 hours for the case with the undamaged elastomer, while only 887 hours of crack growth was predicted for the damaged elastomer.

Verification Testing

As a step towards certifying this "new" damage tolerance design using the equivalent structures concept, verification tests were run on the newly designed steel pitch rod. For these tests it was easier to react the test loads by removing the elastomeric bearing. The test loads were constant amplitude with a stress ratio of zero. A corner notch was placed at the inside corner of the banjo housing, using electric discharge machining, to start the growth of the crack. To verify the "equivalent structures" design concept in a damage tolerance assessment, a crack growth analysis using the "equivalent structures" concept was also performed on the test set-up. The crack depth against cycles results from the analysis and tests compared very well. For the crack length dimension, c, the analysis predicted a crack growth time to failure about a factor of two shorter than the test results. These results helped verify that the equivalent structures concepts in a damage tolerance assessment produces conservative crack growth times when compared to test results, thus lending confidence to the certification of this "newly" designed pitch rod in a damage tolerance setting.

Similar results to these were shown for the failure mode in the last engaged thread in the bearing shank. The equivalent structures analysis gave a crack growth time of 16,864 hours assuming a crack geometry that was a 0.050 inch deep and 0.010 inch long semicircular crack at the root of the last engaged thread. For the thread failure mode the verification testing produced very conservative results when compared to analysis. For the crack depth dimension, a, the analysis predicted an order of magnitude shorter life than the test results. For the crack length dimension, c, the crack growth time determined from analysis was over two-orders of magnitude shorter than the test results. These excessive conservatisms were attributed to the high stress concentration factors assumed in the root of the thread which was used in the stress intensity solutions. When an analysis was run assuming no stress concentration , the analysis showed very similar crack growth lives compared to the test results. This same problem occurred for the HH-53 study as stated previously. This pitch rod study gives further evidence that crack growth analysis in the area of threads needs further research.

TOGAA REVIEW

As a result of the Aloha Airlines incident of April 1988 where a high time commercial airliner lost 18 feet of the upper fuselage before landing safely, the U.S. Congress mandated that preventive actions be taken to minimize future catastrophic failures due to aging phenomena. This failure started a national program in the U.S. that included the FAA, NASA, aircraft manufacturers, and the airlines. Shortly after the beginning of the national program the FAA commissioned a panel of noted aircraft expects to review this program over several years. The review panel became known as TOGAA, the technical oversight group for aging aircraft. After the TOGAA review of the fixed-wing commercial aging aircraft program they were commissioned to review the commuter fleet. Then in-turn a review of the commercial rotorcraft fleet was instituted. In November of 1994 the first meeting of TOGAA with the helicopter industry, helicopter operator's and the FAA's certifying agency for rotorcraft was held. As a result of this meeting a series of 15 "comments" was issued by TOGAA which addressed aging rotorcraft issues as well as the basic methodologies and design philosophies used by the rotorcraft community. Some of these comments are shown in Table 2. Since the initial meeting, a series of four additional meetings have discussed these "comments". During these meetings the rotorcraft community has responded to these "comments" with one of the results being the producing of a "white" paper which will be used to aid the FAA in revising the rotorcraft FAA regulations and its accompanying advisory circulars, AC. The "white" paper will present how the rotorcraft manufacturers and operators will meet the newly immerging damage tolerance requirements for rotorcraft as currently required by the U.S. federal air regulations, FARs. In this section, some of the discussions of the TOGAA with the RC community will be presented, but mostly on those issues that pertain to the use of damage tolerance in rotorcraft design and life management. Since at this writing the review of the TOGAA with the RCcomminity is still ongoing the final outcome of this review will need to be reported later.

Table 2. Some of TOGAA's Original 15 " Comments"

1. Accident rate of rotorcraft appears to be high compared to fixed-wing.
2. Aging rotorcraft. Non-issue for rotorcraft components because of component replacement times set by safe-life method. Method to determine these replacement times is an issue (i.e., safe-life method).
3. Aging possible issue for rotorcraft airframes.
4. Definitions in the FAA/FAR's advisory circulars (AC's) for rotorcraft are inconsistent with the AC's definitions for the large transports and commuter aircraft.
5. Research related to rotorcraft including HUMS. Need to know more.
6. Reliance on safe-life method needs further explaining. Can not know about sensitivity of components to manufacturing or in-service flaws.
7. Handling of military surplus helicopters not under control.
8. (a) Maintenance actions dependent mainly on service experience.
   (b) Fracture Mechanics technology needed to supplement service experience to further improve structural/mechanical integrity of rotorcraft.
9. New rotorcraft should be designed by damage tolerance.
10. Older rotorcraft critical components should be re-examined with today's technologies to determine their sensitivity to manufacturing and in-service flaws.
11. Need more research:
    (a) ΔK_{th} and crack growth data for small cracks.
    (b) Establish nondestructive inspectable flaw sizes and improve inspection techniques.

TOGAAs 15 "comments"

The 15 original "comments" issued by TOGAA covered such technical areas as aging issues in rotorcraft, maintenance and inspection intervals, sensitivity of safe life designed parts to manufacturing and in-service flaws, handling of military surplus helicopters and parts, disapproval of the new enhanced "safe life" design approach, an interest in HUMS, that new rotorcraft be designed using damage tolerance, and the need for research on such damage tolerance topics as crack growth threshold and crack growth rates for small cracks.

Aging Rotorcraft

From these original 15 "comments" some were combined and others evolved into new issues. As to aging rotorcraft issues, the rotorcraft community stated that aging related failures are minimal and are adequately managed with present methods. This conclusion was based on the position that a potential aging accident is a fatigue failure in the rotors, flight controls, drive train or airframe systems and only relates to failures that occur prior to a components predicted retirement life. Most aging rotorcraft issues, according to the rotorcraft community, are related to corrosion where corrosion control and repair procedures are in place and working well. Because almost all existing rotorcraft were built with the safe life approach, TOGAA agreed that many of the old helicopters may have to stay on the expensive part replacement program until they are retired. A previous section in this paper on the HH-53 damage tolerance assessment highlighted this situation. However, the TOGAA further stated that selected safety critical components of some older rotorcraft, where service cracking or failure problems exist, should be re-examined using today's technology's to determine their sensitivity to manufacturing and service induced flaws and as necessary implement corrective actions.

Maintenance and Inspection

On the "comment" about maintenance actions and inspections, TOGAA perceived that maintenance actions are currently derived mostly from service experiences rather than being analytically based. The current successes are the results of effectively applying the lessons learned from past difficulties. With today's loads and stress analysis methods and fracture mechanics technology, TOGAA stated that the experienced- based-maintenance plan could be further improved by incorporating these advances in analytical methods to gain a quantitative understanding of the sensitivity of the safety critical parts to manufacturing and service induced flaws. The inclusion of the new advances in these analytical technologies would supplement the experienced-based-maintenance plan and further improve the structural and mechanical integrity of the rotorcraft. The sensitivity to small defects was highlighted in a failure of a U.S. Army UH-60A [9] helicopter in 1985. This accident demonstrated the sensitivity of some dynamic components to small defects which in this case were from almost imperceptible machining abnormalities.

As to inspections, TOGAA cited that a review of existing service bulletins should be undertaken by the RC community to determine if modifications to existing rotorcraft should be incorporated early in their life rather than relying on continued inspections. A new design could also employ the damage tolerance approach. This procedure has been employed by the fixed wing industry for some years.

Safe Life Shortcomings

In a 1995 debriefing with the rotorcraft community, TOGAA commented that for more than six years they have "consistently pointed out the many serious shortcomings of the *safe life* methodology". One of the major concerns to TOGAA is the sensitivity of safe life designed parts to manufacturing and in-service flaws. This stems from the fact that the safe life approach may lead to the selection of wrong materials from a damage tolerance perspective as it gives very little understanding of the physics of the fatigue and fracture process. The safe life approach often leds to the choice of high strength metal alloys with very low fracture toughness that have a good resistance to crack initiation in a laboratory environment, but their low fracture toughness and often small critical flaw sizes produce structures with very poor damage tolerance. These safe life material choices are often very sensitive to environmentally-induced stress-corrosion cracking and/or corrosion-fatigue. TOGAA further commented that the rotorcraft industries use of 6 "nine" reliability designs are mostly achieved for conforming parts and has probably been achieved because of using very conservative loads and a statistical reduction of the design S/N curve developed from constant amplitude tests. As it has been stated, most rotorcraft failures are not brought about because of problems (or errors) in the determination of the retirement times predicted by the safe-life approach, but are due to such damage tolerance issues as manufacturing defects, maintenance errors, corrosion, and unanticipated loads.

Military Surplus Rotorcraft

During these meetings with the TOGAA, the FAA, the manufacturers, and some operators expressed possible concerns about the influx of retired military rotorcraft and parts into the civilian arena. It had been stated that the U.S. Army has been and will be planning to retire perhaps greater than 3000 helicopters. It was speculated that as many as 1000 of these could see service in civilian use. It was also stated that military accidents are a factor of two greater than civilian accidents. Helicopter operators felt that these military surplus helicopters must be FAA certified. TOGAA's biggest concern was the FAA's ability to provide the necessary resources to process these certification requests. A further concern was the FAA's ability to provide adequate oversight in the remote operating areas where some of this type of equipment probably will be used. TOGAA also expressed a strong opinion that "the FAA mandate" that spare parts with uncertain or undocumented maintenance histories should be destroyed.

Future Research Needs

As to future areas where more research and development are needed to insure the evolvement of damage tolerance concepts and designs into existing and future rotorcraft, the TOGAA suggested several technical areas. Among these were the need for understanding the role of crack growth thresholds in rotorcraft design, crack growth data for small cracks over the range of stress ratios of interest, establishing NDI inspectable flaw sizes, and improving inspection techniques. To the list of research needs the FAA added the development of flight-by-flight load spectrum and rotorcraft health and usage monitoring systems(HUMS).

Up through 1997 the meetings of the TOGAA with the rotorcraft community have continued. Many of the comments and issues stated previously in this paper have been addressed by the rotorcraft community, and many questions have been answered to where the current status will be presented at a winter meeting in 1998 where the "white" paper should be presented and discussed. Since these "comments" were first generated, several pertinent presentations have been made which highlighted the following topics.

The Aging Rotorcraft Program is now a part of the National Aging Aircraft Program. One presentation highlighted a comparison of helicopter and fixed-wing fatigue substantiation similarities, differences, and methodology applications pointing out the differences in loading frequencies, needed inspectable crack sizes, and the practicality of the different fatigue substantiation methodologies. Another presentation by a U.S. industry rotorcraft manufacturer gave the application of a damage tolerance design using a 0.015 inch rogue flaw on a pylon isolator. This presentation concluded that "the application of damage tolerance was a key to resolving premature partial failures and extending the life of this part to an acceptable level within the given design restraints."

A final topic of great concern to the TOGAA must be addressed. This is the inclusion of the so-called *flaw tolerant safe life method* currently in the FAR's. One of the main concerns of the TOGAA with the flaw tolerant life method is the scatter that could be in a component's life limits by unconservative selection of the flaw sizes and flaw distribution for each component. Further concerns about this method is the problem of defining all the possible types of flaws and the many number of S/N tests required for life analysis. Another possible problem with this technique concerns the method to put these flaws in the test articles. It was thought possible that the methods used to put these flaws into the parts may actually cold work the parts thus retarding crack initiation.

Many of the issues stated in this review of the TOGAA's review of possible aging helicopter problems, the short comings of the safe life approach, concerns with the new flaw tolerant method, and the forthcoming "white" will be discussed in the meetings to come.

CURRENT ROTORCRAFT RESEARCH

In response to new and changing requirements by procuring and regulatory agencies(DOD and the FAA), U.S. rotorcraft manufacturers are conducting several research programs to address the issues needed to design and certify rotorcraft according to damage tolerance life management. The USAF first started the move to rotorcraft damage tolerance designs in a somewhat thorough study of the HH-53 as discussed previously. In the late 1980's the FAA added to FAR 29.571 a flaw-tolerant safe life design approach besides the more classical crack growth damage tolerance method as used by the USAF for the last two decades.

Program Objectives

Under the RC manufacturers research programs a damage tolerance for helicopter structures study has been initiated with three main objectives. The first is to investigate the relationship between safe-life and damage tolerance. Second, is to establish and validate a method for determining replacement times and inspection intervals based on damage tolerance concepts. And third, to develop for the rotorcraft industry guidelines for damage tolerant design, certification, and maintenance of rotorcraft structure.

Technical Program

This damage tolerance research program was conceived as a building block approach. First, crack growth analysis on several materials used in helicopter design will be performed using the computer codes developed at NASA called NASGRO and FASTRAN. Crack growth times to failure will be determined for "initial material quality" and damage tolerance crack sizes and the results of these analysis will be compared with test data found in the literature. Second, these crack growth models will be updated and validated by comparisons with coupon S/N and crack growth data. These models will be further evaluated by comparison with existing full scale component tests, as well as some additional full scale component crack growth tests. From the results of this building block analysis and test program, recommendations will be made on improvements needed in the existing crack growth models, crack growth inspection intervals, and the assumed component load spectrums.

Part of this work is being aimed at addressing the issues where past studies (i.e., HH-53, etc.) have encountered difficulties in applying damage tolerance to existing and aging helicopters. Other technical areas to be addressed that are deemed important in rotorcraft damage tolerance is the so-called "small" crack effects on crack growth, as well as the establishment of crack growth thresholds. Concerns with crack growth thresholds will be discussed further in the next section of this paper.

CONCERNS AND ISSUES

In concluding this review of damage tolerance rotorcraft technical issues and concerns, a list of these issues and concerns as stated previously in this paper along with others expressed by various U.S. rotorcraft manufacturers, certifying agencies, and concerns from DOD engineers (in private communications) is given as follows:

1. Stress levels may need to be lower for damage tolerance compared to safe life designs causing some weight increase.
2. Crack growth thresholds are critical for rotorcraft. Most, if not all of the stresses need to be below the crack growth threshold. Are the long crack thresholds currently in the literature too high ? Are "small" crack thresholds different than long crack thresholds ?
More research is needed on the fundamental nature of crack growth threshold development.
3. Without some universal agreement on thresholds and crack growth rates, could have similar wide-spread variation in crack growth inspection intervals as in safe-life retirement times.
4. Are "small" crack effects significant for rotorcraft ?
5. Stress intensity solutions more difficult because of complex loadings, and 3-D stress fields in complex 3-D geometry(e.g., main rotor grip, mast).
6. Safe-life designed parts often have low crack growth lives.
7. Damage tolerance inspections could make operating costs excessive.
8. Environmental effects.
9. High frequency content of load spectrum (10,000 to 60,000 cycles per hour) can cause rapid crack growth.
10. How to handle load counting(e.g., rain-flow, top-of-scatter).
11. Load-interaction effects.
12. Complex load issues such as rotor load phasing and variable RPM.
13. Variation of external loads with such things as pilot techniques.
14. Usage spectrum varies widely because of rotorcraft multipurpose applications. HUMS needed.

This list is quite varied but by no means all inclusive. Only as all aspects of life management of rotorcraft with damage tolerance are worked, will hopefully all of the issues be understood. But like all of our technologies, other problems will surface during rotorcraft operations and these problems will be addressed as the rotorcraft community moves toward an ever evolving life management of rotorcraft with damage tolerance.

CONCLUDING REMARKS

In this paper issues related to the use of damage tolerance in life managing rotorcraft are reviewed. It was noted that for the most part, existing helicopters have been designed using the safe life method which has at its center the Palmgren/Miner nominal stress rule. But as a result of analytical round-robins on component replacement times and several helicopter accidents, structural life management is moving through a "six nines" reliability era towards that of damage tolerance. The paper reviewed the study conducted for the USAF where a damage tolerance assessment of the HH-53 rotorcraft showed that on existing helicopters it is very difficult, if not impossible, to life manage safe life designed parts using damage tolerance. A further review of a newly designed pitch rod illustrated a conservative equivalent structure concept of validating the damage tolerance characteristics of this component. Next, the FAA's evaluation of a potential aging rotorcraft problem, as conducted by the TOGAA panel, was reviewed highlighting the concerns of this technical panel as regards to possible aging rotorcraft concerns, the shortcomings of the safe life approach, concerns with the new flaw tolerant method, and other issues effecting the design of rotorcraft. Brief statements are also made on some of the U.S. rotorcraft manufacturers research programs in which industry is not only attempting to answer some of the concerns expressed by the TOGAA, but are also addressing their own evolvement towards a better structural integrity helicopter. In conclusion, a list of concerns and issues towards damage tolerance assessments of rotorcraft is given which includes such issues as crack growth thresholds, stress intensity solutions, and the complex loading environment of rotorcraft.

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