AGING AIRCRAFT - USAF EXPERIENCE AND ACTIONS

John W. Lincoln*

In the seventies and eighties the United States Air Force performed damage tolerance assessments of their major aircraft systems to establish inspection and modification programs to maintain their structural integrity. These airplanes have now become older and in many cases there are indications of aging. This means in some cases their maintenance programs may need modification. This need arises because the airplanes have experienced one or more of the following problems: operations beyond the design service life, corrosion, onset of widespread fatigue damage, or repairs. It is the purpose of this paper to discuss the results and implications of some of the recent reviews of aging aircraft and describe the actions taken to ensure their safety and continued economical operations.

INTRODUCTION

The United States Air Force (USAF) adopted the damage tolerance methodology in the early seventies as documented in (1). As indicated in that paper, the damage tolerance assessments (DTAs) of older aircraft took place in the twelve year period starting in 1972. This effort required over one million hours of effort by engineers and technicians to generate a maintenance plan based on damage tolerance principles for all the major weapon systems. Figure (1) shows the approach used in the assessments and Figure (2) identifies the aircraft reviewed. Operational experience confirms its success. The failure rate for all USAF aircraft resulting from structural problems is less than one in ten million flight hours.

In the future, however, the results of these DTAs may not adequately protect aircraft safety. This could happen when airplanes fly beyond their design service life and open the possibility of introducing new critical areas. Corrosion could affect the inspection intervals through the acceleration of crack growth. The onset of widespread fatigue damage (WFD), causing loss of fail-safety, will require the

* Aeronautical Systems Center, Wright-Patterson Air Force Base, Ohio

structure be modified to prevent failure. Repairs typically reduce the inspection intervals in the affected areas and may introduce new critical locations. The USAF classifies an aircraft as aging if any one of these four events occur to the extent they must modify the maintenance plan. They observed these events may occur in combination. For example, corrosion could shorten the time to the onset of WFD. The case histories discussed below illustrate the specific approach utilized in dealing with these aging issues.

F-16 REVIEW

The F-16 is an aging aircraft because of operating beyond its original design life. As a consequence the USAF has discovered many new critical areas as reported in (2). The actual hours on these airplanes are below the original 8,000 hour operational life, but the equivalent damage hours in many of them are in excess of this number. There are several reasons for this. First, the use of load limiting in the F-16 allowed the pilot to maneuver close to the aircraft limits without concern about overload failure. This increased the severity of the all missions, but particularly the air-to-ground mission. Second, the mission of the aircraft changed from predominantly air-to-air to predominately air-to-ground. Finally, the miniaturization of electronics permitted higher density electronic packaging and consequently increased aircraft mass. This increased usage severity and increased mass caused cracking in many locations. The USAF had not previously recognized these as being critical based on the original DTA and durability testing performed in the mid seventies. They found most of the early cracking problems in the wing attachment bulkheads. More recently, they found cracking in one of the bulkheads supporting the vertical tail. The location of the cracks in operational aircraft was the Fuselage Station 479 bulkhead. Figure (3) shows the upper portion of this bulkhead. Figure (4) shows the details of the attachment pad wherein a stress concentration causes cracking. The cracking problems in this area were difficult to assess because the USAF had not determined the vertical tail external load spectrum with sufficient accuracy. There was also uncertainty in the internal stresses in the vertical tail support bulkheads. Unfortunately, the USAF did not require instrumentation on the static or durability test articles to validate the existing finite element analysis. To some degree this lack of an adequate data base was the consequence of the pressure to produce these airplanes. This pressure considerably shortened the engineering effort normally required to perform these tasks adequately in the Engineering and Manufacturing Development phase of acquisition. Also, both the contractor and the System Program Office (SPO) placed considerable emphasis on maintaining production rates and operational performance.

In 1995 the USAF chartered an independent review team to assess the cracking found in the Fuselage Station 479 bulkhead. This team was to provide the

F-16 program office an evaluation of the actions taken to minimize risks and ensure these actions adequately addressed flight safety. The first meeting of the independent review team occurred on 29 June 1995 to get briefings from the F-16 System Program Office and Lockheed Martin on this subject.

At this meeting the review team learned that three bulkheads located approximately at Fuselage Stations 446, 462, and at 479 support the vertical tail. The USAF identified in-service cracking in the Fuselage Station 479 bulkhead in 1993. They used local machining to modify the stress concentration and inspected them with eddy current on a recurring basis. This modification was not as successful as hoped since the USAF found cracks within approximately 300 hours from the time of modification. The USAF accomplished this modification on approximately 800 pre-Block 40 aircraft. The USAF believed there is a potential for cracks in the additional 1200 pre-Block 40 aircraft (474 USAF) that have not been subject to an inspection. In March of 1989, an F-16 operating out of McConnell Air Force Base suffered a partial failure of the 479 bulkhead. The aircraft returned safely, but part of the fin and the rudder were missing. The USAF has been unable to explain the reason for these missing portions of the structure. The fracture surfaces do not show evidence of a midair collision and the tail failure direction was wrong for a wake turbulence induced load. Further, for a given sideslip angle, the strength of the upper portion of the tail at the failure location is approximately four times larger than it is at the root. Also, they found no significant change in the calculated flutter speed as a result of the change in stiffness of the partially failed vertical tail support.

The vertical tail loads have gone through several iterations since the start of the program. For the F-16C/D Block 30 durability testing, the USAF used the bending moment exceedance function shown in Figure (5). The Block 30 durability testing started on 1 September 1987 and completed on 22 March 1993. Figure (5) also shows the current usage exceedance function. This is a remarkable difference between these two functions when one considers the airplanes have been in service since the middle seventies. In September of 1994 the USAF authorized the development of a vertical tail load spectrum that was more representative of operational usage. Lockheed Martin developed the spectrum from a sample of 500 hours of Block 30 Crash Survivable Flight Data Recorder data. They did this using vertical tail load regression equations developed from Block 40 flight test data. The USAF acquired this 500 hour sample prior to October 1993 when they placed rudder usage restrictions in the flight handbook. Also, they chose this 500 hour sample from flights that had very severe lateral load factors. This data showed vertical tail root bending moments could be higher than the Block 30 design of 84,980 newton-meters. Investigation of larger data bases revealed fewer occurrences per lifetime, but there were still high maximum bending moments. The contractor determined that rolling maneuvers with rudder input from the pilot caused the high loads. The USAF then acquired 7,834 hours of Block 30 data for operations subsequent to the new handbook rudder usage restrictions. This data indicated the loads were slightly less severe than the earlier data.

The finite element modeling for the initial qualification of the vertical tail bulkheads was inadequate to determine the internal load distributions with the desired accuracy. They did not properly determine the stresses in any of the individual bulkheads nor did they accurately determine the distribution of loads between the bulkheads.

The contractor had completed component testing with both the original 2024-T8 aluminum bulkheads and an Alcoa aluminum lithium alloy designated as 2097. This relatively new alloy demonstrated a longer life in these tests than the original 2024-T8 conventional aluminum alloy. The reason for this is the aluminum lithium alloy shows improvements in both crack growth rate and fracture toughness over the conventional alloy.

As indicated above, inspections have revealed significant cracking in the vertical tail to Fuselage Station 479 bulkhead attachment pads for the majority of pre-block 40 airplanes. The Ogden Air Logistics Center has considerable experience in the eddy current inspection for these cracks and has established appropriate guidelines for the disposition of cracks found in operational aircraft. As indicated above, the root cause of this cracking is the stress concentration inherent in the design compounded by increased usage severity and mass increases in the aircraft without compensating structural modifications. The stresses at the point where the cracks emanated are undesirably high although they do decrease with increasing crack depth. Figure (6) is the elastic stress distribution through the bulkhead starting from the point of initial cracking.

Normally, the USAF would not fly these airplanes with known cracking. In this situation the USAF concluded the structure had adequate fail-safe capability in the event of a first member failure. The contractor has analytically shown the vertical tail structure has fail-safety in the event of a failure of the Fuselage Station 479 bulkhead. The USAF, however, has not demonstrated this by laboratory testing. This would be difficult because of the complexity of simulating the failure and fracture surfaces as they may be in an operational aircraft. Therefore, USAF developed the recovery program with the goal of preventing another in-service failure. One of the concerns about too much reliance on fail-safety is there may be cracking in the adjacent bulkheads that would degrade the fail-safe capability. Even though the USAF placed emphasis on preventing the first member failure, they must ensure the integrity of the adjacent bulkheads through inspections. This is an example of potential WFD with a single crack in the fail-safe load path.

At the request of the review team, Lockheed Martin showed the aluminum lithium bulkhead had successfully met the five element technology transition guidelines (3). Therefore, the review team recommended this material be used for the Fuselage Station 479 bulkhead replacement. There is a lingering concern about the cost of these bulkheads because of the potential for a high rejection rate resulting from failure to meet the processing requirements. Therefore, it may turn out more economical to use conventional aluminum if the USAF can not control the aluminum lithium cost to an acceptable level. The USAF placed emphasis on replacing the existing cracked bulkheads with new bulkheads to maintain a low risk of failure of the Fuselage Station 479 bulkhead. The replacement program, based on the Lockheed Martin crack growth predictions, should be able to define a replacement program that maintains an acceptable risk of bulkhead failure. The review team recommended the fail-safe test not be conducted based on the assumption the replacement program was logistically viable.

The review team found Lockheed Martin's newly developed computer codes were doing a credible job in establishing an adequate simulation of the rolling maneuvers. This enabled them to establish appropriate wording in the pilot's handbook that will define the limits of the pilot's use of rudder. Subsequently, Lockheed Martin could determine limit load based on this wording. They could then make the appropriate calculations to ensure there is adequate strength to accommodate this new limit load.

707 REVIEW

The Boeing Company derived the 707, 720, and KC-135 aircraft from the prototype designated as the 367-80 developed with their own funds. The 707 proved to be a very successful aircraft for the Boeing Company as well as the operators. The USAF has successfully used the 707 for several important programs. The most widely known of these has been the E-3A. For these airplanes, the USAF contracted with Boeing in the early seventies to modify the 707-300 by placing a large rotating antenna over the rear fuselage. They procured these airplanes new. Boeing modified the structure to accommodate the antenna during production. The USAF bought in excess of thirty of them. At the time of the initial contract with Boeing, the USAF had committed to a shift in structural philosophy to damage tolerance. At this time Boeing had not performed a damage tolerance assessment of the commercial version of this aircraft. Boeing did this assessment later to remove the life limit of 60,000 flight hours placed on the aircraft by the airworthiness authorities in the United Kingdom (4). The USAF contracted with Boeing for a damage tolerance assessment of E-3A aircraft. They did this for the E-3A missions - radically different from the commercial 707 mission. Boeing successfully completed the E-3A damage tolerance assessment in the mid seventies and these airplanes have been in operation since then. The USAF has found no major problems with them although they are now showing signs of corrosion damage.

During the time when the E-3A damage tolerance assessment was underway, Boeing took advantage of an opportunity to perform a teardown inspection on a relatively high time aircraft. This opportunity came when terrorists severely damaged Trans World Airline aircraft number N761TW - a 707-320. This aircraft had 44,421 hours and 13,655 flights. The inspection performed by Boeing, completed in 1976, revealed numerous cracks in the aircraft. The cracks that caused the most concern were in the lower wing splicing stringers and the large stringers around the lower wing inspection holes adjacent to the splicing stringers. This paper refers to these as the large adjacent stringers. Boeing evaluated these inspection results and reported their findings in (5). Boeing published several Service Bulletins as a result of these wing crack findings. These Service Bulletins called for either a high frequency eddy current inspection inside of the wing or an external low frequency eddy current inspection. These inspections have revealed major damage including a severed stringer and skin cracks in excess of 44 millimeters.

The USAF found another use of the 707 for the Advanced Range Instrumentation Aircraft (ARIA) in the mid eighties. They called this aircraft the EC-18B. Since they needed only a few of them, they opted to procure airplanes that were currently in operational commercial service. They selected aircraft that had approximately 35,000 flight hours. Therefore, they were relatively young when compared with many of the airplanes in the commercial inventory. At the time of procurement of these airplanes, Boeing had completed the Structural Supplementary Inspection Document (SSID) for the commercial 707. Therefore, the USAF contracted with Boeing to perform an analysis to get the ratios of usage severity between the commercial and the military usage. They then used these results to modify the inspection intervals that are in the existing SSID. This initiative gave the USAF an adequate inspection program at the least possible cost.

The USAF elected in the eighties to use the 707 aircraft for Joint Stars (Joint Surveillance Target Attack Radar System). There was considerable discussion over whether to procure new or used airplanes for this mission. The USAF finally chose to procure new aircraft, but before they were able to get their funding Boeing closed the 707 line. This action forced the USAF to procure older aircraft. When Northrop Grumman, the contractor for Joint Stars, selected the aircraft, the configuration was the primary concern - not the age. Figure (7) shows the number of flight hours and flights on the first ten airplanes selected for this program. As seen in this figure, these airplanes were close to or above the original life goals of sixty thousand flight hours and twenty thousand flights established by Boeing for the 707. At the time Northrop Grumman selected these airplanes, the nondestructive capability for the detection of corrosion damage was marginal at best. Consequently, the USAF found that corrosion in some of the aircraft procured for this program was significantly more severe than their expectations. For the fuselage, the corrosion is quite extensive. The contractor has aggressively attacked this problem. For example, they have removed the fasteners from every longitudinal lap splice. They removed the corrosion found in these splices through grinding or part replacement. Many of the fuselage skins and stringers have required complete replacement because of corrosion damage. The contractor has used modern corrosion protection procedures in the refurbishment of the fuselage. For the wings, there is a major corrosion problem around the fasteners attaching the wing skins to the wing inner structure. They have ground out these areas to remove the corrosion damage. In some cases the number of fastener locations that has been subject to grinding has been so extensive that Boeing performed static strength analyses of the wing to ensure their integrity. Boeing worked with the contractor to help them to determine modification actions to restore the lost strength. On some aircraft Boeing has recommended the cold expansion of the order of fifteen thousand fastener holes that have been subject to extensive grinding. The USAF has given the contractor direction to perform the modifications needed to correct the corrosion problems. Where possible, the contractor made repairs according to the Boeing Repair Manual. Typically, the Boeing Repair Manual does not cover from one to two hundred repairs performed on each of the airplanes. Some of these are in structure that is flight safety critical. To incorporate damage tolerance in these repairs the contractor has had to perform extensive in-house external and internal load assessments to generate a basis for the fracture analysis of these details.

The largest concern about the structure of this aircraft, however, is the potential for the degradation of fail-safety because of WFD in the wing. The USAF believes fail-safety is an extremely important asset to any aircraft. The failure of the two deHaviland Comets in the mid-fifties (6) graphically demonstrated the lack of capability of this structure to maintain its integrity after a single member failure. These failures led to the use of fail-safe designs to ensure safety of flight. Fail-safety is also important for reducing the maintenance burden of inspections. There are thousands of details in areas of an aircraft that would be extremely costly to inspect for cracks if the structure was not fail-safe. Such details include longitudinal and circumferential splices in the fuselage and fastener locations in the lower wing skin. Further, it is extremely doubtful an inspector could accomplish these inspections efficiently because of the boredom of such a task. It was for these reasons the USAF incorporated the commercial aircraft standard for fail-safety in the C-17 during its design.

Many studies show, however, that even small cracks can significantly degrade fail-safety (6,7). The Aloha Airlines 737 incident in April 1988 was misleading about the severity of this problem. In that case, the fuselage failed at the time it had accumulated 89,680 flights. This aircraft flew to the point of link up of the fastener hole cracks in the lap splices. The FAA determined, subsequent to the failure, that the airplanes had a much earlier degradation of the fuselage's capability to sustain discrete source damage. After approximately 40,000 flights, for those airplanes where the lap splice bond had failed, there was evidence of significant fail-safety degradation. It is essential, therefore, that the analyst perform the appropriate analyses and tests to determine when cracking will degrade fail-safety below acceptable limits. This is the time of the onset of WFD. The structural analyst must identify this onset time so the operator can make modifications to eliminate the problem.

When WFD occurs, inspections derived from the DTA will normally not reveal its presence (8). It is therefore essential that the structural analyst make a prediction for the time of the onset of WFD. The most difficult part of this prediction is the determination of the fatigue crack distribution at a specific number of flight hours and flights. The analyst can determine the crack distribution function from a teardown inspection of a questionable region of the aircraft. A procedure for finding the cracks in a structure from this type of inspection is described in (9).

In previous teardown inspections of wings from 707 aircraft, Boeing has found evidence for the potential for WFD. They performed these teardown inspections on aircraft that had less time than the Joint Stars aircraft. Relative to the wing, the most important teardown inspection performed by Boeing was on the TWA aircraft as described above. That data base, however, was not definitive enough to be usable in an assessment of the risk of failure. Consequently, the USAF contracted with Boeing to examine higher time aircraft parts taken from retired aircraft at Davis Monthan Air Force Base to quantify the risk associated with WFD.

Boeing performed teardown inspections on two 707 wings from aircraft taken from Davis Monthan Air Force Base. One of these airplanes was a 300 series aircraft, representative of the Joint Stars aircraft, that had experienced 57,382 flight hours and 22,533 flights. Boeing delivered this aircraft to Pan American in 1966. They operated it until 1977 when they sold it. The aircraft operated in Africa until Boeing purchased it for the USAF in 1990. The second aircraft was a 100 series aircraft that had experienced 78,416 flight hours and 36,359 flights. The reason for the selection of the high time 100 series airplane was to ensure they would find an aircraft that had experienced some degree of cracking. The USAF could use this data to make projections for cracking for future Joint Stars operations. They performed the teardown inspections on the wing lower surface and the wing stringers. Stringers and skins where Boeing used steel fasteners contained most of the cracks found. This was typically in the area of the wing skin splices and the large adjacent stringers. The large adjacent stringers rest on lands machined into the skins. The skin at the lands is typically approximately 9.5 millimeters thick and the wing skin under the splicing stringers is approximately 4.0 millimeters thick. Figure (8) shows a plan view of the 707 wing. This figure indicates the location of the production joints and splicing stringers. Figure (9) shows a cross section of the wing with the splicing stringer and the adjacent stringers. The beneficial effects of the aluminum rivets attaching the other stringers to the wing skins apparently reduced the amount of cracking there. There was, however, some cracking found in these locations.

Boeing found that cracking in the aircraft in the area of the steel fasteners was quite extensive. In the 300 series aircraft there was a total 1915 cracks found in the five sections removed from the aircraft shown in Figure (10). They found the 100 series aircraft also extensively cracked. However, the cracking found in the 300 series aircraft was sufficiently extensive to perform an assessment of the risk of failure. Most of the cracks found in the 300 series aircraft were small. However, they found a significant population cracked to the point of considerable concern. Figure (11) shows the cracks in the stringers in one of the sections removed from the aircraft. The circles indicate crack findings. They found that increasing the size of the holes in the splicing stringer and the large adjacent stringer would not remove all of the cracks. About twenty percent of these holes would still have stringer cracks. Further, they found significant cracking outboard of the Wing Station 360 production joint. Therefore, the problem involved most of the wing. Typically, the large adjacent stringers had more large cracks than the splicing stringers. The largest crack found in Stringer 7 was approximately 38 millimeters in length. Figure (12) shows the fracture face of this crack. It was near the point of rapid fracture. Such a crack in Stringer 7, since it is a large adjacent stringer with a thick land, would be difficult to detect with low frequency eddy current from the outside of the wing. An inspector, however, would likely detect a completely severed stringer by low frequency eddy current. It would not be readily detectable from the inside of the wing by high frequency eddy current since the crack initiated on the faying surface. There were, however, many cracks found that would have gone to failure in the planned life span of the Joint Stars aircraft. There was a concern about cracking that would degrade the capability of the structure to sustain discrete source damage. There was also a concern about the fatigue failure of the stringers and subsequent catastrophic loss of the aircraft after a skin failure. Figure (13) shows the largest skin crack found in the teardown inspection.

The USAF made the decision to assess the Joint Stars structure in the same manner as they used in addressing WFD problems on the C-5A and on the C-141. For these airplanes the USAF used probabilistic methods to determine their risk of catastrophic failure. The procedure for accomplishing this, described in (10), considers the crack length distribution and the stress distribution as random number sets. The procedure assumes the crack growth function and the stress that would result in rapid fracture for a given crack length are deterministic.

Boeing calculated the stress intensity of each of the cracks found. They then determined for each of them the size of the corner crack with the same stress intensity. From these cracks, the USAF derived the crack distribution function. They used a population taken from the largest of them to approximate the crack distribution with a two parameter Weibull distribution function. It is typical that a single Weibull distribution function will not approximate the longer cracks as well as the shorter cracks. This is not a problem since only the longer cracks will have a significant effect on the risk of failure. Figure (14) shows the crack distribution functions for the splicing stringers and the large adjacent stringers.

The USAF needed two stress distribution functions for the assessment. The first is the stress distribution function for the intact structure. Boeing derived this in the usual manner from the intended usage of the aircraft, the external load analysis, and the stress analysis of the wing. Second, for the cases where discrete source damage was present they determined the local stress increase from the damage. In many cases the local stress increased to the point where there was significant plastic deformation of the structure. When this occurs it is essential the plastic deformation be included in the analysis. A linear analysis in these cases would likely lead to serious errors in the determination of risk.

For this study the USAF considered two discrete source damage scenarios. The first was the assumed failure of a splicing stringer and failure of the skin on each side to the next stringer. The second was the failure of the large adjacent stringer and the skin on each side to the next stringer. For these cases the resulting stresses were so high the finite element program had to account for plastic deformations in the skin and stringers as well as the fasteners. Boeing performed this nonlinear analysis by using Nastran solution 66 with manual restarts for changing joint and fastener stiffness. They did this analysis in approximately fourteen steps from a zero intact structure stress to the 255 Mpa limit stress. The Boeing analysis showed the structure, with the discrete source damage, and no cracking in the structure, was barely able to sustain limit load. Figure (15) shows the stress distribution functions used in the analysis.

Figure (16) shows the crack growth functions for the splicing stringer and the large adjacent stringer. Figure (17) shows the stress that would result in rapid fracture for a given crack length (residual strength functions). Boeing determined these functions for the 7075-T6 stringers from material fracture toughness data with short crack regime corrections.

One aspect of the analysis that is somewhat subjective is the determination of the threat of the discrete source damage. The threat for a military aircraft must include battle damage as well as other sources such as engine disintegration. The USAF performed a risk assessment for the C-5A in the late seventies. For this assessment they made a judgment the risk of discrete source damage was 10^{-3} in a random single flight of the aircraft (8). For the 707 they reasoned this number should be somewhat less because the portion of the wing that would be susceptible to damage was smaller. Consequently, they selected 10^{-4} for this analysis as the discrete source damage probability of occurrence.

The risk assessment also requires a decision be made on an acceptable probability of structural failure. The USAF provides guidance for this in (11). In this document they considered a single flight failure probability of 10^{-7} as being acceptable. The USAF would not expect an aircraft failure in a given population in their planned lifetime with this single flight failure probability. They used this single flight failure probability in the assessment of the 707 for the Joint Stars mission. Therefore, for the cases of discrete source damage the maximum single flight failure probability allowed was 10^{-3} and for the intact structure the maximum single flight failure probability allowed was 10^{-7}.

The USAF performed the risk assessment initially for an aircraft that had the same number of flight hours, 57,382, as the aircraft subjected to the teardown inspection. They found, based on the criteria identified above, that the risk for the discrete damage cases was unacceptable. To make a determination when the risk initially became unacceptable, they transformed the crack distribution. They mapped the cracks found in a 57,382 flight hour aircraft to an aircraft that had 40,000 flight hours in commercial operation. Figure (18) shows the dependence of the risk on the number of Joint Stars aircraft usage flights when there is discrete source damage to the large adjacent stringer and neighboring skins. The discrete source damage case for the splicing stringer and neighboring skins is slightly less critical. The USAF found the aircraft would have failed the given criterion when it had the equivalent of 50,000 hours of commercial usage.

For the intact structure risk assessment, the more critical case was the failure from fatigue of the large adjacent stringer. Figure (19) shows the risk dependency on the number of Joint Stars usage flights from 57,382 commercial usage flight hours. This figure shows the risk is unacceptable after only a few flights of Joint Stars usage.

As indicated above, Boeing found significant cracking outboard the Wing Station 360 production joint. Therefore, any modification action must include the structure up to the production joint at Wing Station 733.

For the stated criteria for discrete source damage, the USAF found significant degradation of fail-safety beyond 50,000 flight hours of commercial usage. Therefore, for some aircraft, there will be unacceptable fail-safety degradation before the end of the planned 20,000 hours of Joint Stars usage. This will occur for Joint Stars aircraft with more than 36,000 commercial usage hours. Further, for the case of no discrete source damage, there will be safety degradation beyond 58,000 hours of commercial usage. Therefore, aircraft with initially more than 44,000 hours of commercial usage will have a high probability of failure before operationally flying 20,000 hours.

There are two possible approaches for solution of this problem. The first is to remove the steel fasteners in the area of concern in the lower wing surface and perform an eddy current inspection. If the inspector finds no indication of a crack or if increasing the size of the hole would remove the indication, then this hole would be cold expanded. For cracks that are too large for this remedy, the USAF could utilize a repair such as composite patching. This approach appears to be viable for aircraft with less than 45,000 commercial usage flight hours. It also may be viable for aircraft in the 45,000 to 55,000 flight hour range. A second alternative would be to replace the wing panels and stringers in the area of concern. This may be the only alternative for aircraft with more than 55,000 commercial usage flight hours.

KC-135 REVIEW

The KC-135 has already enjoyed a long life in the USAF inventory (12) although the program has endured many challenges. As indicated above, Boeing developed the KC-135 aircraft from the same prototype as the 707. The USAF acquired the KC-135 tanker aircraft, to replace the KC-97 to fulfill the need to refuel the B-52 fleet. The USAF ordered limited production of the KC-135 in 1954. The first flight occurred 31 August 1956. Production continued until 1965 when Boeing had manufactured 820 airplanes. A total of 37 different designations of the "-135" airplanes have existed in the inventory. After considering the strength to mass ratios for various aluminum alloys, the USAF chose to use 7178-T6 aluminum for the lower wing skin. This material and 7075-T6 found applications in other areas of the aircraft also. These were materials of choice at this time for many airplanes. Engineers did not recognize then they were prone to corrosion and stress corrosion cracking. In addition, the USAF used the existing technology corrosion protection for the KC-135. This system was incapable of protecting the materials selected for design.

The KC-135 did not have a design service life originally specified. Considering a fatigue test conducted in 1962, the USAF believed by performing certain modifications a service life of 13,000 hours was viable. Contradictory to the 1962 fatigue test results, the KC-135 aircraft experienced service problems early in its life. The 7178-T6 lower wing skin design had stresses approximately fifty percent higher than the 707 aircraft that had the higher toughness 2024-T3 alloy for the lower wing skins. Consequently, these airplanes experienced numerous cases of unstable cracking in the lower wing skins. In all, there have been approximately thirty unstable cracking cases in the interval 1,800 to 17,000 flight hours. The longest of these cracks was approximately 1.1 meters. The extensive cracking of the wing indicated the need for a much earlier replacement of the wing lower surface than originally planned. This alerted the USAF to the possibility WFD could degrade the fail-safety of the wing structure. The inspections of the other major components (that is, the fuselage and empennage), although much less extensive, found only a small amount of localized cracking.

The teardown inspections of six wings removed for a wing skin replacement permitted an examination of the structure for WFD. The examination found positive evidence of WFD. Consequently, in 1977 the USAF made a decision to replace the 7178-T6 lower wing skins up to Wing Station 733 production joint. They made the replacement between 8,000 and 9,000 flight hours with the same material used in the 707, that is, 2024-T3.

On 26 September 1977 the USAF initiated a durability and damage tolerance assessment on the KC-135 aircraft. Since the wing skin replacement removed much of the concern about the wing, this assessment focused primarily on the fuselage and empennage structure. This assessment, patterned after previous assessments, identified first all areas of the structure that would need inspections. Next, it identified the external and internal loading at these points, and then the degradation of residual strength of the structure from cracking at these locations. The USAF incorporated the results in a Force Structural Maintenance Plan (FSMP) that defined the inspection and modification program to the year 2040. As is typical, the analysis did not consider degradation due to corrosion except that Boeing determined the crack growth rate experimentally in a high humidity environment to represent the expected operational usage.

The concern about corrosion led to a teardown inspection for corrosion. Oklahoma Air Logistics Center accomplished this on an aircraft retired to Davis Monthan Air Force Base in 1991. This aircraft, delivered to the USAF in 1962, had spent twenty-nine years at Mildenhall Air Base in the UK. Therefore, the aircraft saw a severe corrosion environment during its life. The inspection interrogated the structure for cracking as well as corrosion. The USAF found little cracking since the aircraft had only 16,521 flight hours and 2,942 flights. They cleaned the parts and etched them approximately thirty micrometers to enhance corrosion and crack detection. The USAF, for this study, classified corrosion as light if it was less than 25 micrometers, moderate if it was between 25 and 250 micrometers, and severe if it was greater than 250 micrometers. For the fuselage there was extensive light corrosion in the skin and doubler faying surfaces. They found limited moderate and severe corrosion below the cargo door, lower bilge, and at the spotwelds. None of the fuselage corrosion was severe enough to affect flight safety. For the wing there was extensive moderate and severe pitting at the steel fasteners in the upper surface. Most of these had not progressed to exfoliation and none was severe enough to affect safety of flight. There were several areas of severe corrosion at the upper wing skin and spar interface. The horizontal tail center section suffered from severe exfoliation on the lower spar caps. They also found stress corrosion cracking in the horizontal tail. This inspection is significant in that it provided considerable insight on the extent of corrosion. It also served as an excellent representative aircraft for identifying areas of hidden corrosion that the USAF did not address in previous depot maintenance activities. In addition, it assessed the ability of available nondestructive inspection (NDI) procedures to locate corrosion. However, the NDI community has made significant progress in detecting corrosion since the completion of this work.

After the wing skin replacement, the KC-135 structurally more closely resembled the 707. There are, however, some important differences in the fuselage structural details and in the materials used in some areas. For example, the KC-135 unlike the 707 does not have tear straps in the fuselage and shear ties from the fuselage frames to the skin. The USAF carries fuel in the lower section of the KC-135 fuselage and consequently they do not pressurize it. A few of the joints in this area have spot welds in addition to rivets.

Stress corrosion cracking has been and continues to be a problem with parts made from some of the older 7000 series aluminum alloys. This generally occurs in the short transverse grain direction and as a result the cracks are parallel to the primary flight stresses. Thus, they seldom cause safety problems before detection and subsequent repair or part replacement. However, this type of cracking may become a safety problem for cracks that remain undetected. The problem is not unique to the KC-135 in that it has been occurring in essentially all of the older military and commercial fleets. Part replacement with a part made from a new stress corrosion resistant alloy or heat treatment is usually better than repair.

Although corrosion has caused significant maintenance on all of the older military and commercial aircraft fleets, it seldom has been directly the cause of major accidents. In recent teardown inspections performed on C-141, 707, and 747 aircraft there is little, if any, evidence wherein exfoliation or general pitting corrosion accelerated fatigue initiation or growth of cracks from fastener holes.

As indicated above, the Joint Stars program has utilized used 707 aircraft. These airplanes are being modified at the Northrop Grumman facility in Lake

Charles, Louisiana. Some of them are at or above the design service life of the 707. Although carefully selected from the available aircraft, the contractor found significant corrosion in all their components. In some of them, fuselage panels needed replacement to restore the fuselage to its design strength. The Boeing examination of these fuselage panels found no cracks although they typically found more damage by corrosion than the KC-135 selected for the teardown inspection. The excellent fatigue performance of these airplanes is the result of relatively low stresses in the fuselage skins. These low stresses resulted from the ultimate design condition of three times the nominal design pressure. This is a significant finding for the KC-135 since it infers the onset of WFD may never occur in the fuselage. The teardown inspection and subsequent analyses of the 707 described above serve to eliminate any concern about cracking in the wing lower skins and stringers. Therefore, the USAF would not expect fail-safety degradation of this structure during its planned time in service.

Even after two full-scale durability tests, a damage tolerance assessment, and a teardown inspection to search for corrosion there has been no definitive economical life established for the aircraft. In 1995 the program manager was considering conducting many structural tasks to respond to the Air Mobility Command's (AMC) desire to know the life remaining with the effects of corrosion included. The USAF included plans for these structural tasks in the Aircraft Sustainment Master Plan (ASMP) developed by the C/KC-135 Aging Aircraft Integrated Product Team, known as Coral Reach. This is a single document that identifies an integrated set of aircraft sustainment requirements. The objectives of the ASMP are to enhance flight safety, reduce the cost per flying hour, and improve aircraft availability. The plan includes proposed actions for the aircraft subsystems as well as the airframe. At the request of the program manager a panel consisting of USAF, FAA, and NASA engineers met at Tinker Air Force Base. Their purpose was to advise the program manager on efforts required to address the need of the AMC.

The KC-135 Aircraft Sustainment Master Plan (ASMP) dated June 1995 addresses the three essential metrics that measure the health of an aging aircraft such as the KC-135. These metrics are flight safety, aircraft availability, and cost per flying hour. The plan does this through the establishment of programs that consider the four characteristics identified above that would classify an aircraft as aging. As would be expected, corrosion is the sharpest focus for the ASMP.

The metrics identified above when sufficiently reliable, are the most useful for judging the life remaining in an aging aircraft. When one of the three metrics moves in an adverse direction such that alternatives appear to be attractive, then one could then project the useful life is at an end. Alternatives could be replacement or major modification. One of the problems with predicting these metrics in the future, however, is that funding shortfalls or changes in commitment to maintenance could affect the validity of the projections and cause earlier retirement. For this reason alone, the prediction of the life of an aircraft such as the KC-135 is very difficult to determine. Another problem is the reliability of the data base for computing costs. Data base reliability has been a problem of long duration in the USAF. It is very difficult to determine maintenance costs individually associated with repair, inspections, or corrosion prevention. Fortunately, this latter problem may be correctable by the data base system that is being developed by Boeing for Oklahoma Air Logistics Center. This system, called Stratotanker Condition Analysis and Logistics Evaluation (SCALE), provides detailed corrosion information by tail number. Decisions on future reduction of the force could help the System Program Director make decisions on basing and maintenance funding. This would ensure the airplanes are meeting their life objectives as economically as possible.

The SCALE program will be particularly useful in tracking the maintenance actions derived from stress corrosion cracking. This failure mode has occurred frequently on the KC-135 and normally requires part replacement or structural modification. It is usually better to replace the part with a material that is more resistant to stress corrosion cracking if the part is available. SCALE should be able to better track where these failures have occurred and be the basis for funding for long lead time procurement that should reduce downtime while saving money. An upgrade of the SCALE program will permit the tracking of inspections of critical areas. Identification of critical fastener holes and locations will allow the maintenance activity to provide a history of findings by location. This addition would help in the determination whether the damage tolerance critical areas of the structure are being influenced by corrosion.

The 707 teardown inspection effort as discussed above indicates the onset of WFD for regions of this aircraft that are similar to the KC-135 is somewhere beyond the year 2040. At the current utilization rate of the KC-135, for areas that are structurally similar to the 707, the onset of WFD should not occur in less than sixty years. Consequently, the USAF should direct any effort at the onset of WFD at those details that are different from the 707.

The USAF may gain useful generic information from the testing of pre-corroded material to measure the growth rates of cracks at fastener holes of the sizes that could indicate the onset of WFD. However, based on the results from the 707 teardown inspections, this does not appear to be a near term problem for the KC-135. In view of the crack growth tests already conducted it is difficult to visualize how the corroded surface could be any more critical than the crack front itself. Notwithstanding this point, there does appear to be a slight influence of exfoliation on the growth of cracks. Magnification of this effect may take place to some extent in the short crack regime.

There has been a concern as to whether or not fatigue crack growth rates might be higher in corroded material than in uncorroded material. If these rates were higher, then the crack growth calculations may be unconservative. To obtain an answer to this concern Boeing has been obtaining crack growth rate data on precorroded and uncorroded laboratory specimens. Results to date indicate the rates for the precorroded material are within the scatter band of the uncorroded material, but are on the upper side of the band. One may speculate this is merely the result of the material thinning rather than differences in basic crack growth properties. The USAF needs to determine whether or not this is the case.

The most effective way to account for the effects of corrosion is to make sure it never becomes severe enough to affect fatigue life. Achievement of this is primarily through proper protective treatment and adequate periodic inspections. The core sample technique, for example, to determine the extent of corrosion in fuselage skins appears to be very effective. The USAF has evaluated the Mizz-22/KKB Dual-Frequency eddy current, MAUS III eddy current, and D-sight NDI systems under C/KC-135 depot overhaul conditions. The engineers at Tinker Air Force Base are currently using the MAUS III system on selected parts to help confirm hidden corrosion in lap splices. However, they assert the USAF needs several breakthroughs in the technology of effectively detecting and accurately quantifying hidden corrosion before they can reliably detect it. When they develop this capability they will be able to rapidly and economically scan all parts of all aircraft as they enter the depot overhaul process. Only then, they believe, will all hidden corrosion be detected and repaired before becoming a safety of flight concern.

Corrosion around the fasteners in the wing is a very difficult problem with no solution identified. The USAF plans to initiate a program to further examine this problem. This problem exists in all fastener locations, but the steel fasteners used at the splicing stringers are particularly prone to corrosion problems. Replacement of these fasteners with titanium is being considered, but there is a question on the benefits from this change. This is a problem that, without a solution, could adversely affect the cost per flying hour metric. There is a need for continued emphasis on prevention methods that use sealants and newer coating systems that might effectively adhere to both the skin and the fastener. Development of low surface tension wicking compounds that contain inhibitors could alleviate this problem. However, some experts believe the alleviation may not last the desired life of the KC-135. Currently, the most effective inhibitors are not environmentally friendly and are not being considered for use. The key to this problem is either keeping moisture out of the area through a barrier seal on the wing skin or reducing the galvanic potential between the fastener and the wing skin.

The USAF does not accurately know the consequences of very long term exposure of the materials used in the manufacture of the KC-135 to corrosive environments. The data base does not include this type of information. The Oklahoma Air Logistics Center has initiated programs that are a good start to obtain environmental exposure data. This type of information should provide data on factors contributing to degradation and should serve to support studies on protection and basing scenarios to extend aircraft life.

There is no argument among the experts that environmental factors play a significant role in corrosion (13). The USAF needs to look aggressively at practical alternatives that would protect aircraft. Areas of be investigated include basing sequences between Program Depot Maintenance visits, washing frequencies and compounds, and protection through the use of environmentally conditioned shelters.

The panel concluded the maintenance program for the KC-135 is excellent. The maintenance team appears to be fully aware of this aircraft's importance to the operational readiness of the USAF. In areas where corrosion damage has forced replacement of parts, this is being accomplished with materials that are considerably more corrosion resistant than those originally used. In some cases, there are redesigns to eliminate some corrosion prone features such as spot welding. In addition, maintenance uses the current technology corrosion protection procedures for part replacement. There is also a commitment to the use of corrosion prevention compounds where possible.

There is no evidence of any near term significant change in the three metrics: flight safety, availability, or cost per flight hour. However, there is a concern about the continued structural integrity of the 7178-T6 aluminum outer wing, outboard of the Wing Station 733 production joint. This area needs additional investigation. Additionally, the panel concluded the maintenance program needs to maintain its aggressiveness in controlling corrosion thus preventing it from becoming a safety issue. The engineers at Tinker Air Force Base, however, do not believe their program is completely controlling damage from corrosion. Their maintenance personnel are reacting to corrosion they find. They do not look everywhere and do not disassemble to uncover hidden corrosion. They spray the inside of the fuselage, the leading and trailing edges of the wing, and wheel wells with a corrosion prevention compound (CPC). This is not totally effective since the USAF seals the joints on the inside to maintain pressurization. Therefore, the CPC does not get into the fuselage lap splices or in spot welded doubler areas. Also, as a consequence of the aggressive wash schedules in highly corrosive environments, the

USAF removes some of the CPCs. The panel expressed concern about the potential for a wing skin replacement because of corrosion damage. A solution to this corrosion problem could eliminate a significant economical impact. From the available data, it does not appear there will be an onset of WFD in the fuselage, empennage, or the 2024-T3 lower wing skins until after the year 2040. The USAF derived this data mainly from the work performed for the Joint Stars aircraft.

The USAF has operated this aircraft well beyond the normal retirement calendar time. However, the panel concluded the risk is low that the KC-135 because of structural considerations will reach the point of retirement before the year 2040. The year 2040 is consistent with the selection the USAF made when Boeing drafted the FSMP after the completion of the damage tolerance assessment in 1980. The panel supposed for this conclusion the aircraft will continue to receive the same good maintenance practices it currently enjoys. Failure to do this could terminate the life of the aircraft significantly short of the desired goals. Further, the panel based their conclusion on the successful completion of the revised Coral Reach programs. While the panel believed that there will not be any change in the materials with age, excluding the effects of corrosion, the USAF plans to operate these airplanes well past existing data bases. Further, the panel believed there will not be a breakthrough in the science of corrosion prediction within the time that will be useful for this aircraft. The key to success with corrosion for this aircraft or any other is enhancement of the capability for detection and for prevention. The results of the ASMP combined with the efforts outlined above should provide improved visibility of the cost per flying hour and availability issues associated with the continued operation of the aircraft. Also there will be an improved data base for any future decision about its retirement.

C-141 REVIEW

Lockheed designed and manufactured the C-141 as a long range heavy logistics transport. They manufactured and delivered it to the USAF during the interval from January 1964 to February 1968. They analyzed and tested the aircraft to ensure it would achieve its original design service life of 30,000 flight hours. Later, when the USAF desired to stretch the fuselage to increase its volume, additional analyses showed the aircraft should reach an economical life of 45,000 flight hours of the then current usage.

The C-141 like the KC-135 has had many challenges to its structural integrity over the years. Many of these challenges are the result of the severe training and high speed low level operations that have accelerated damage to the lower surface of the wing. There have been cases of cracking in the fuselage, but in most cases the loss of fail-safety of the wing was the main reason for concern. There have been many cases in commercial and military operations where fail-safety has saved aircraft from catastrophic failure. This is also true for the C-141. There have been numerous cases of single member failures in the wing of this aircraft that would have been catastrophic without the fail-safe features built in this structure. Examples of problems derived from these operations include cracking in the inner to outer wing splice (Wing Station 405) (14) and the cracking of the fuel transfer holes (9). In both of these cases, the USAF made modifications to restore the wing's fail-safe capability. In many of the wings, the USAF made composite repairs to the metallic structure. This was a good example of technology transition. The Australians initially developed this technology and transferred it to the Wright Laboratory Materials Directorate. The Materials Directorate subsequently transferred it to the Warner Robins Air Logistics Center for incorporation on the C-141. More recently, the USAF became concerned about the potential for another crisis associated with cracking in the wing. This occurred in the spanwise splices of the lower inner wing surface. Figure (20) shows the location of these splices. Figure (21) shows the details of the spanwise splice joint. The USAF originally estimated spanwise splice cracking would significantly degrade fail-safety at an equivalent of 45,000 flight hours for the inner wing lower surface (15). They based this conclusion on teardown inspection data from the durability test article. Considering more recent teardown information from operational aircraft, Lockheed Martin revised this estimate downward to 37,000 flight hours This cracking, according to analyses performed by Lockheed Martin, had significantly degraded the fail-safety of the wing. The USAF therefore found themselves with many of these airplanes over 37,000 flight hours, but less than the planned retirement time of 45,000 flight hours. Figure (22) shows this population. They made the judgment, however, that it was not economically viable to restore the wing fail-safety through a modification program. The USAF considered several options for the restoration of fail-safety. The most promising of which was the replacement of the lower wing panels. Also, they considered the use of composite straps that would act as fail-safe straps (analogous to fuselage fail-safe straps). They adopted none of these alternatives. Consequently, the wing safety depended largely on the ability to inspect a large area of the wing for cracks. The total number of fasteners that require inspection exceeds 6,000. Also, because the crack would need to be approximately 25 millimeters outside the spanwise splice joint for it to be inspectable with high frequency eddy current, the inspection interval was very short - 120 days. Figure (23) shows the crack growth function for the spanwise splice fastener holes. Figure (21) shows the details of the spanwise splice illustrating the difficulty in inspecting this structure with resulting small intervals for recurring inspections. The concern over the viability of this approach motivated an ad hoc review. It was the purpose of the ad hoc review to determine if the risk associated with this procedure was acceptable. If they determined it to be unacceptable, then they were to seek alternatives to attain an acceptable level of risk.

At the initial meeting of the ad hoc team, they concluded the current program of inspecting with eddy current should be adequate to protect the safety of the wing. Figure (24) shows the calculated risk of failure from spanwise splice cracking. However, the team emphasized the success of the management philosophy using "slow crack growth" is totally dependent on the performance and accuracy of the NDI mandated by the C-141 program manager. The ad hoc team recommended the performance of the inspectors be examined by a team of experts. These experts would conduct an on-site audit of the inspectors and conduct proficiency tests. This would determine if inspections on the C-141 wing are being effectively performed as required by the technical orders.

The audit team evaluated both C-141 dedicated inspectors and other USAF inspectors during their assessment. There were many found to be able to adequately perform the inspection. They also found a significant number of them that were not proficient. However, the team found the C-141 inspectors were typically better than the others. The audit team came to two main conclusions resulting from their review of the adequacy of inspections performed on C-141 wings. First, it is was not realistic to expect that the sensitivity and reliability of inspections performed by the inspectors who demonstrated poor or questionable proficiency would be adequate. Second, the team could not predict the performance of inspectors who scored average or better in the proficiency tests. The C-141 inspectors inspect the wing manually with no data recorded. Therefore, they believed the non-proficient inspectors will not meet the C-141 inspection requirements but they have no way of determining how successful proficient inspectors would be.

These disturbing results precipitated two action plans. The USAF aimed the first plan at improving the ongoing inspection with high frequency eddy current. This plan included the continued testing of C-141 inspectors and the elimination of those that did not appear to be proficient. This activity was somewhat subjective since the USAF had not correlated the test demonstrated proficiency with the degradation of the detection probability for cracks. For inspections that are so dependent on human factors such as this, the USAF needs to establish quantitatively the influence of the inspector on the reliability of the inspection.

Second, they planned to replace existing labor intensive eddy current with an automated NDI system that uses ultrasonic technology. Ultrasonic inspection would be a significant advantage for this structure since it can inspect both layers of the spanwise splice joint for cracks as small as two to three millimeters. This is possible since the joint has a sealant that will act a coupling and transmit the ultrasonic energy. The technology used in this inspection is not new. The mechanization of the technology will relieve the uncertainty associated with the human factor aspect of the inspection. Another advantage of this approach is that the USAF would need to perform this inspection only at depot visits rather than the current 120 day interval.

CONCLUSIONS

The programs discussed above indicate the aging problem has unique features on any given aircraft. However, the reasons for aging all appear to fall in the four categories: operations beyond the design service life, corrosion, onset of WFD, or repairs. As seen from the examples discussed above, the USAF found there may be vast differences in the technology needs from one aircraft to another. In most cases the technology is available to solve the problem. The FAA Technical Center and the NASA Langley Research Center have contributed significantly to many areas through their aggressive aging aircraft research and development programs. Some of the technologies studied under these programs are: finite element modeling, residual strength determination of highly loaded structures, stress intensity solutions, crack growth codes, growth of short cracks, repairs for metallic structures with metal or composite patches, and NDI. This work combined with the capability developed by the USAF primarily during the damage tolerance assessments described above provides a basis for aging aircraft evaluations. For example, the USAF accomplished the risk assessment for the Joint Stars aircraft with existing technology. It may have been more economical to have had a procedure for determining the crack distribution in the 707 wing structure without the expense of a teardown inspection. However, the development of technology for an alternative approach may also be expensive. Also, the alternative may never be as reliable as a teardown of the aircraft that has the loading and the environmental effects already integrated. The USAF must exercise considerable care in formulating a plan for further research and development for the aging aircraft problem. They must carefully consider the benefits from such efforts. The researcher must consider whether the investment will be useful and if it is competitive with alternatives. The investment must also include, where appropriate, the cost of transitioning the technology to the depot where the USAF would use it as a maintenance tool. It is important that the researcher and the operator communicate so that the operator has a clear understanding of the expectations from research and development efforts. The researcher must be capable of defining the cost and schedule for accomplishing the needs of the operator so that they can reach appropriate compromises. Otherwise, there will be considerable funds expended and disappointments associated with failure to achieve promised results. There are some generic technologies that span many of the aging aircraft problems. These include studies WFD, corrosion, repairs, and NDI. In addition, the USAF should place major emphasis on developing data bases from which they can determine the extent of the aging aircraft problem on a specific weapon system. This will permit the quantification of the metrics of cost per flying hour and availability that are the best indicators for the structural life of an aircraft.

ACKNOWLEDGMENTS

The author gratefully acknowledges the technical support from Messrs. Kyle Dixon, Anthony Hug, and Joseph Luzar from the Boeing Defense and Space Group in Wichita, Kansas on the 707 stress analysis and teardown inspection. Also, the author is grateful for their advice and council of the members of the review teams.

REFERENCES

(1) Lincoln, J.W. "Damage Tolerance - USAF Experience", Proceedings of the 13th Symposium of the International Committee on Aeronautical Fatigue, Pisa, Italy, 1985.

(2) Babish, C.A. "F-16 Structural Integrity Program", WL-TR-92-4045, Proceedings of the 1991 USAF Structural Integrity Program Conference, San Antonio, Texas, 1992.

(3) Lincoln, J.W. "Structural Technology Transition to New Aircraft", Proceeding of the 14th Symposium of the International Committee on Aeronautical Fatigue, Ottawa, Ontario, Canada, 1987.

(4) Richards, P.F. "Continuing Structural Airworthiness of Civil Transport Aircraft", The Aeronautical Journal of the Royal Aeronautical Society, November 1980.

(5) Craig, L.E. and Goranson, U.G. "Airworthiness Assessment of Boeing Jet Transport Structures", Proceedings of the 10th Symposium of the International Committee on Aeronautical Fatigue, Brussels, Belgium, 1979.

(6) Swift, T. "Damage Tolerance in Pressurized Fuselages", Proceedings of the 14th Symposium of the International Committee on Aeronautical Fatigue, 11th Plantema Memorial Lecture, Ottawa, Canada, 1987.

(7) Swift, T. "The Influence of Slow Growth and Net Section Yielding on the Residual Strength of Stiffened Structure",. Proceedings of the 13th Symposium of the International Committee on Aeronautical Fatigue, Pisa, Italy, 1985.

(8) Lincoln, J.W. "Risk Assessments-USAF Experience", Proceedings of the International Workshop on Structural Integrity of Aging Airplanes, Atlanta, GA, March 31 - 2 April 1992.

(9) Elliott, W.R., Lincoln, J.W., and Register, D.C. "Assessment and Terminating Action for Widespread Fatigue Damage in C-141 Wing Fuel Transfer Holes", Proceedings of the 18th Symposium of the International Committee on Aeronautical Fatigue, Melbourne, Australia, 1995.

(10) Lincoln, J.W. "Risk Assessment of an Aging Military Aircraft", Journal of Aircraft, Volume 22, Number 8, 1985.

(11) Department of the Air Force "Air Force Guide Specification, Aircraft Structures, General Specification for", AFGS-87221A, 1990.

(12) Lincoln, J.W. "Life Management Approach For USAF Aircraft", Proceedings of the 72nd AGARD Structures and Materials Panel Meeting, AGARD-CP-506, Bath, United Kingdom, 1991.

(13) Schütz, W. "Corrosion Fatigue - The Forgotten Factor in Assessing Durability", Proceedings of the 18th Symposium of the International Committee on Aeronautical Fatigue, 15th Plantema Memorial Lecture, Melbourne, Australia, 1995.

(14) Alford, R.E., Bell, R.P., Cochran, J.B. and Hammond, D.O. "C-141 WS 405 Risk Assessment", WL-TR-92-4045, Proceedings of the 1991 USAF Structural Integrity Program Conference, San Antonio, Texas, 1992.

(15) Lincoln, J.W. "Reassessment of the C-141 Structural Life", Proceeding of the 16th Symposium of the International Committee on Aeronautical Fatigue, Tokyo, Japan, 1991.

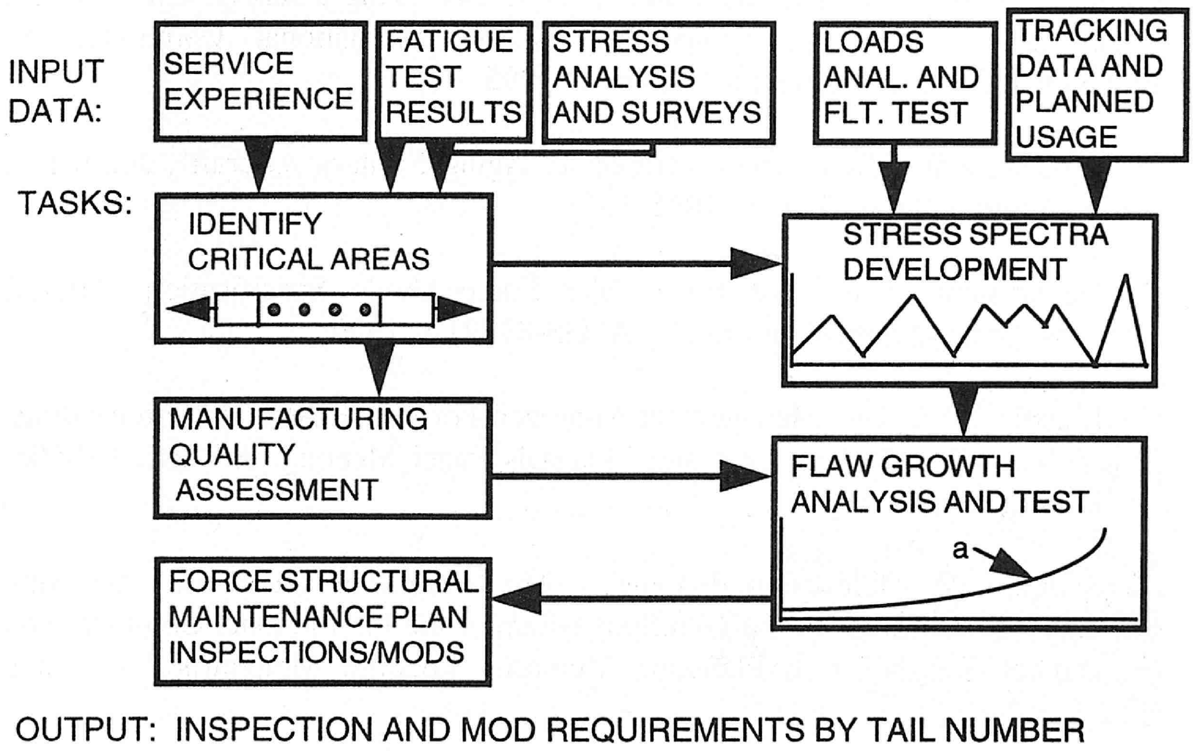

Figure 1 Damage tolerance approach

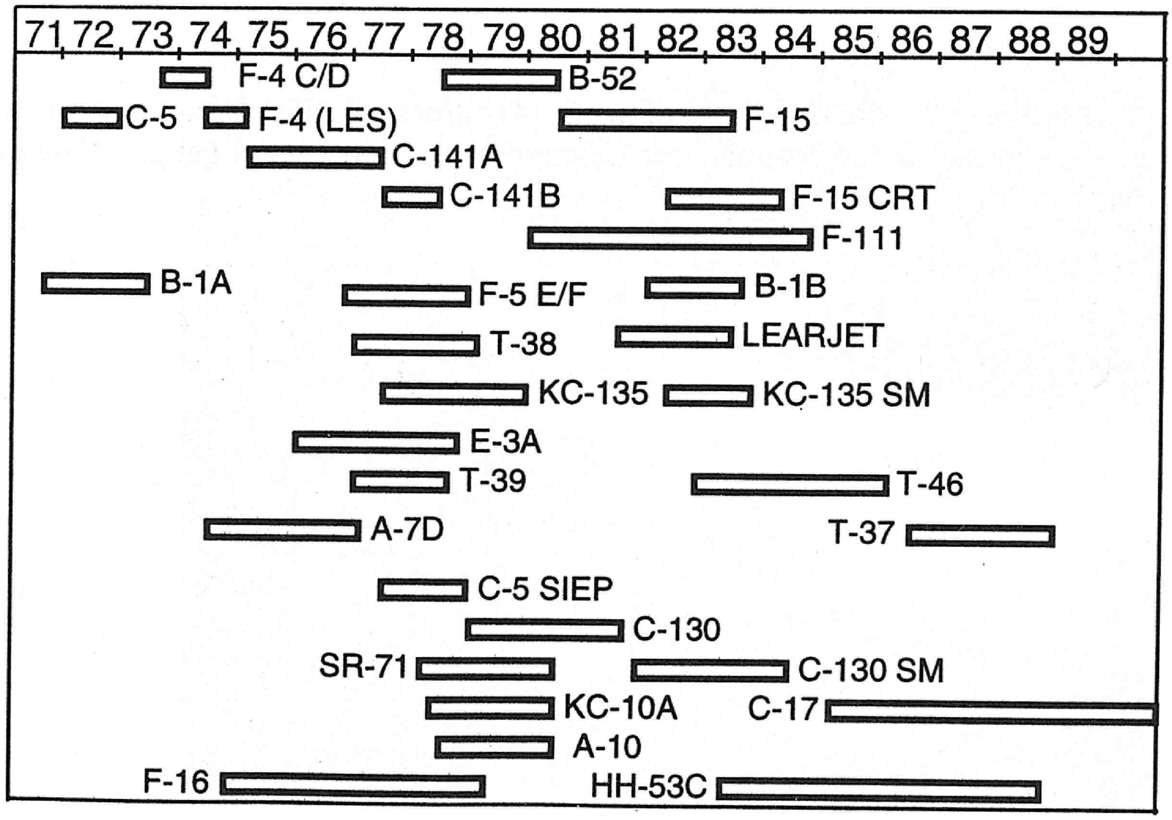

Figure 2 Damage tolerance experience - aircraft

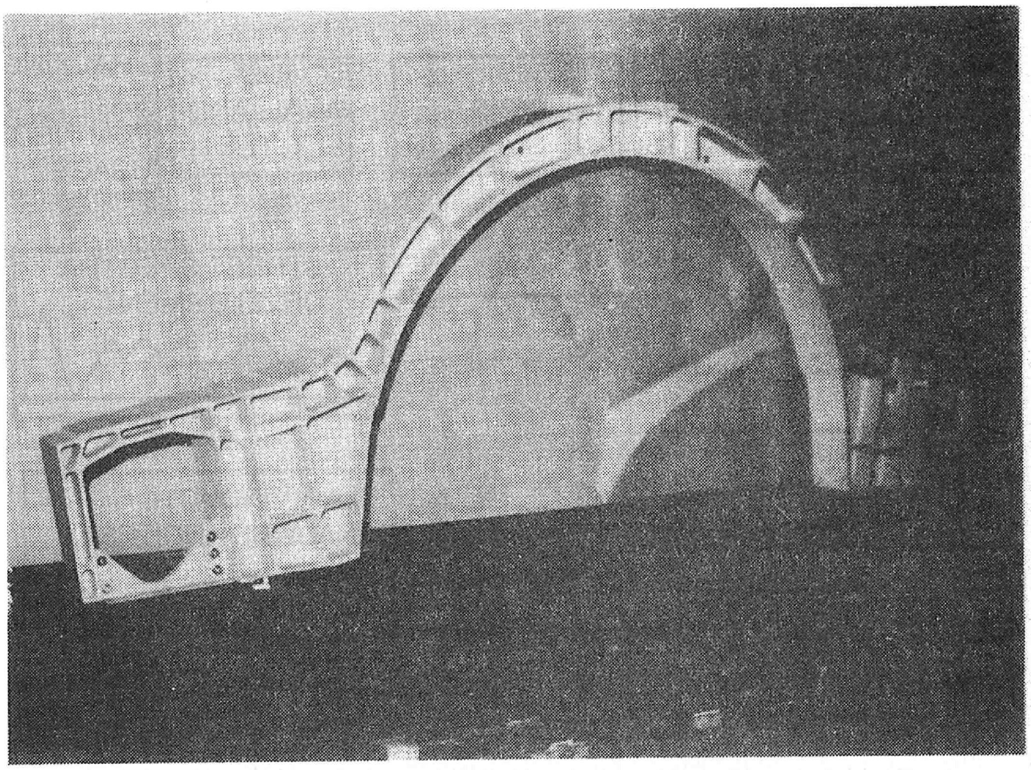

Figure 3 F16 Fuselage Station 479 bulkhead

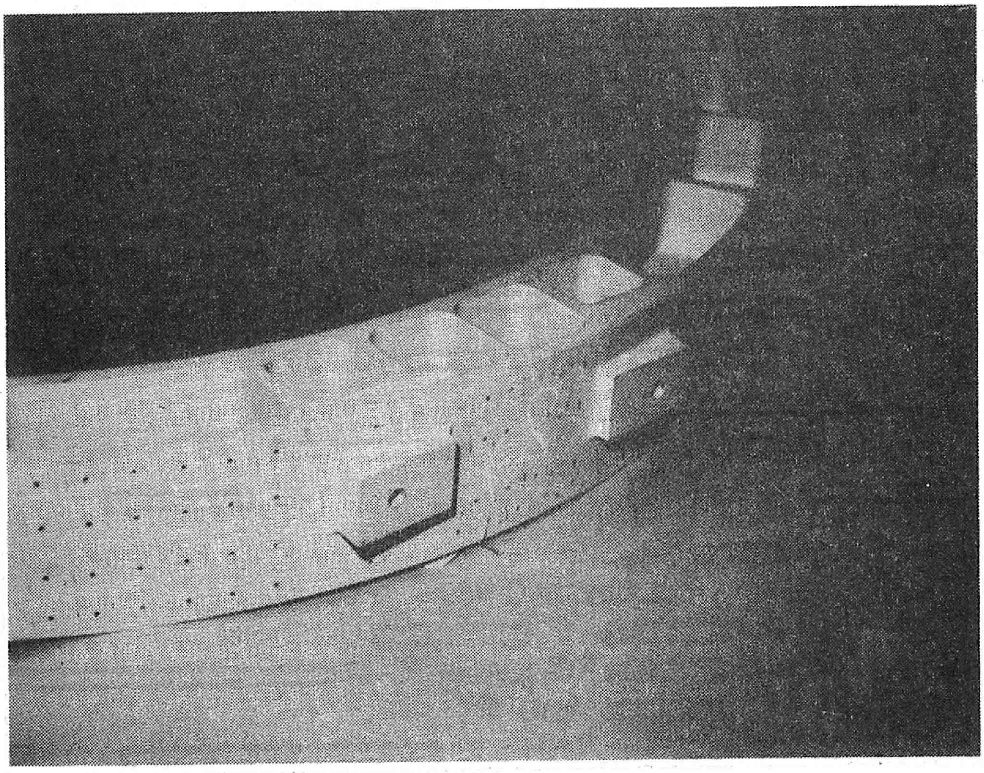

Figure 4 F-16 vertical tail attachment detail

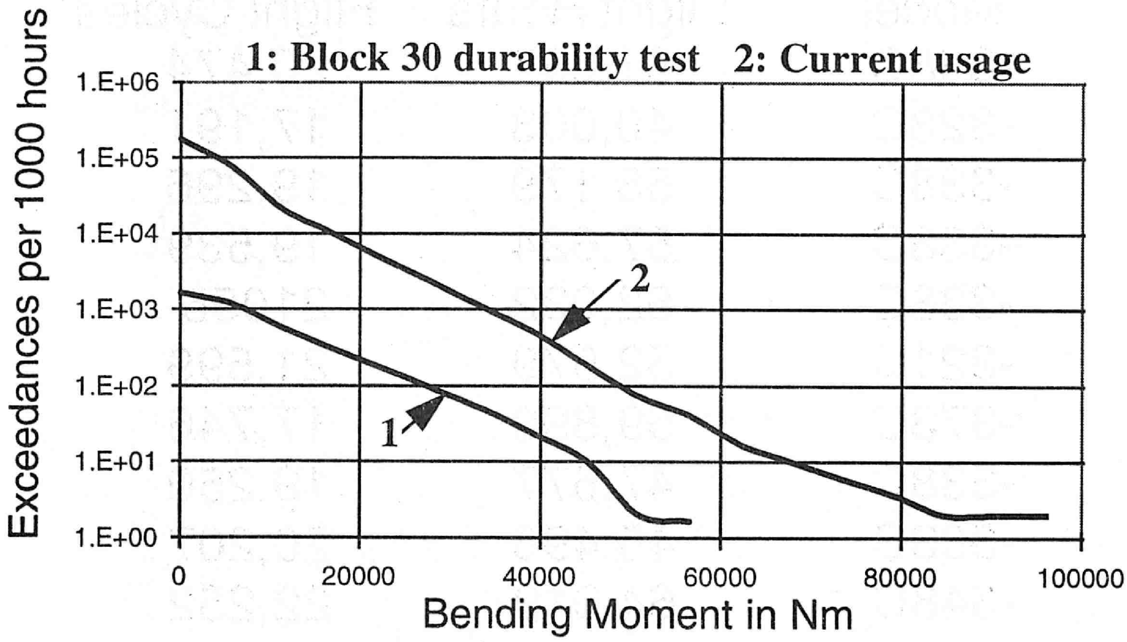

Figure 5 Vertical tail bending moment exceedance function

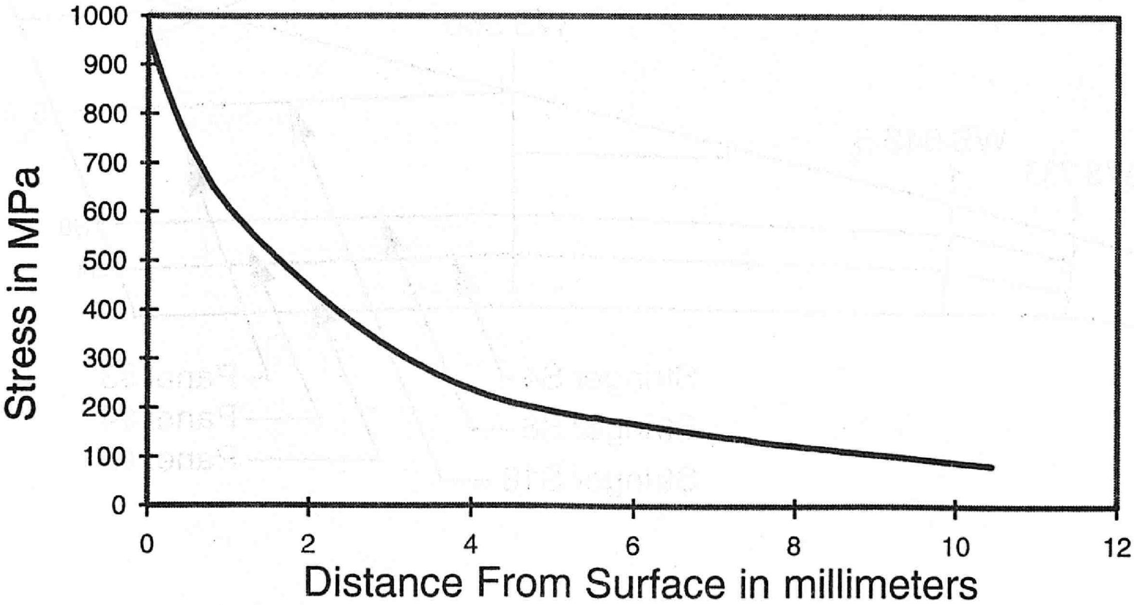

Figure 6 Elastic solution stress distribution

JOINT STARS FIRST TEN AIRCRAFT

Model	Flight Hours	Flight Cycles
-338C	51,647	20,474
-323C	40,008	17,191
-338C	55,179	19,296
-338C	57,624	19,539
-338C	62,032	21055
-321C	52,579	21,599
-373C	59,890	17,746
-338C	47,677	19,250
-303C	45,496	20,207
-348C	64,019	22,252

Figure 7 Prior commercial usage of Joint Stars aircraft

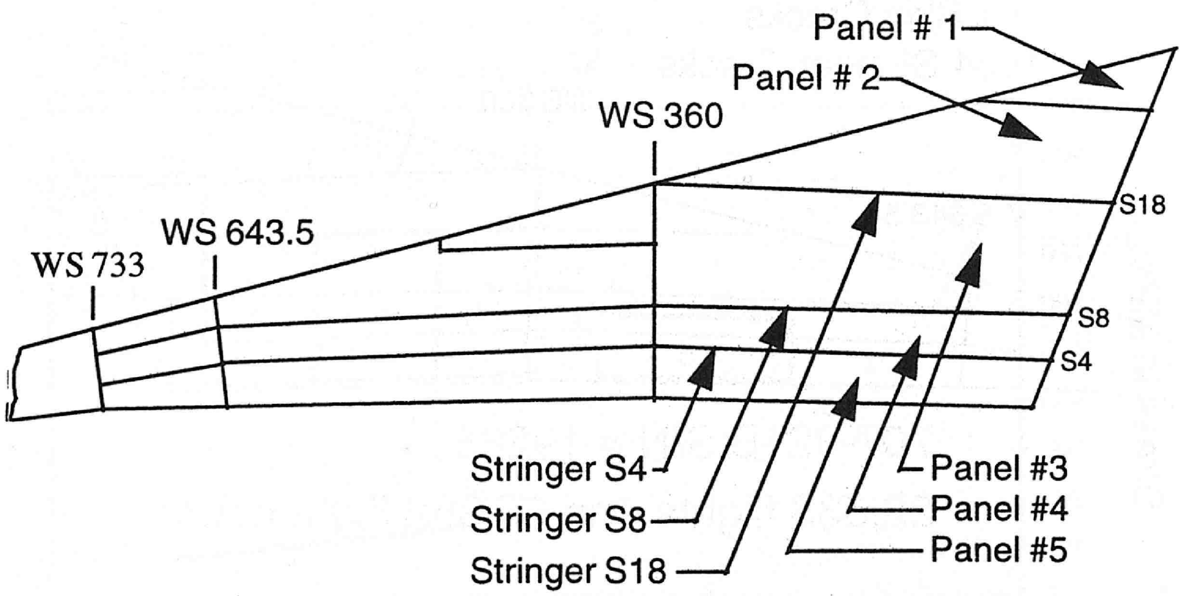

Figure 8 Plan view of the Boeing 707 wing

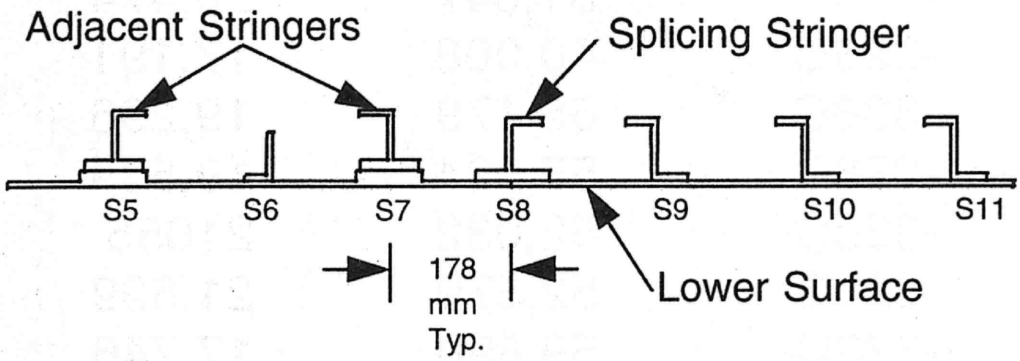

Figure 9 Boeing 707 wing cross section

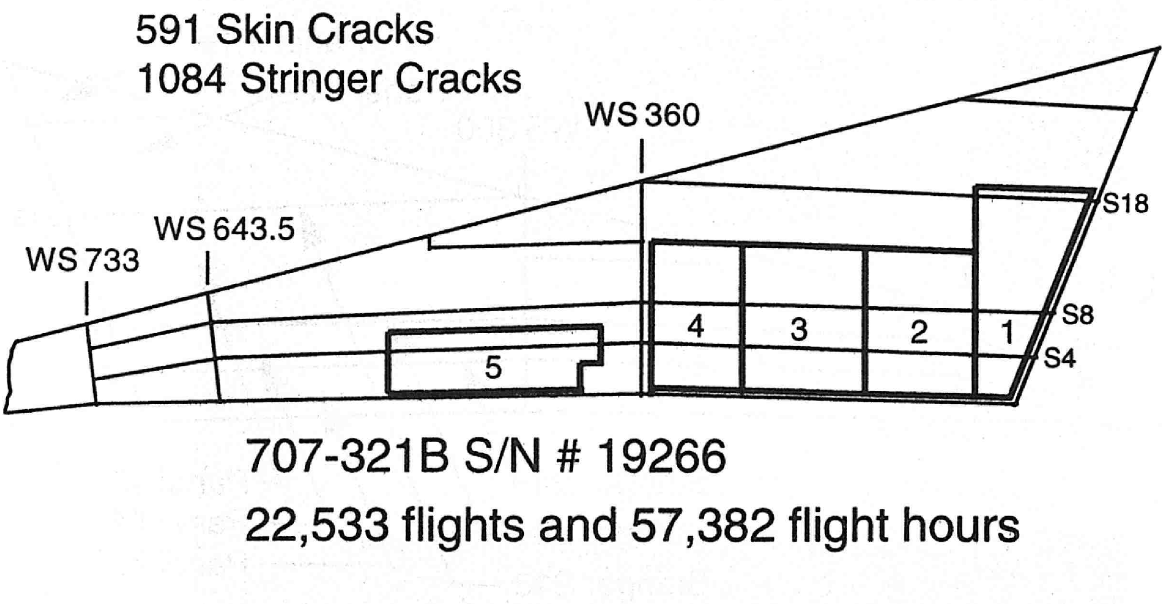

Figure 10 Sections of wing removed for teardown inspection

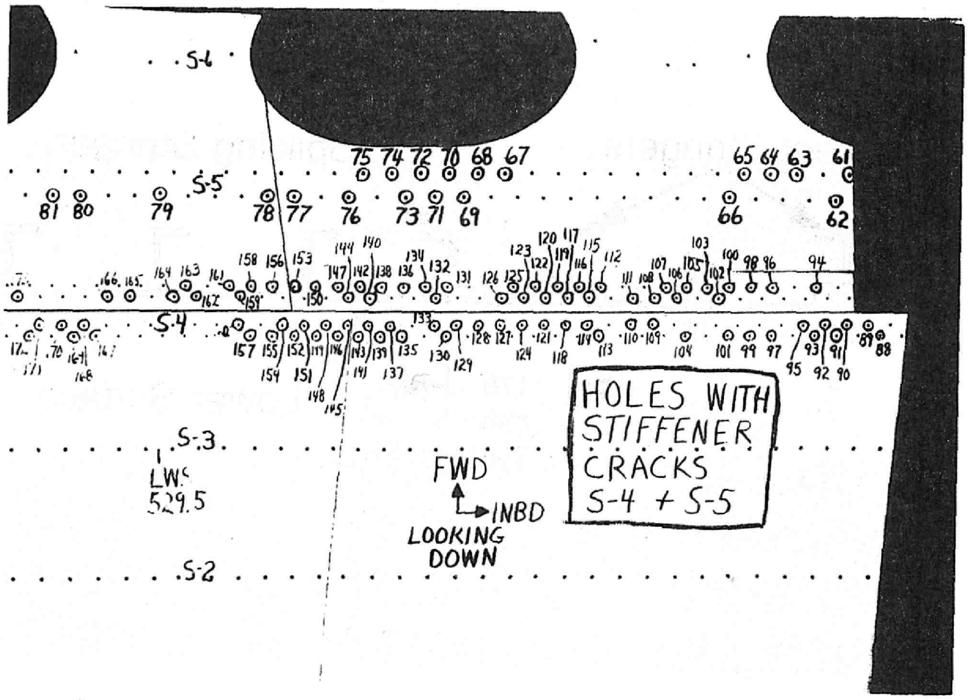

Figure 11 Stringer cracks found in teardown inspection

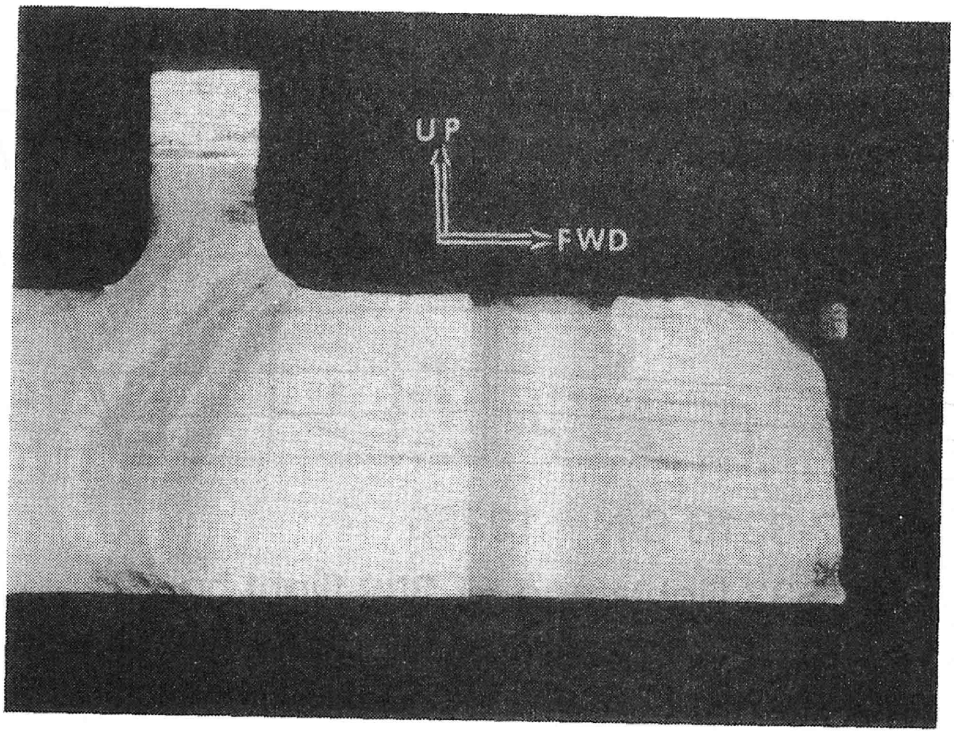

Figure 12 Stringer 7 crack found in teardown inspection

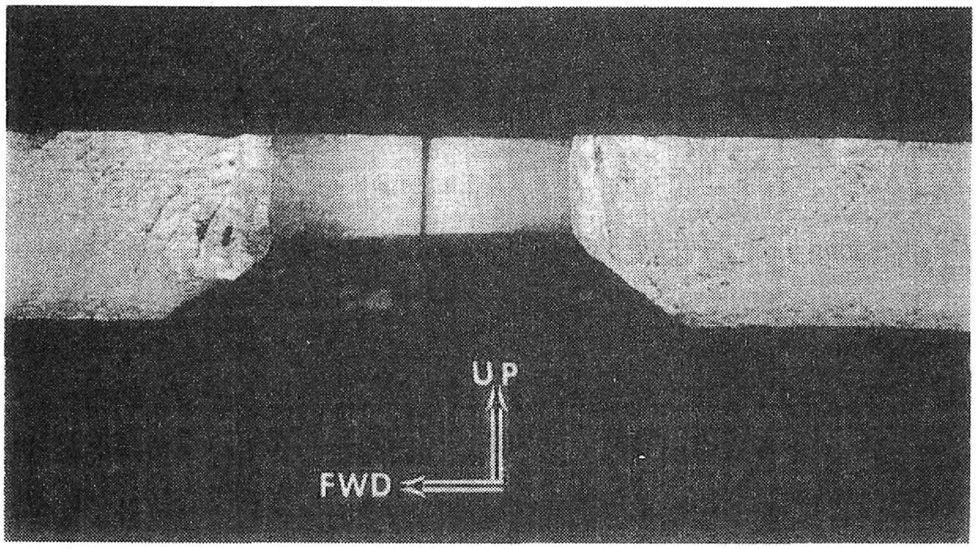

Figure 13 Skin crack found in teardown inspection

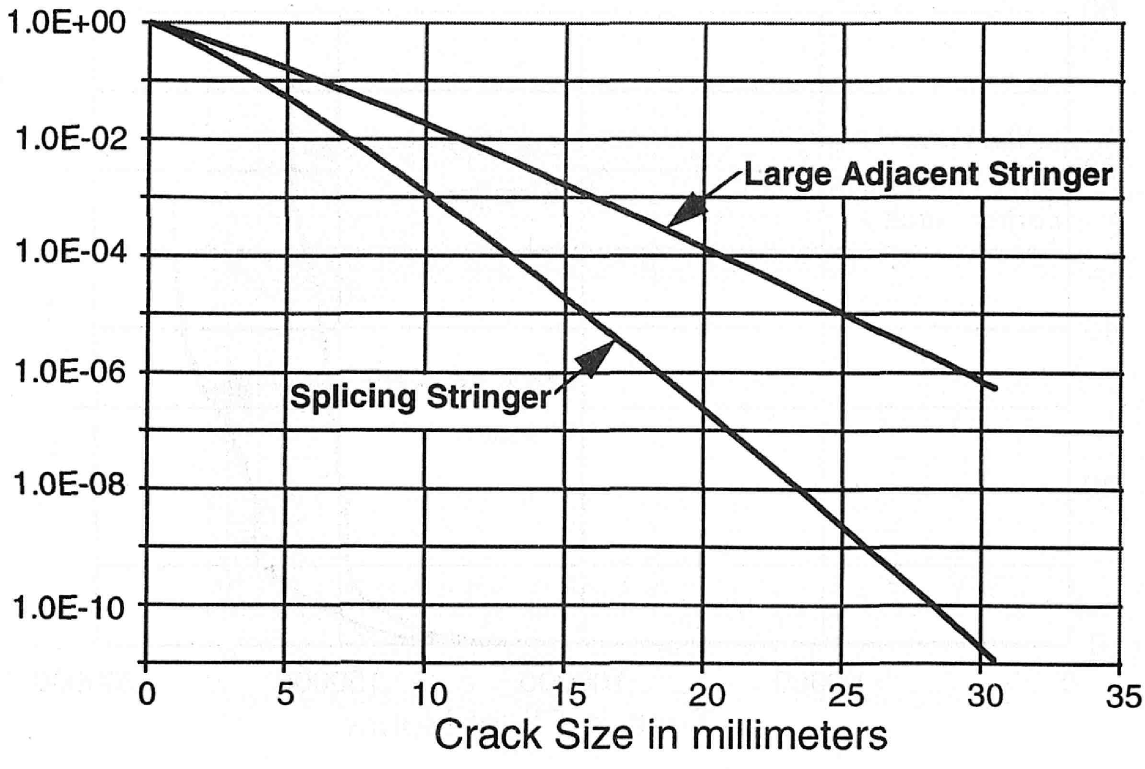

Figure 14 Crack distribution functions for the stringers

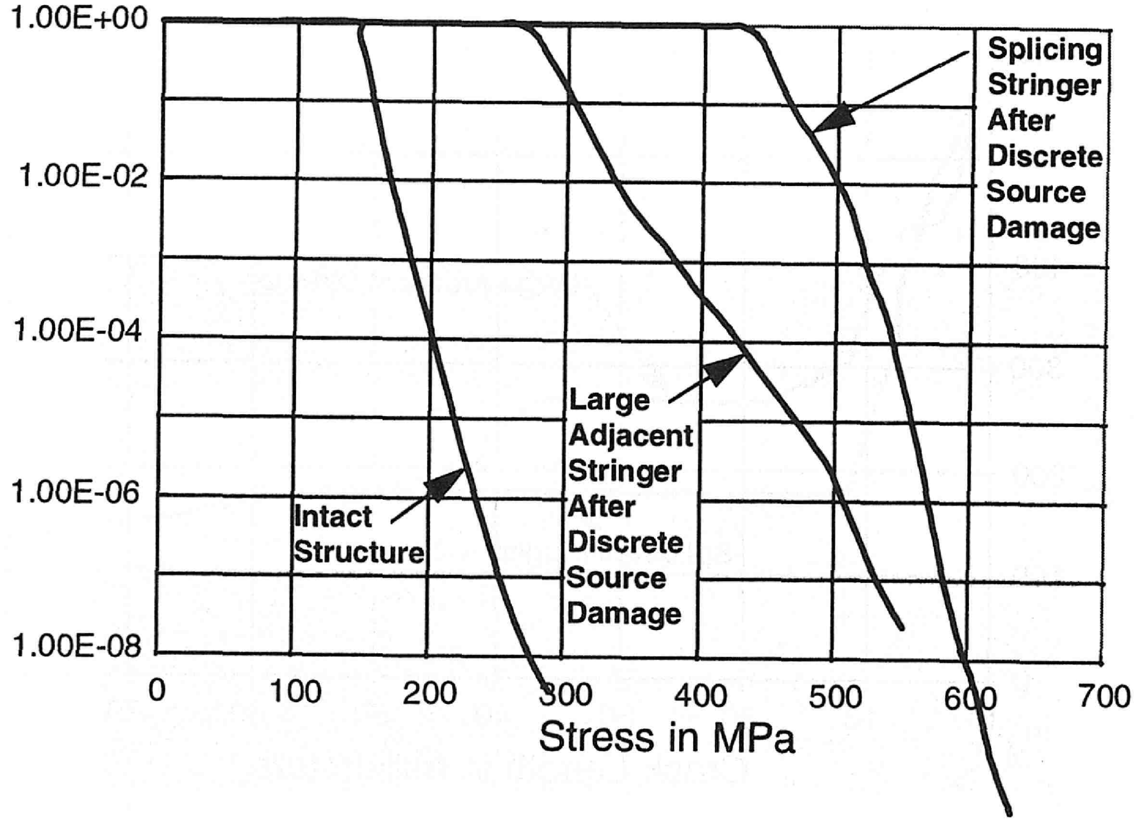

Figure 15 Stress distribution function for wing structure

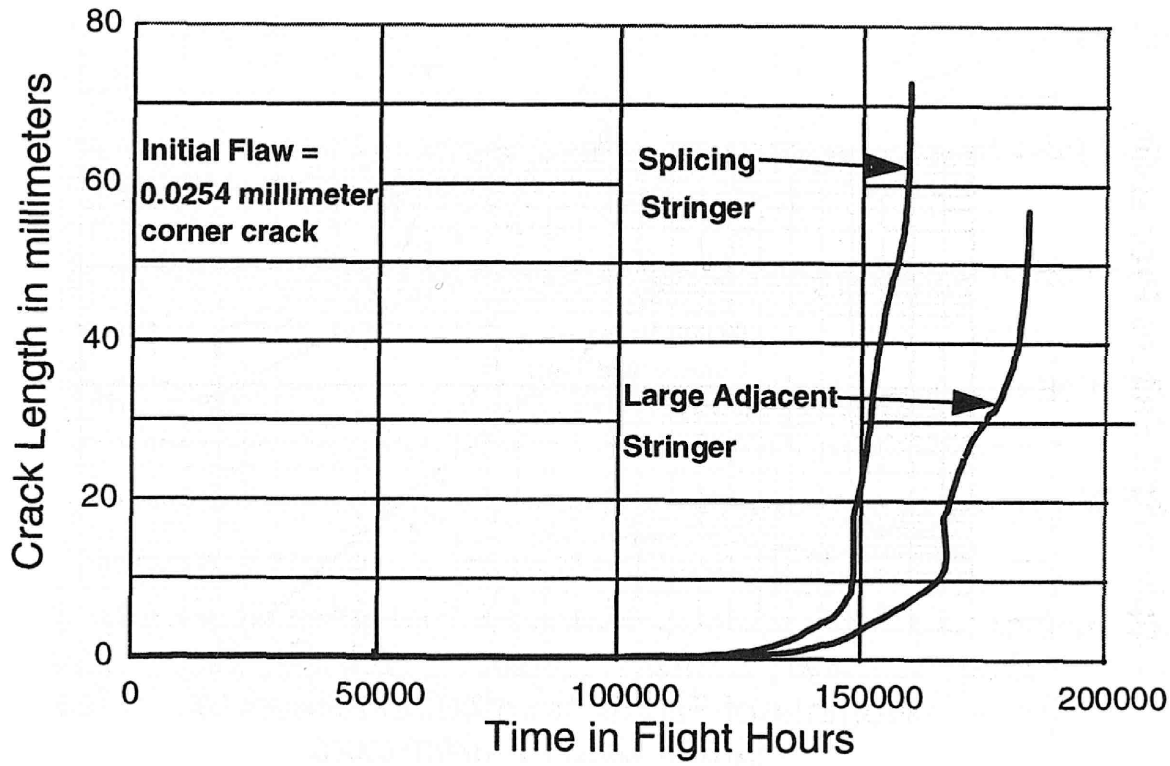

Figure 16 Crack growth functions for the stringers

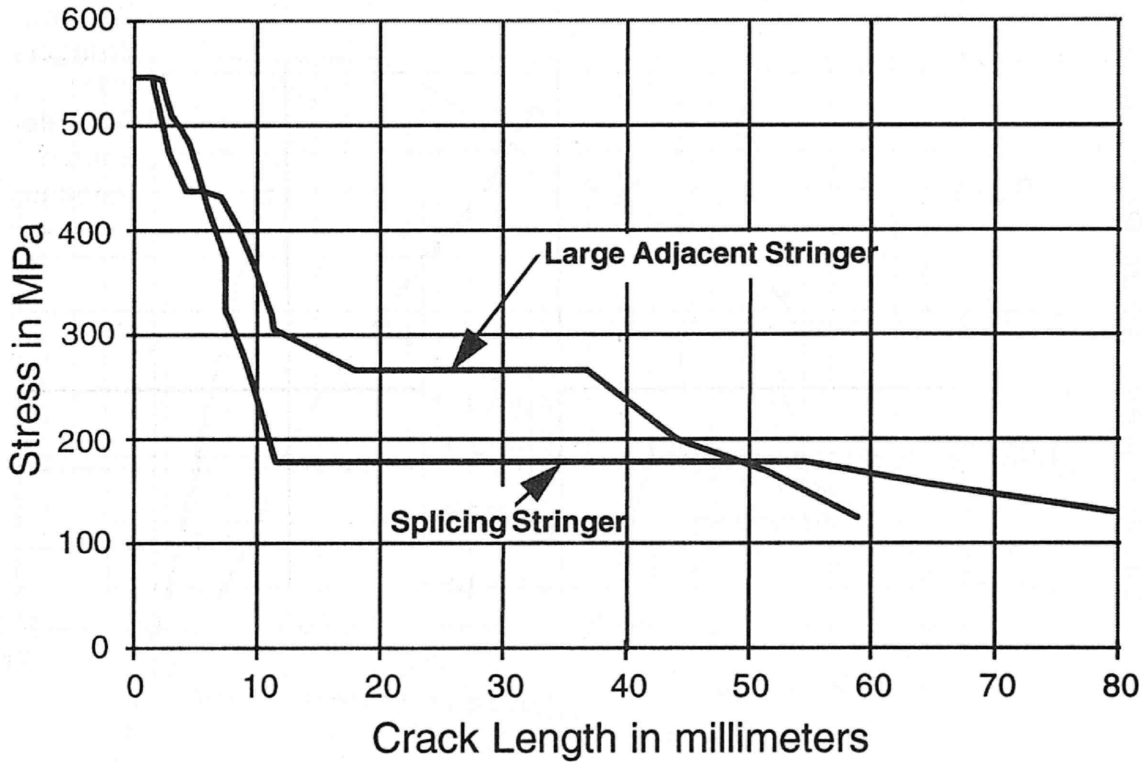

Figure 17 Critical stress functions for the stringers

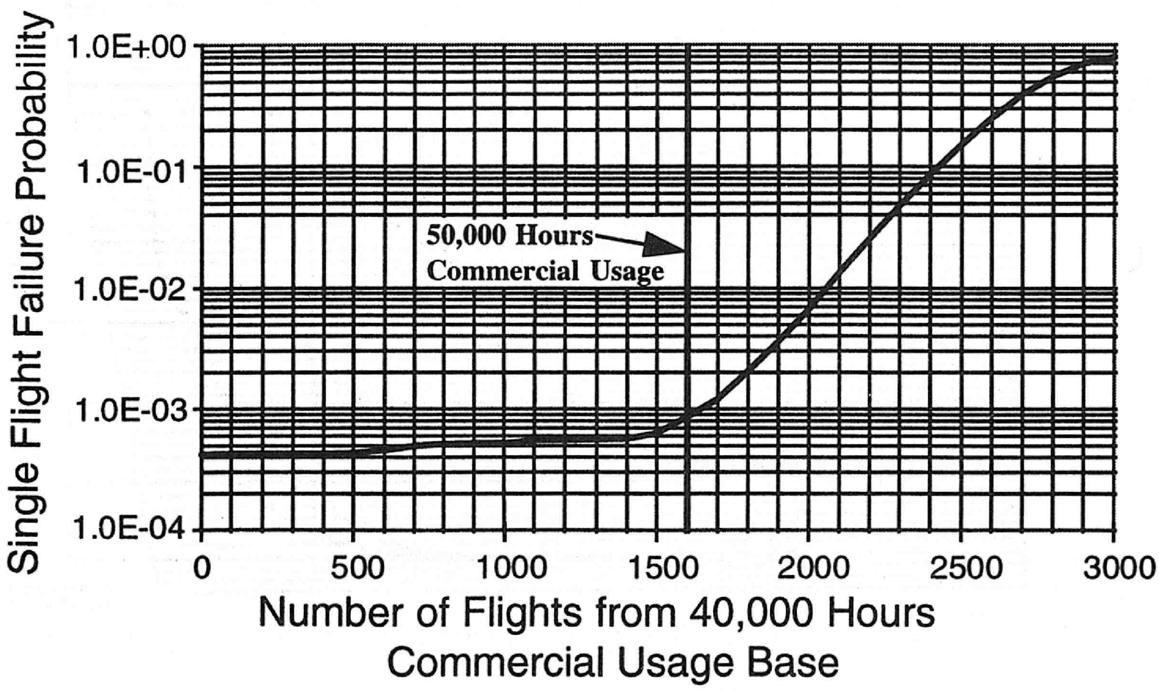

Figure 18 Risk assessment for discrete source damage

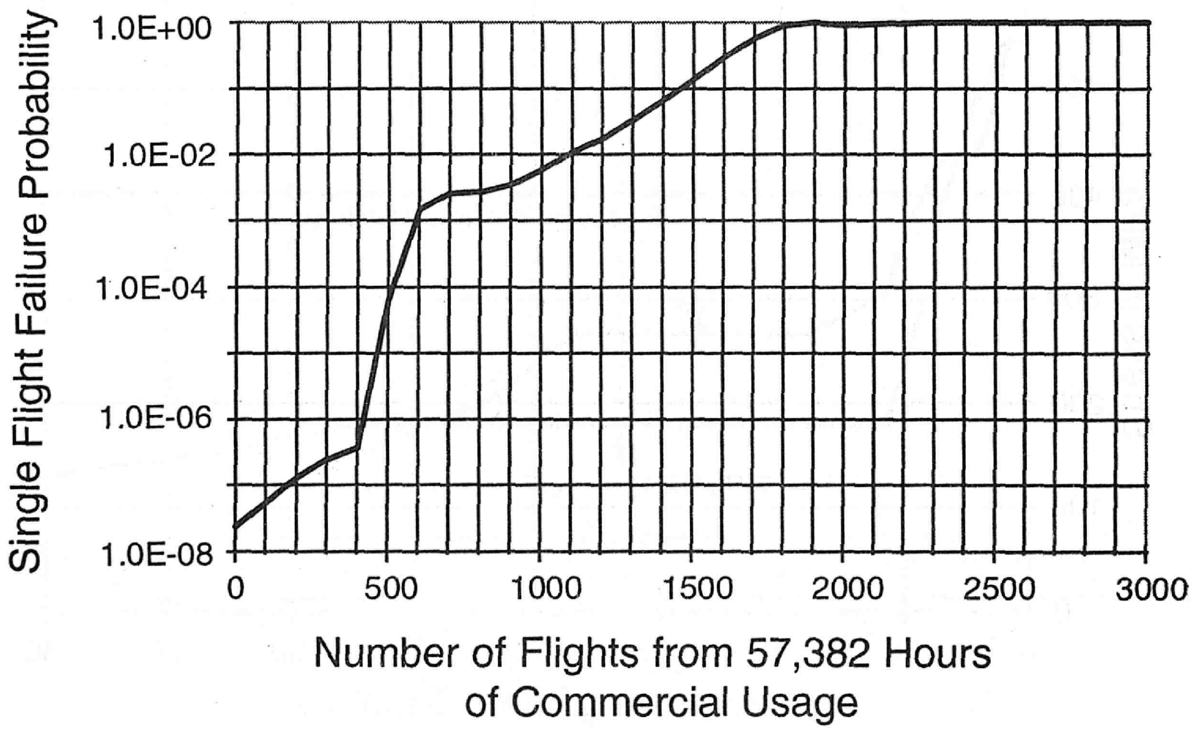

Figure 19 Risk assessment for intact structure

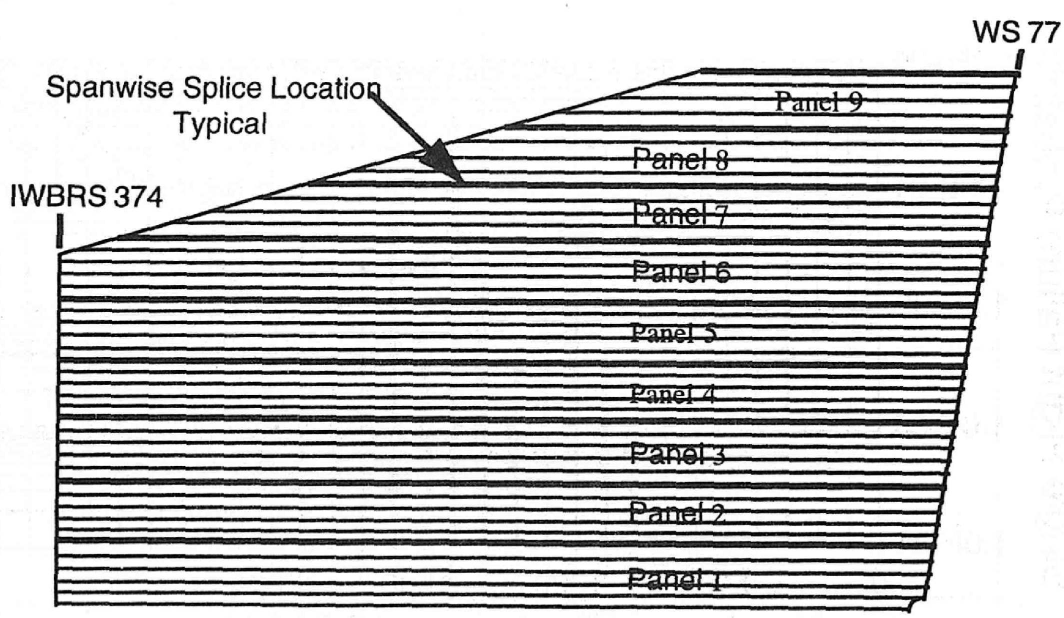

Figure 20 C-141 inner wing spanwise splice locations

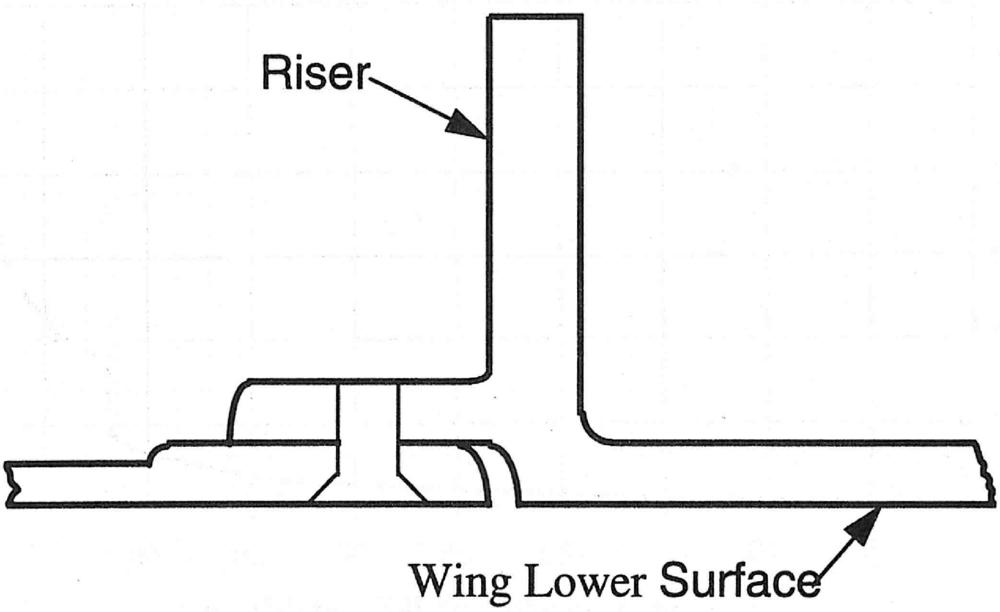

Figure 21 C-141 spanwise splice detail

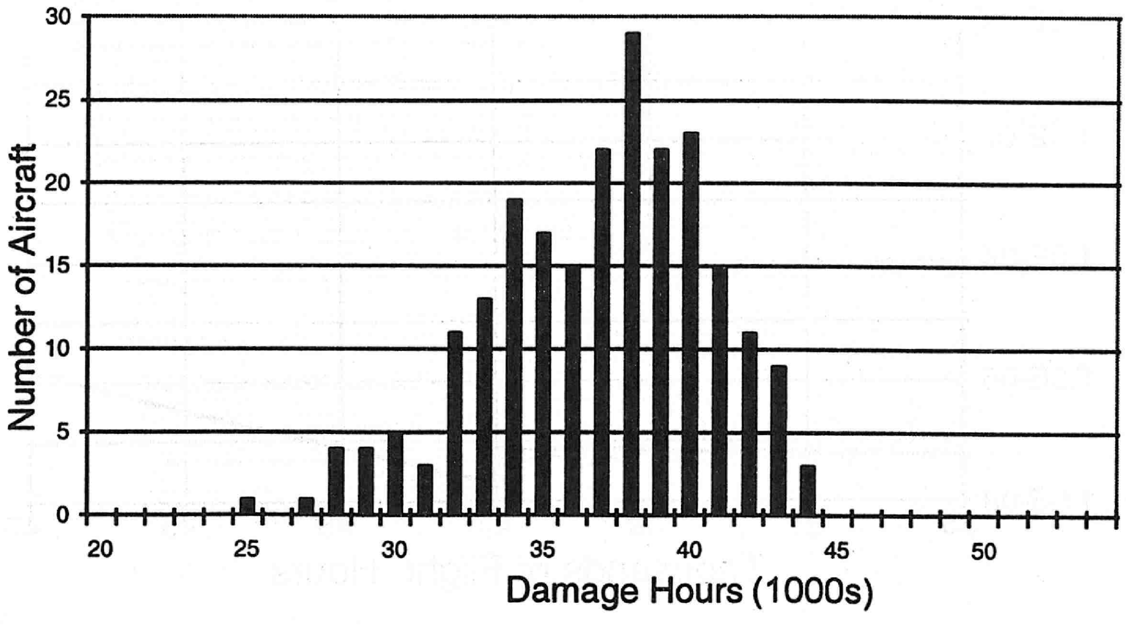

Figure 22 C-141 damage hour distribution

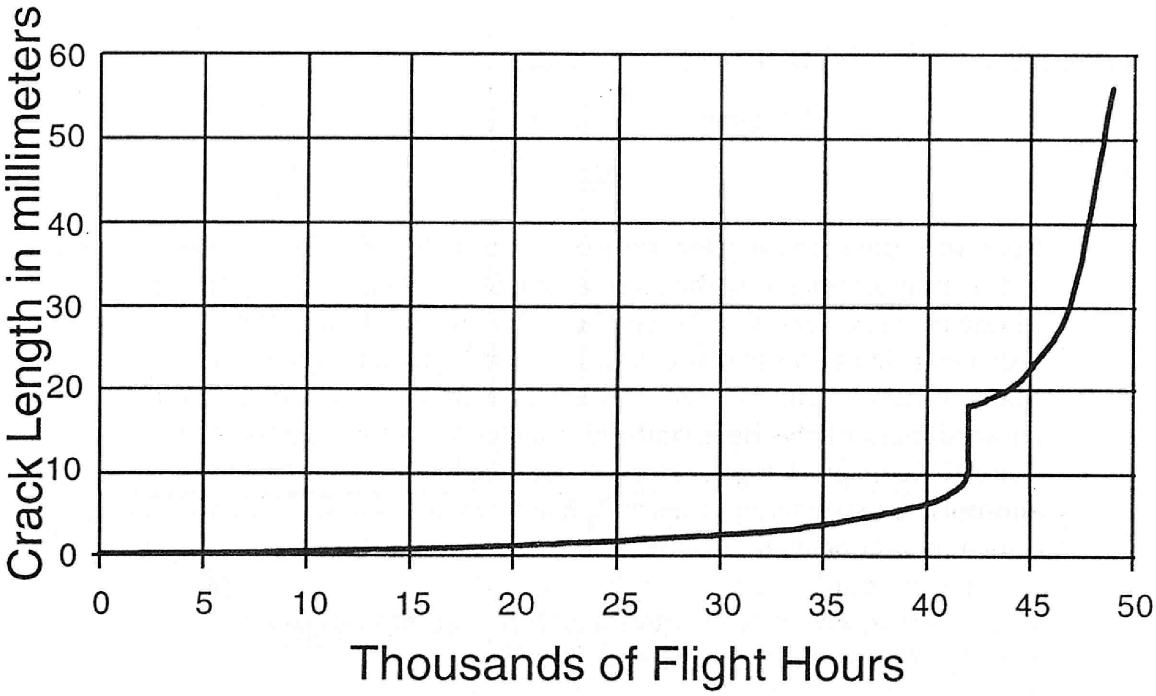

Figure 23 C-141 inner wing spanwise splice crack growth

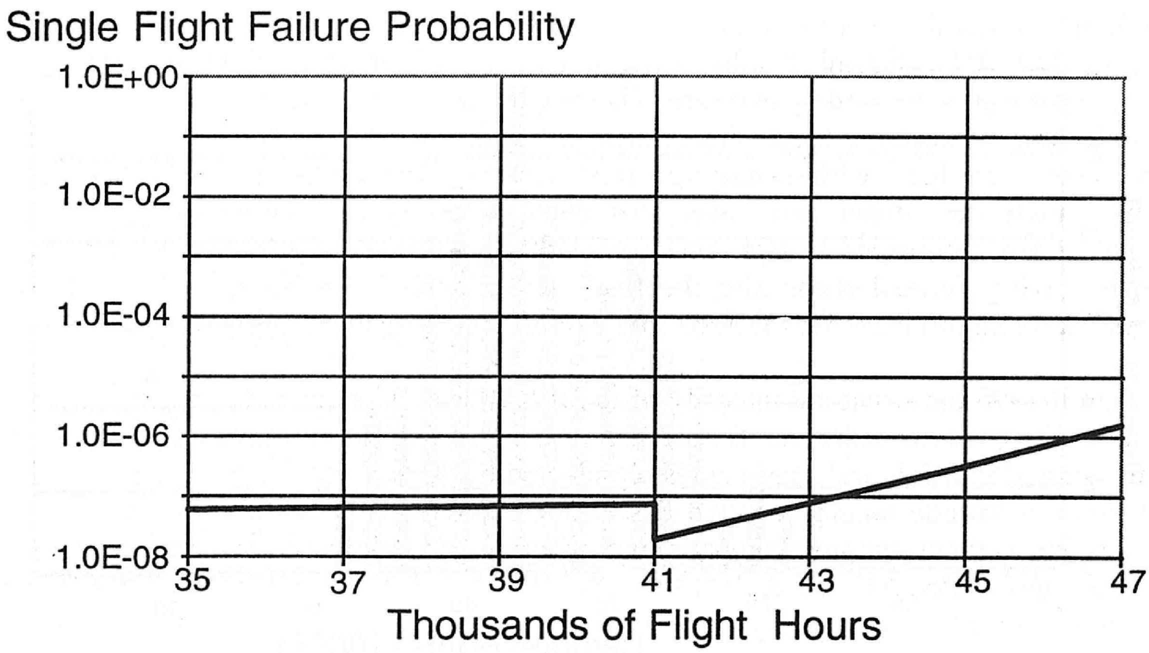

Figure 24 Risk of failure from spanwise splice cracking