FLAW TOLERANCE SUBSTANTIATION TEST RESULTS FOR S-92 DYNAMIC COMPONENTS

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Abstract

The Sikorsky S-92 is the first all-new helicopter design to provide tolerance to flaws and damage on fatigue-loaded components, as required by the FAA's landmark 1989 rule change. This was accomplished on the S-92 dynamic components by a low stress robust design approach that was substantiated by "Flaw Tolerance" methodology. This methodology provides fixed retirement times that preclude fatigue failure even if the part has critical flaws in critical areas. Full-scale S-92 fatigue test results are now available which show that the goals of Flaw Tolerance – excellent structural reliability and long retirement times even with flaws - have been achieved. An inspection interval determination using Flaw Tolerance also produced good results – easy inspections at reasonable intervals. It is concluded that Flaw Tolerance is a viable and effective method of improving component fatigue substantiations, and that it can offer advantages over other methods.

Introduction

The Sikorsky S-92 (Figure 1) is the first all-new helicopter design for which application for civil certification included Amendment 29-28 of the US Federal Aviation Regulations Part 29 (Transport Category Rotorcraft). This amendment changed Paragraph 29.571, "Fatigue Evaluation of Structure", and was released in October 1989, Reference 1. This rule requires that catastrophic failure due to fatigue "considering the effects of environment, intrinsic/discrete flaws, or accidental damage" will be avoided.

These few words provided the impetus for the most significant change in how helicopters are substantiated in fatigue since the implementation of the first safe-life substantiations in the 1950's.

The rule offers two methods to meet the requirement – "Flaw Tolerant Safe-Life", which uses conventional crack initiation safe-life methods to establish retirement times for components with flaws; and "Fail-Safe (residual strength after flaw growth)", which uses crack growth methods to establish inspection intervals for flawed components. The terminology used by many

Figure 1. Sikorsky S-92 Helicopter

Presented at the 28th European Rotorcraft Forum, Bristol, UK, September 19, 2002

players is “Flaw Tolerance” for the first method (crack initiation from an initial flaw), and “Damage Tolerance” for the second (crack growth from an initial crack).

This paper will discuss why the most frequent design choice for S-92 dynamic components was Flaw Tolerance, and how this choice has been successfully substantiated by means of full-scale fatigue testing of flawed components. Although the current FAR 29.571 is used for both metals and composites, only its application to metals will be included in this paper.

The S-92 is scheduled for its initial FAA Civil Type Certificate in December of this year. Future military applications of this design will also be able to take advantage of the robust long life components that provide low maintenance costs associated with a Flaw Tolerant design.

Background

The Flaw Tolerance method, also called Flaw Tolerant Safe Life or Enhanced Safe Life, has been in place as an acceptable approach for Civil Helicopters since 1989, References 1 and 2. Only a few applications of this new rule occurred in the nineties, and, until 1999, there was no generally available public view of how it could be specifically implemented. At that time, a group representing European and American rotorcraft industries provided a “White Paper” addressing the implementation of Damage Tolerance and Flaw Tolerance to rotorcraft. This document was prepared for the Technical Oversight Group for Aging Aircraft (TOGAA), and was later presented at the European Rotorcraft Forum, Reference 3. Another paper, Reference 4, addressed the Flaw Tolerance method specifically and how it was envisioned to be used on the S-92. Reference 5 compared the potential application of both the Damage Tolerance and Flaw Tolerance methods to the S-92 main rotor hub.

The premise of Flaw Tolerant Safe Life is to establish retirement times (and inspection intervals) based on the strength of parts that have critical flaws in critical locations. It is a crack initiation method in that the life (or inspection interval) is fixed by the first sign of cracking from the flaw, or any growth of the flaw. The component life (or inspection interval) is based on the same analysis, same flight loads, and same usage spectrum employed in conventional safe life calculations for that model helicopter.

The flaw types and sizes should be based on a “hazard assessment” unique to the model helicopter, the principal structural element, the materials and protective coatings used, the manufacturing processes used, the inspection procedures in place, the maintenance operations employed, and specifics of the environment of operation. The flaw sizes considered in the fatigue substantiation can be limited to the maximum in the hazard assessment, or to one of two pre-established sizes depending on whether a life analysis or inspection interval determination is being conducted. The first are “barely detectable flaws”, intended to represent a worst case of undetectable flaws. The barely detectable flaws are used to establish the component replacement time since they can be conservatively expected to remain in place for the life of the component without detection. The second type of flaw is the “clearly detectable flaw”, intended to represent the largest size flaw that could remain in place for a limited period of operation in spite of routine visual inspections for general condition (such as “pre-flight” inspections). Clearly detectable flaws would be expected to be easily found during scheduled directed inspections. The interval for this inspection is based on the safe life of the part with the clearly detectable flaws imposed in critical areas.

The use of the Flaw Tolerant Safe Life method to establish inspection intervals is not specifically addressed in the current FAR 29.571 rule. For this reason, inspection procedures generated by this method for the S-92 will be limited to “manufacturer’s recommendations”. Thus, the inspections will not be FAA-mandated, but the retirement times will be. However, future revisions to the rule and associated Advisory Circulars, which are now in the review stage, will specifically allow the determination of inspection intervals using Flaw Tolerant Safe Life.

One detail of Flaw Tolerance which has been heavily debated is what working curve reduction factor is appropriate for use with flawed parts. AC 29-2C MG11 talks about using strength reduction factors less than the standard “3-sigma” in order to “preclude a dual penalty situation”, and relates this idea to unmistakable flaw indications, multiple elements, and benign failure modes. The standard proposal is to use a “2-sigma” strength reduction for flawed parts, but this degree of reduction is not agreed to be appropriate in all cases. A stronger case can be made for the use of 2-sigma in the inspection interval application than in the retirement time application. In any case, it has been generally agreed that the conventional safe-life retirement time (as-manufactured strength, 3-sigma reduction) would be used if it was lower than the Flaw Tolerant retirement time (flawed parts, < 3-sigma reduction)

Finally, it was also recognized that it would be impossible to accurately and simultaneously place each type of critical flaw in each critical location on the full-scale fatigue test parts. The use of "equivalent" flaws was proposed, i.e., easily applied and controlled flaws that produce the same strength-reducing effect as the worst type of flaw in the hazard assessment. This equivalent flaw approach would be evaluated and verified by means of a coupon program.

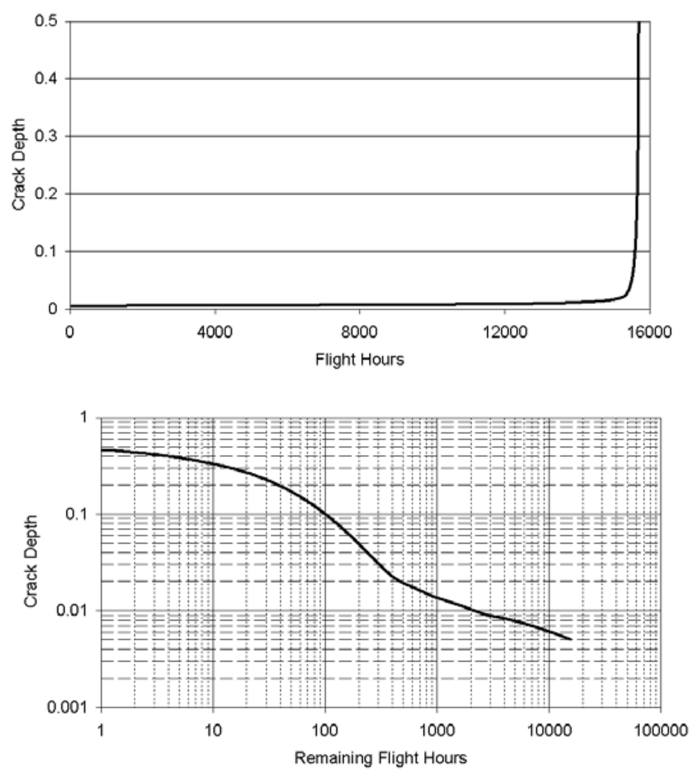
Design of S-92 Dynamic Components

The S-92 dynamic component design approach was to produce robust parts that operate at peak stresses 20 to 25% lower than in earlier designs, in order to meet the FAA requirement which included consideration for flaws and damage in all Primary Structural Elements (PSE's). Weight was added in critical areas to achieve this goal. At the time of the preliminary design of the S-92 in 1994, both Damage Tolerant and Flaw Tolerant substantiation methods were evaluated as compliance candidates. Two Damage Tolerant approaches were considered – Crack Growth With Inspections, and Crack Growth Without Inspections.

Crack Growth With Inspections. This method, which utilizes fracture mechanics methods, offers the advantage that the source of damage or cracking does not matter because an inspection program is formulated to find structural cracking before any crack reaches critical size. This approach has been used with considerable success in fixed-wing transport aircraft, both military and civil. However, application to helicopter dynamic components was, and is still known to be, difficult (References 6 – 8). But the difficulties were considered to be surmountable if damage tolerance features and crack sizing were included in the initial design of a component.

The main rotor yoke, Figure 2, is a good example of a critical component that was thoroughly analyzed and design parameters traded to achieve a satisfactory mechanical and weight-efficient design that could be managed in service by inspections for cracks. However, the complex mechanical constraints that applied to the yoke – extremes of motion and limits in space, combined with accessibility for inspection – resulted in higher stresses than desired at critical locations. Plus, the inevitable increases in observed vs. predicted loads exacerbated the situation, and the crack growth prediction shown in Figure 3 resulted .

Figure 2. Main Rotor Yoke.



Figures 3a and 3b. Predicted Main Rotor Yoke Crack Growth.

Figure 3a has the characteristic "hockey stick" appearance of a crack growth curve on a helicopter dynamic component. For small initial cracks (depths < 0.020 inches) the Ground-Air-Ground (GAG) load will propagate the crack, providing relatively slow stable growth as seen in the horizontal portion of the curve. Then, as the crack becomes larger (depths >~ 0.020 inches), the crack growth becomes dominated by the high frequency rotor n-per-rev loads, causing a rapid increase in growth rate as seen by the vertical portion of the curve. The transition to rapid growth generally occurs at crack depths of about 0.020 inches, which is generally smaller than reliably detectable cracks (Reference 9).

It was essential to achieve a minimum 1250-hour interval for major inspections for the S-92, especially if disassembly would be required. Frequent use of sophisticated equipment and/or inspection expertise to find very small cracks had to be avoided because of the prohibitive training and maintenance cost required for the operator. So as an example in Figure 3b, if a 0.030" deep crack size is chosen as the minimum size that is reliably and practically detectable in service, the crack propagation time to critical size is about 300 hours. An inspection interval of 75 hours would be necessary in this case to provide the necessary safety margin. Conversely, the crack that must be detected to achieve the desired 1250-hour interval with margin is only 0.008 inches deep. Neither of these options is viable.

In general, successfully implementing a classic crack-growth-with-inspection Damage Tolerance approach was found to be very difficult on the S-92 dynamic components, especially those with combined high frequency and low frequency loading.

<u>Crack Growth Without Inspection</u>. An alternate Damage Tolerant approach was also considered – namely the possibility of establishing a retirement time without inspections based on the presence of a maximum anticipated flaw size which could occur in manufacturing or service, but which would not grow to critical size within the specified retirement time. This method has the advantage of no scheduled inspections, but unlike the inspection method, its success depends on the quality of a hazard assessment. Such an assessment was obtained from field history data of several S-76 aluminum and titanium rotor components, Figures 4 and 5 (from Reference 5).

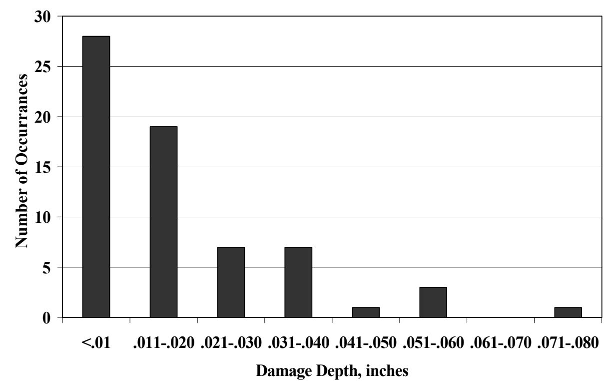


Figure 4. <u>Mechanical Damage on S-76 Main Rotor Head Components.</u>

It should be noted that this population of 66 flawed parts is only a small fraction of the total (about 8000) of all fielded S-76 rotor parts. (At the time of the study, all parts needing repair of flaws were returned to Sikorsky.) A Weibull plot representation of this data is shown in Figure 5.

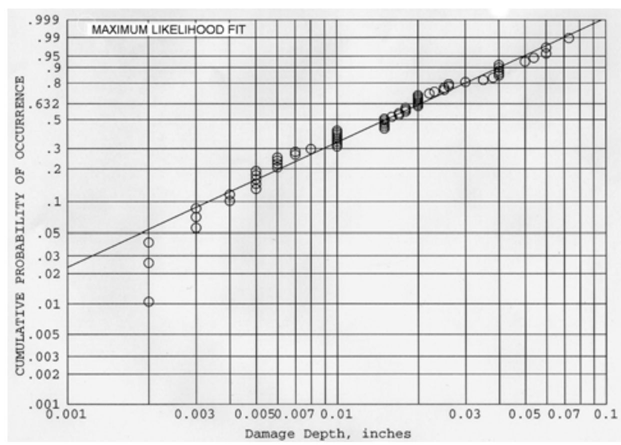


Figure 5. <u>Weibull Plot, Mechanical Damage on S-76 Main Rotor Head Components.</u>

It can be seen that at least one defect larger than 0.070 inches depth has occurred. However, defects of this size are very unlikely, only a 1% chance in this database, which is compounded by the small chance of having a flaw at all (approximately 1 in 120 for the S-76 database). But more importantly, a flaw of this size would be very unlikely to be missed in frequent routine visual inspections for general condition ("pre-flight inspections" for example). A more reasonable but still very conservative choice of a maximum flaw size that could remain in place for the life of the component is 0.040 inches depth. This is later characterized as a "Clearly Detectable" flaw, although for the crack-growth-without-inspection method, it is left in place.

A flaw size (gouge) of 0.040 inches depth can be shown to be equivalent to an initial crack size of approximately 0.027 inches depth. Getting back to the design option of establishing a retirement time based on this flaw (crack) size, Figure 3b indicates that about 350 hours of propagation time remains to critical crack size. No more than 175 hours would be available as a retirement time. Again, not a practical option.

Flaw Tolerant Safe Life. This method offers the advantage that fixed retirement times can be established without the need for inspection for small flaws or cracks, and easy inspections for large flaws can be implemented at reasonable intervals. A hazard assessment is required to establish flaw types and sizes.

Using conventional safe-life analysis including a strength reduction for the effect of flaws, it was found that good retirement times could be obtained with small flaws in place in critical areas on S-92 dynamic components. These are the "Barely Detectable" flaws, intended to be a worst-case representation of undetectable flaws. This representation is essentially a definition of the flaw sizes that could conservatively remain in place without detection for the entire life of the part. For S-92 dynamic components this flaw size is 0.005 inches depth, which was shown to be easily detectable by visual means alone.

By means of a coupon program, the worst type of flaw for aluminum and plain steel was shown to be corrosion, and for non-corrosive metals, a gouge. The coupon program also showed that the effect of the 0.005" flaws on fatigue strength was small, 5% or less. This result would not be expected at the higher stress levels of a more conventional design.

Additionally, it was shown that reasonable inspection intervals could be obtained for components with 0.040 inches deep "Clearly Detectable" flaws in critical areas. This was done by simply determining a high-margin safe life for components having these flaws imposed in critical areas. (The inspection is a simple visual for the presence of the flaw, not for cracks.) For analysis purposes, the effect of the Clearly Detectable Flaws was taken from the coupon program where a reduction of as much as 57% in fatigue strength could be expected. As mentioned before, this inspection interval application of Flaw Tolerance methodology is not considered appropriate under the current Amendment 28 FAR 29.571 Rule, however, the information will be provided to operators as manufacturer's recommendations.

Comparison of Methods For the same flaw sizes, a more practical management procedure was found with Flaw Tolerance methods than with Damage Tolerance methods. The reason is simply that there is always a usable period of crack initiation from the flaw (except for flaws which are already true cracks and shouldn't be treated by Flaw Tolerance anyway). Damage Tolerance, however, must assume the presence of the initial equivalent crack, so does not take advantage of any crack initiation time. So Damage Tolerance conservatively ignores the initiation time and Flaw Tolerance conservatively ignores the propagation time. Flaw Tolerance will yield longer times. This is also helpful to understand why damage tolerance retirement times are typically established with lower imposed margins than would be acceptable in a safe life calculation.

In summary, the design approach most commonly selected for S-92 dynamic components was Flaw Tolerance, because it represented a practical and achievable means to achieve the cost, weight, and maintainability requirements of the program, and is compliant to the rule.

Full-Scale Fatigue Test Program

As of this writing the S-92 full-scale fatigue substantiation is not completed, however at least one example of each component has been tested in the "as-manufactured" condition (i.e., no deliberate flaws added), which was necessary to establish flight test limits and to provide the analytical correlation. In addition, many components have been tested with Barely Detectable Flaws, and a few with Clearly Detectable flaws. Four of these components are shown here as specific examples of the application of Flaw Tolerance.

It is also worth noting here that the objectives of a full-scale fatigue test program include obtaining actual cracking modes on every fatigue-substantiated part. This is accomplished by extending the test cycling or increasing the test load as necessary, even well beyond the predicted flight loads with margin. Runouts (non-fractures) cannot always be avoided, and still may be high enough to show a good retirement time, but they are not a full measure of the actual strength of the part. Obtaining a crack determines the actual component strength, quantifies the available margin, allows for the inevitable loads growth, correlates with design analysis, and provides an opportunity to evaluate crack growth characteristics. For a Test Engineer, initiating a fatigue crack is a success, not a failure.

Imposing Flaws on Full-Scale Parts. The approach used for the S-92 was not different from a conventional safe-life substantiation, except for the presence of flaws in critical areas. "Equivalent Flaws" were used, based on the results of a coupon test evaluation which compared the strength-reducing effects of the various types of flaws in the hazard assessment. For the "mechanical" type of flaw (scratches, gouges, notches, dents, impacts), a gouge which displaced material was found to have the greatest effect. A method was developed which provided a controllable gouge of the desired depth. This was accomplished by pressing the squared end of a drill rod into the part at a 45° angle with a force sufficient to produce the desired flaw depth, Figure 6.

Figure 6. Method for imposing a gouge, Main Rotor Yoke.

Not only did this method produce a feature with the desired depth and with a sharp stress concentration, it left areas of tensile and compressive residuals due the displacement of material, Figure 7 (From Reference 5).

The coupon shown in Figure 7 can be seen to be cracked at the gouge flaw. The crack originates at the corners of the "smile", where the tensile residual stress is highest.

Figure 7. Titanium Coupon with .040" Scratch and .040" Gouge.

Corrosion was the second type of flaw imposed on full-scale parts. The corrosion pits were simulated by using a drill point to produce a flaw at the correct location and nearly the desired depth. The hole was then electrochemically etched with salt water to achieve the final desired depth. This process produced the same rough surface and intergranular attack as natural corrosion. These flaws are indistinguishable from actual corrosion but are much more repeatable and controllable.

Test Results – Main Rotor Rotating Scissors Bracket. This titanium component, shown in Figure 8, provides the upper attachment points for the two main rotor rotating scissors assemblies. Scissors load is reacted by the lugs on the bracket in shear and bending. The bench test facility tests and loads the two scissors assemblies simultaneously, but they are substantiated separately. A cracking mode was obtained on the as-manufactured bracket in a radius of a mounting hole counterbore. No working curve intercepts occurred using this strength ("infinite" calculated life).

Barely Detectable gouge-type flaws were imposed in critical areas, including the counterbore radius, Figure 9. A fatigue crack was also obtained on the BDF specimen, as shown in Figure 9. The crack can be seen to pass through the BDF, and metallurgical analysis later confirmed that it actually originated at the BDF.

Figure 8. Rotating Scissors Upper Attachment Test Set-up.

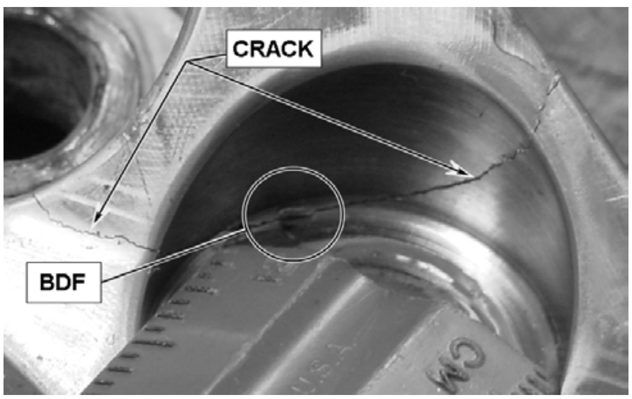


Figure 9. Rotating Scissors Bracket Flaw and Fatigue Cracking.

A comparison of the one as-manufactured and the one flawed specimen showed no difference, Figure 10.

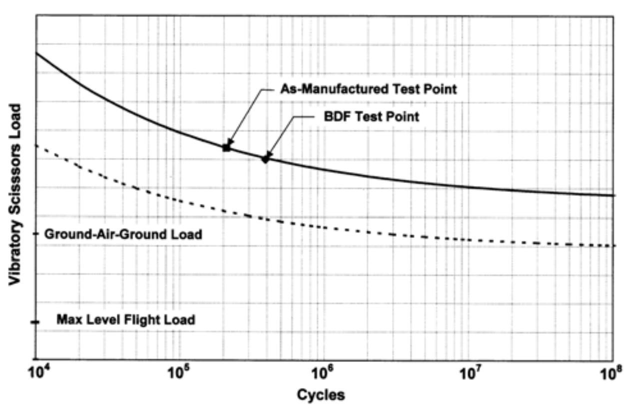


Figure 10. Rotating Scissors Bracket S-N Curve.

Test Results – Tail Rotor Pitch Change Shaft. This stainless steel component provides the tail rotor pitch input to the pitch beam (spider) from the control servo and the rotating adapter. Most of this component remains inside the tail gearbox, so it was approved by the FAA to be substantiated as a conventional safe life component. It is tested with the pitch beam in a "nutating" load arrangement with a dummy gearbox housing as shown in Figure 11.

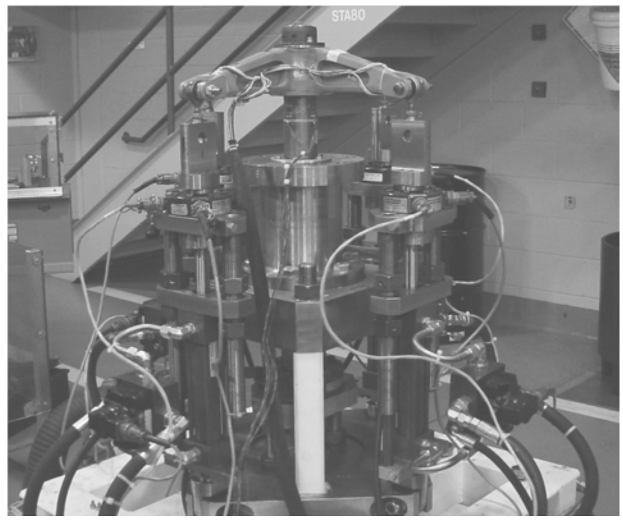


Figure 11. Tail Rotor Rotating Controls Test Set-up.

The loading in the shaft is axial and bending. A good safe-life was obtained for this component with the cracking mode found to be at an internal machining detail at the outboard end of the shaft. However, because some of this shaft is exposed, an evaluation was conducted with a Clearly Detectable gouge flaw imposed on the outside diameter opposite the origin site previously demonstrated on the inside diameter, as shown in Figure 12.

This is a gross and startling flaw. It has a depth of .040", is about .25" wide, and can be detected by touch as well as visually. There is no question that it would be detectable in a pre-flight inspection, so it's use here as remaining in place for a directed inspection interval is very conservative.

Testing with this flaw resulted in the same cracking mode and the same fatigue strength as was obtained for the as-manufactured specimens. This result was not expected, but it demonstrates that the stress concentration of the internal machining detail has a stronger effect than the

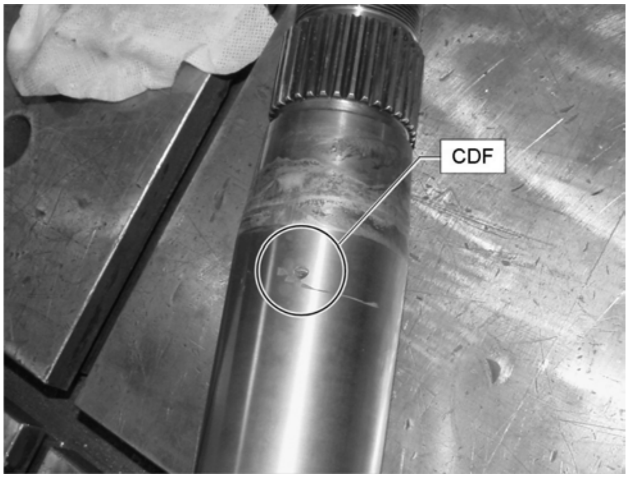


Figure 12. Clearly Detectable Flaw Imposed on Pitch Change Shaft.

imposed flaw on the outside diameter. Flaw Tolerance is therefore demonstrated with this component. A simple un-aided close visual inspection of the shaft outside diameter will be recommended to be included in the scheduled 1250-hour general inspection of this aircraft. This type of inspection is already a standard practice.

Test Results – Main Rotor Damper Bracket. This titanium component provides the inboard attachment of the main rotor damper to the main rotor hub, shown in the test set-up, Figure 13. It is simply loaded by axial damper load applied at its extreme angle. Testing on the as-manufactured specimen resulted in a runout (non-fracture) even though the loads were increased to highly elevated levels (Figure 14). An "infinite" calculated life is obtained for this point since there are no GAG or flight load intercepts on the associated working curve. (This working curve is not shown in the figure to reduce crowding.)

Testing with Barely Detectable Flaws included the .005" deep gouges applied in critical areas, and also the deliberate omission of a protective coating on the lug bores associated with a cold-working process. The rationale for including this is that it had been observed that the coating could be scratched or abraded during assembly, and also that the coating was wearing and flaking in some other fatigue tests. Discrepant coating was therefore considered a flaw, and since it could not be inspected after initial assembly, it was treated as a Barely Detectable (i.e., undetectable) Flaw. Two fractures at the lug bore, with chafing (fretting), have been obtained in the BDF testing so far.

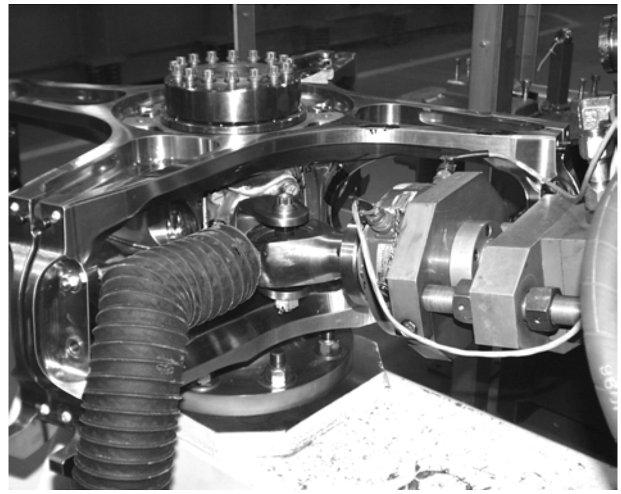


Figure 13. Main Rotor Damper Bracket Fatigue Test Set-up.

These are also shown in Figure 14, and indicate at least (comparing fractures to a runout) a 22% reduction in strength. The working curve associated with these points also has no intercepts (This working curve is also not shown in the figure.). The Flaw Tolerant Safe Life would therefore be listed as "unlimited" based on this data.

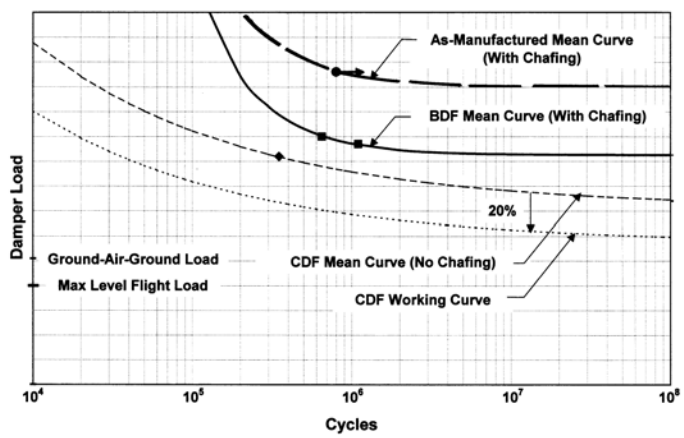


Figure 14. Comparison of Damper Bracket fatigue Strengths.

.040" deep Clearly Detectable gouge flaws were added to critical areas on one specimen as shown in Figure 15. Again it can be seen how these flaws are a conservative representation of the concept of a clearly detectable flaw.

S-N testing of this component resulted in a fatigue crack originating at a flaw in the flange radius, as shown in Figure 16.

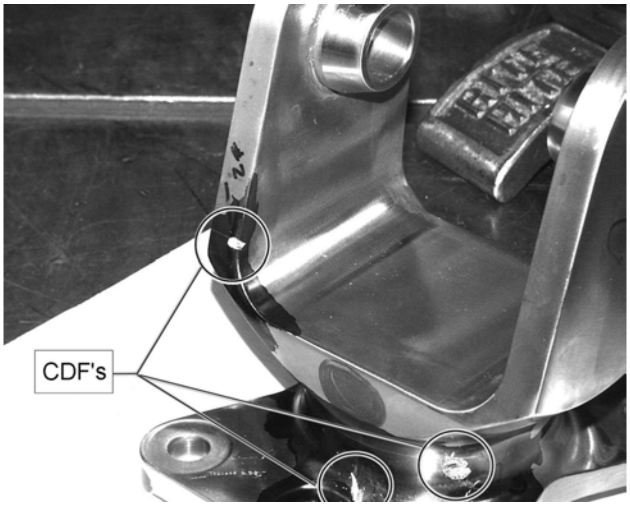


Figure 15. Clearly Detectable Flaws Imposed on the Damper Bracket.

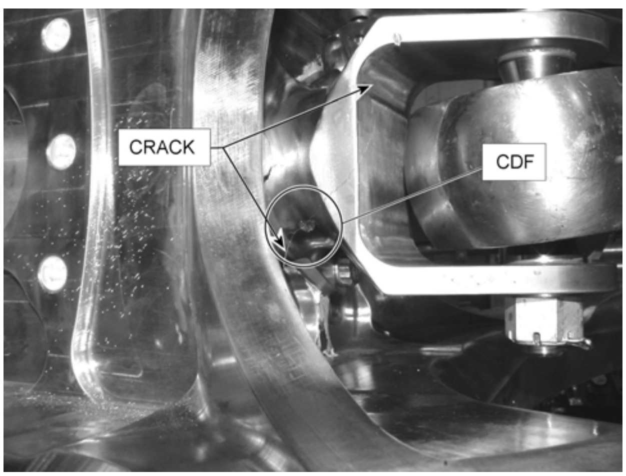


Figure 16. Damper Bracket Fatigue Crack Origin at a Clearly Detectable Flaw.

Comparison of fatigue results for the CDF case is also shown in Figure 14. The CDF data point is an open-section or non-chafing origin, and therefore has a different standard curve shape than for the fretting case. The strength-reducing effect of the CDF is a minimum of 38%, taken at 10^8 cycles. The coupon program had indicated the possibility of a 57% reduction, but the effect is expected to be less on the full-scale parts. The calculated safe life, and therefore the inspection interval, based on the CDF and a 2-sigma working curve, is again infinite. It can be seen however, that only a modest increase in damper load, from additional flight testing or perhaps a growth situation, would result in a finite inspection interval. The initial recommendation would be to recommend a simple close visual inspection of the damper brackets at the 1250-hour general inspection.

Test Results – Main Rotor Hub. An analysis of the expected behavior of the main rotor hub is included in References 3 and 4, which relied entirely on predicted strength. Fatigue testing has been accomplished since then on an as-manufactured hub, and these results can now be used to improve the conclusions for this component.

This titanium component, shown in Figure 17, is the centerpiece of the S-92 Flaw Tolerant design approach, because of its critical function and cost of manufacture. It is intended and fully expected that only one MR Hub will be needed for each aircraft over its lifetime. It is machined all over from a single forging, has an open design for ease of inspection, and is easily repaired.

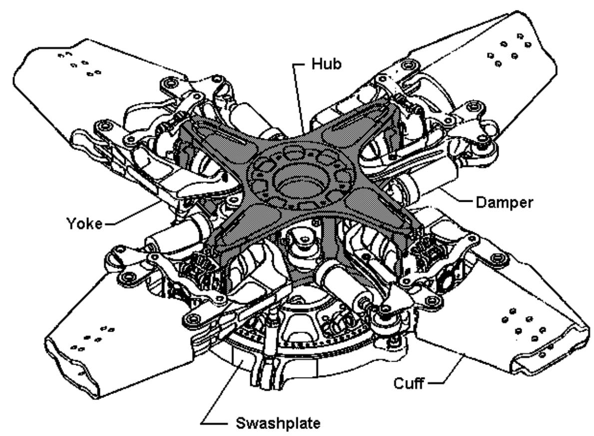


Figure 17. S-92 Main Rotor Head.

The loading on the hub is complex, and is simulated in the test facility by means of a nutating flapping/lead-lag motion that includes full centrifugal and thrust loads. On the first as-manufactured specimen, a cracking mode was induced at the runout of the lower arm radius, Figure 18. This was a non-chafing mode at an analytically critical location. A conventional safe life of over 44,000 hours is calculated for this mode, which would result in an unlimited recommended retirement time.

When the Barely Detectable Flaws are added to the critical locations, we would now expect that little

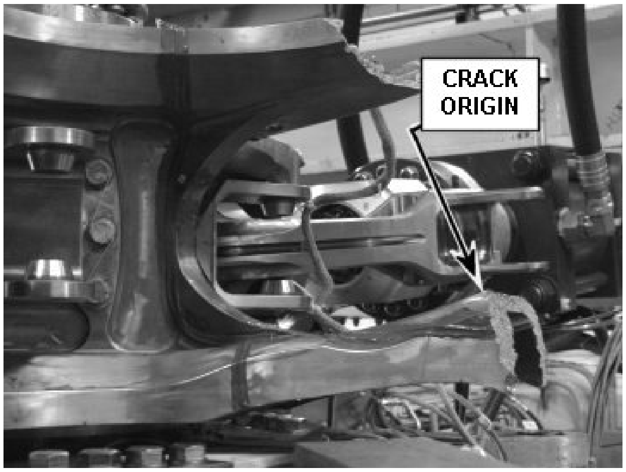


Figure 18. Main Rotor Hub Cracking Mode.

or no effect would be found, allowing the hub to be substantiated by Flaw Tolerant Safe Life with the unlimited retirement time. It is intended to also conduct testing on at least one hub with Clearly Detectable Flaws in critical locations. Using the very conservative assumption that the full 57% reduction from the test mean strength will occur with CDF's imposed on a full-scale hub, the S-N curve shown in Figure 19 would result.

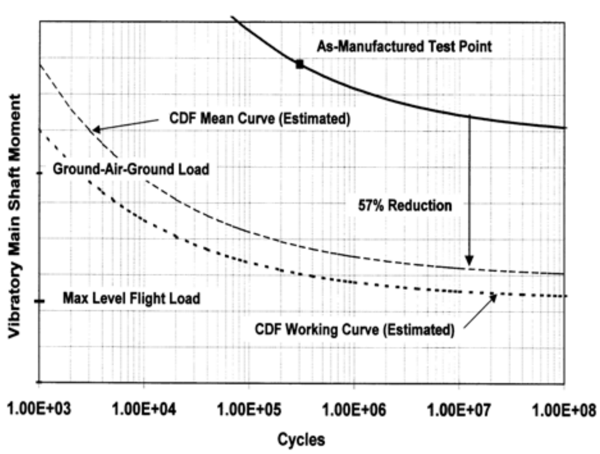


Figure 19. Main Rotor Hub S-N Curve.

Figure 19 shows the use of a 2-sigma working curve on the flawed mean strength, and a safe-life calculation on this working curve indicates a 100-hour inspection interval. This relatively low number results from the use of the overly conservative coupon-based CDF reduction factor, combined with the preliminary unconstrained flight loads data available at the time of this writing. It is fully expected that the final flight and full-scale fatigue data will allow this to be increased to at least the 1250-hour goal. In any case, the inspection itself would consist of a simple unaided close visual examination which would only take a few minutes.

Discussion

The mechanical Barely Detectable Flaws used to establish a retirement time turned out to have little or no effect. This was indicated in the coupon program and verified in the full-scale tests. This event can be attributed to the fact that the effect of stress concentrations (flaws) is generally less at lower stress levels used in the S-92 design. This same result would not be expected if these flaws were imposed on existing components that are more highly stressed by design. One BDF – lack of a protective coating in a lug bore – did indicate a significant strength reduction, but it did not reduce the replacement time of that particular part below "unlimited".

Few of the fatigue modes obtained were related to the imposed BDF's. The modes obtained were related to fretting (considered a flaw on its own), or design details in areas not susceptible to flaws. The discussion over the use of 2-sigma or 3-sigma working curves turned out to be a moot point for the establishment of a life, since it had been agreed that the conventional safe life (3-sigma) would be used if it was less than the Flaw Tolerant result. The stress is so low on these parts that unlimited life (greater than 30,000 hours for the S-92) is easily achieved on most dynamic components.

The establishment of inspection intervals based on Clearly Detectable flaws, although not a Rule-based procedure at this time, does appear to be working as expected. The strength is dramatically reduced with these large flaws, but the inspection intervals determined from the testing are still reasonable considering that the inspections generally require no disassembly and the critical flaws are easily seen without special equipment or training. Future applications of Flaw Tolerance will be able to establish mandatory inspections in accordance with the Rule and Advisory Circular changes that are in review.

Potentially then, the emphasis in future Flaw Tolerance substantiations could shift from the life determination aspect to the inspection aspect. Both a retirement time and an inspection program will be required by the pending new rule. However, the implementation of high-payoff, directed, and focused inspections will have a greater impact on safety than conservative retirement times will, as long as they do not become a burden for the operator. This idea – inspections add safety – is well founded and proven in fixed wing and airframe applications of Damage Tolerance. We can achieve the same benefit in helicopter dynamic components by using practical Flaw Tolerance–based inspections. The first step in making this change would be to change the mix of full-scale fatigue test specimens assigned to the substantiation program. We should assign several specimens, with a wider variety of flaw sizes and types, to the testing to determine inspections, and use only one specimen to determine the retirement time. This is the reverse of the current practice.

This change will also help to address one of the criticisms of Flaw Tolerance, namely that all of the types and sizes of flaws that could possibly occur in service far exceeds the number that can be evaluated by full-scale test. The very conservative treatment of this question on the S-92 (very conservative flaw sizes and the selection of worst-case flaw types, applied to all critical areas) provides confidence in the answers obtained for that aircraft. However, if a wider variety of flaws and flaw sizes were evaluated, longer intervals and better focus could be obtained. Inspection intervals and procedures could be tailored to the specific threats in the hazard assessment. For example, all of the Clearly Detectable Flaws in the current S-92 fatigue program were assumed to be .040" deep, because flaws of this size were found on fielded S-76 rotor parts, including aluminum parts. Titanium parts however, will never see flaws that large because the tremendous energy and force needed to cause them is just not available in service. A more extensive inspection interval test program could have included more flaw sizes and provided longer times with smaller flaws in titanium.

Conclusions

1. The S-92 requirements for safety, cost, weight, and maintainability of dynamic components were validated using Flaw Tolerance methods.

2 The mechanical Barely Detectable Flaws used to establish retirement times have had little or no effect on the strength of S-92 dynamic components.

3. Flaw Tolerance with Clearly Detectable Flaws provides a method for establishing a safe and efficient inspection program, based on the limited data available at this time.

4. Future applications of Flaw Tolerance should shift emphasis from the determination of retirement times to the determination of inspection methods and intervals.

5. Practical inspection intervals and retirement times could not be obtained using Damage Tolerance methods on S-92 dynamic components, especially those with combined high-cycle and low-cycle loading.

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Acknowledgements

The authors wish to recognize and acknowledge the other people at Sikorsky who made significant contributions to the success of Flaw Tolerance on the S-92. They include Dave Hunter, Dave Zinni, Paul Inguanti, Gregg Ambrose, and Bob Young.

Also recognized are the contributions of the Test Engineers who produced the data used in this paper – Hicham Sofiane, Steve Ferris, and Steve Joyce.