EFFECT OF AIRCRAFT FAILURES ON USAF STRUCTURAL REQUIREMENTS†

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Abstract

Structural failures in both commercial and military aircraft have been the primary factor that has changed rules and specifications that engineers use for design. In many cases, the failures have identified threats to structural integrity that were not previously identified by the certification authorities. Commercial failures have influenced the military specifications and the military failures have influenced the commercial aircraft rules. The experience derived from the individual failures is used to describe the lessons learned and to illustrate the evolution of structural criteria that the United States Air Force (USAF) uses for procurement of new aircraft.

1 Introduction

Today, structural failures in USAF aircraft are extremely rare. Except for those failures derived from exceeding the operational envelope of the aircraft, this was true for structural failures until after the mid-forties. One reason for this is early aircraft rarely accumulated sufficient flight time aircraft to suffer from fatigue failures. Further, ductile materials and conservative methods used for analysis tended to preclude failures. Therefore, the designers were led to believe the only threat to structural integrity was failure resulting from loading the aircraft beyond its ultimate strength. The demand for improvements in performance in the late forties introduced new materials with high strength, but few other virtues. Further, the demand for performance improvements reduced analytical conservatism and introduced designs that were to operate in diverse environments. In some cases, the structural engineers needed to design the fuselage for the pressure differential at high altitudes and to design the wings and empennages to operate in turbulence dominated low level regime. However, since they did not understand the implications of these environments, they generally did not properly establish the design of the aircraft components to withstand them. The design community appeared oblivious to the consequences of ignoring threats to structural integrity other than overload. The success they had experienced with earlier designs indicated to them that they were immune to failure from fatigue. Many early textbooks on aircraft structural analysis indicated the designers job was complete when they satisfied the static strength requirements. They made no mention of potential for fatigue failures. This omission resulted in grave consequences.

Increased performance demands pressured the design community to design aircraft at too high of a stress for materials that had marginal ductility. One of the major failures resulting from increased performance demands was the Martin 202. This aircraft failed on 29 August 1948 from fatigue cracking in the wing. Another important commercial aviation failure occurred in the Comet [1]. There were two failures. The first occurred 10 January 1954 and the second occurred on 8 April 1954. These failures ushered in the requirements for fail-safety in large commercial transport aircraft.

Another important commercial aircraft failure was the Boeing 707 horizontal tail failure. This failure demonstrated that structural inspections were required in addition to fail-safe requirements resulting from the comet disasters.

There are many USAF aircraft failures that one could use to illustrate important changes in

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14. ABSTRACT
Structural failures in both commercial and military aircraft have been the primary factor that has changed rules and specifications that engineers use for design. In many cases, the failures have identified threats to structural integrity that were not previously identified by the certification authorities. Commercial failures have influenced the military specifications and the military failures have influenced the commercial aircraft rules. The experience derived from the individual failures is used to describe the lessons learned and to illustrate the evolution of structural criteria that the United States Air Force (USAF) uses for procurement of new aircraft.

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either standards or specifications. One of these is the F-4 accident at Nellis Air Force Base, Nevada on 23 January 1973. This failure conclusively demonstrated to the USAF that a structure was not fail-safe unless there was an inspection program designed to find the broken member before the secondary load path failed. Another important failure occurred in the T-38 aircraft when the training command changed their usage dramatically without advising the maintenance center engineers. Consequently, changes in inspection intervals were not implemented. However, three aircraft had profound effect on the way the USAF accomplishes its mission today. The first is the B-47 that led to the USAF Aircraft Structural Integrity Program (ASIP). The second is the F-111 that led to the adoption of damage tolerance in the USAF. The third is the KC-135 that led to a process for assessing widespread fatigue damage (WFD).

2 The B-47

The catastrophic events leading to the establishment of the USAF ASIP in 1958 [2] are well-documented [3]. Of these events, the B-47 fatigue failures stand out as the most significant. These problems crippled the main striking force of the Strategic Air Command at a time of extreme world tension. The B-47 and other aircraft provided the hard lesson that aircraft designed based on static strength alone would likely not reach their planned service life.

The prototype of the B-47 made its first flight on December 17, 1947. The USAF accepted the aircraft based on a static test of the B-47B in 1950 and started quantity production in 1951. There was a flight load survey demonstration of a B-47B from September 1952 to March 1954. When production ended in 1957, more than 1,200 of these aircraft were serving with the Strategic Air Command at USAF bases throughout the world. The B-47 normally carried a crew of three - pilot, copilot (who operated the tail turret by remote control), and an observer who also served as navigator, bombardier, and radar operator. The maximum mass of the aircraft was 102,494 kilograms and its maximum speed was 981 kilometers per hour. It was powered by six General Electric J47 engines each with a thrust of 32,027 Newtons. The USAF did not specify a service life, but they intended to maintain the aircraft in service until 1965. The engineers based the design on the premise that failure from overload was the only threat to its structural integrity. Fatigue failures of this aircraft demonstrated the fallacy of this premise. On the same day, 13 March 1958, in separate incidents, two of these aircraft suffered failure because of fatigue. Near Homestead Air Force Base, Florida, a B-47B disintegrated at 15,000 feet, three minutes after takeoff. Its center wing section failed at approximately buttock line 45. The aircraft had a total flight time of 2,077 hours at the time of failure. The same day, a TB-47B failed at 23,000 feet over Tulsa, Oklahoma, after the lower left wing failed at buttock line 35. This aircraft had flown 2,418 hours. While the USAF and the contractor were investigating the two 13 March accidents, three more failures followed. This indicated the 13 March failures were not isolated events. The USAF attributed one of these three to overload. They believed the other two were caused from fatigue. One of these aircraft appeared to explode at 13,000 feet just prior to a refueling rendezvous near Langford, New York on 10 April. On 15 April, a B-47E with a total flight time of 1,419 hours took off into a storm from McDill Air Force Base, Florida and disintegrated shortly afterwards.

The USAF approached the solution of this problem through flight restrictions, and a test program as well as additional inspections and instrumentation. They placed flight restrictions on aircraft mass, airspeed, and load factor. The test program included independent cyclic tests in three laboratories. The USAF contracted for these tests at the Boeing plant in Wichita, Kansas, the Douglas plant in Tulsa, Oklahoma, and the National Advisory Committee for Aeronautics (NACA) laboratory at Langley, Virginia. To the surprise of the USAF, the test at Boeing revealed a new fatigue critical location at fuselage station 508. The test article failed after 1,275 cyclic test hours. Although the USAF specified the laboratories to conduct the tests with identical spectra, there were significant differences in the test results. The tests, however, served their purpose by identifying areas that the USAF must modify in operational B-47 aircraft. By the late 1960s, the B-47 was obsolete and was removed from operational service.

The USAF was unsure whether the B-47 failures indicated a chronic problem in all Air Force aircraft. To answer this question the USAF Chief of Staff, General Curtis LeMay gave informal approval on June 12, 1958 to proceed with a program called the “Aircraft Structural Integrity Program” as proposed by the Wright Air Development Center (WADC). The primary objectives of this program were the following:

(a) Control structural fatigue in the operational aircraft fleet.

(b) Devise methods of accurately predicting aircraft service life.

(c) Provide the required design process and test techniques that would avoid structural failures in operational aircraft.

A later policy directive issued by General LeMay on November 19, 1958 formalized the program. The USAF initially defined the detail requirements in ASD-TN-61-141.

The 1958 ASIP included a requirement that an aircraft be designed to withstand the repeated loads expected during its service life. As validation that this requirement was met, the aircraft was subjected to a laboratory (fatigue) test that simulated its expected service loading. The period for which the aircraft would be declared safe to operate (the safe life) was the equivalent flight time to failure in the fatigue test divided by a scatter factor, usually four. Therefore, a fighter aircraft, which a manufacturer designed for an operational life of 4,000 hours would have to successfully pass a laboratory durability test of 16,000 hours.

The safe life approach is based on probabilistic methods. The structural engineer selects materials and designs individual members of the structure based on a failure rate such that operationally no failures are expected within the design service goal for the aircraft. Tests of components representative of production determine the variability of the quality of the structure and consequently the scatter factor needed to interpret full-scale fatigue test results. Either the log normal distribution function or the Weibull distribution function is typically used to represent the reliability of the structure. The development [4] of the reliability function for the Weibull distribution function provides the basis for the reliability calculation.

For aluminum structures, the Weibull shape number is typically in the interval of [4.0,6.0]. Therefore, if the Weibull shape number is 6.0, then the test scatter factor needs to be approximately 3.5 to demonstrate the desired reliability. However, for high strength steel, the Weibull shape number is in the interval of [2.0,3.0]. For a Weibull shape number of 3.0, the test scatter factor may need to as high as 13.0 to demonstrate the desired reliability [5].

Data derived from tests indicate that, in addition to the material influence on the Weibull shape number, there is spectrum content effect on this number. For example [6], test results with a fighter spectrum appear to have a significantly higher Weibull shape number than test results with a constant amplitude spectrum. Further, it is not possible to determine the quality of the test article in relation to the population of production aircraft. The potential exists that manufacturing or service induced damage in some members of the population may violate the basic premise of the reliability concept. All of these influences make it extremely difficult to determine the reliability demonstrated by the full-scale fatigue test.

3 The F-111

The F-111 was originally known as the TFX (Tactical Fighter "X"), and was conceived to meet the USAF requirement for a new tactical fighter-bomber. In 1960, the Department of Defense (DoD) combined the USAF's requirement with a Navy need for a new air superiority fighter, and initiated a competition among aircraft manufacturers for the final design. In 1962, General Dynamics and Boeing were selected as finalists with the General Dynamics TFX design eventually winning. DoD awarded the contract to them on December 21, 1962, to design and build this aircraft. The USAF version was known as the F-111A and the Navy version the F-111B. The F-111A was powered by two Pratt & Whitney TF30-P103 turbofans. The early F-111A engines had extremely bad compressor surge and stall problems. The Navy was never enthusiastic about this program. The F-111B was quite heavy for carrier operations and the Navy cancelled the program when it failed to meet the requirement for handling on an inclined deck. The first flight of the F-111A took place in December 1964, and the first production models were delivered to the USAF in October of 1967. The variable sweep wings were able to move from 16 degrees (full forward) to a sweep angle of 72.5 degrees (full aft). The two crewmembers sat side-by-side in cockpit module that served as an emergency escape vehicle. Using internal fuel only, the plane had a range of more than 4,000 kilometers. External fuel tanks could be carried on the pylons under the wings and jettisoned if necessary. The USAF aircraft was produced in a variety of models, including the F-111A, F-111D, F-111E, and F-111F, as well as the FB-111A strategic bomber version. The FB-111A aircraft had a 1.07-meter increase in wing span and used the TF30-P-7 engine. In all, 563 F-111s of all series were built. All of the F-111s had numerous problems, and only the F-111F actually fulfilled the original TFX design specification. The F-111F had an empty mass of 21,367 kilograms and a maximum takeoff mass of 45,000 kilograms. The F-111F was able to reach Mach 1.2 at sea level and Mach 2.5 at 60,000 feet. All F-111s were retired from the USAF inventory on 27 July 1996.

The demand for high performance from this aircraft placed great demand on the structural mass of the aircraft. The constraints to produce a minimum mass design with a variable wing sweep were met with monolithic wing pivot and wing carry-through box fabricated from heat-treated D6ac steel that included welds. Most components were heat treated to an ultimate tensile strength of 1520-1660 MPa with the upper surface of the carry-through box and access door heat treated to 1790-1930 MPa. These components were typically forged and then welded into the desired assemblies. The manufacturer machined some of them, however, from rolled plate stock.

However, the design, development, and certification of the F-111 structure followed the principles defined in ASD TR 66-57 “Air Force Aircraft Structural Integrity Program: Airplane Requirements.” The USAF published this document following the policy change resulting from the B-47 failures. Consequently, it was qualified for life through the safe life methodology by means of fatigue testing. Many failures slowed the testing of the aircraft, which began in August 1968. After only a few hundred hours, the wing carry-through box failed due to fracture of a Taper-Lok bolt in the rear spar web. The investigation following this failure led to improvements in the installation of these fasteners. The second failure occurred in the wing carry-through box in February 1969. This failure occurred in a straight hole used to mount a bracket. Inspections of 23 similar holes in the structure revealed additional cracking. The contractor modified these holes to accept the fatigue resistant Taper-Lok fastener. The USAF then tested a new test article that incorporated all of the modifications from previous test findings. This article failed in June 1969 with 8,000 equivalent test hours. The failure was located in the outboard closure bulkhead of the wing carry-through box. A simple modification eliminated this problem. The contractor continued testing successfully with a new wing box until the test article reached 16,000 simulated flight hours. The USAF interpreted the test, in which the spectrum was more severe than expected in operational service, as meeting a 6,000-hour life with a scatter factor of four.

The USAF subsequently accepted the aircraft for operational service. On December 22, 1969, F-111A Number 94 (SN 67-049) crashed as a result of a wing failure in the lower plate of the left wing pivot fitting. At the time of failure, the aircraft had approximately 100 hours of flight time. It occurred during a pull-up from a rocket-firing pass at the Nellis Air Force Base, Nevada range. The USAF immediately grounded all F-111 aircraft and sent the failed part to General Dynamics. Metallurgical examination showed that the manufacturing process had produced a defect that served as a nucleation site for a fatigue crack. The principal evidence of this was the presence of a decarburized zone at the surface of the flaw and iron oxides on the fracture face. These findings are characteristic of the high temperatures associated with forging and heat treatment. The metallurgists believed that this flaw was the result of a cooling crack that occurred after the final forging cycle. Normally, the manufacturer detects and removes this type of crack before proceeding with further steps in the fabrication process. The inspection procedures used included ultrasonic pulse echo transmission, conventional magnaflux, and x-ray. Because of the size and unusual geometry of the part, the flux field applied during the magnetic particle inspection was inadequate to cause the migration of the iron particles to the flaw. In addition, the ultrasonic inspection could not detect a flaw with the orientation to the surface as had existed on the crashed aircraft. X-ray was probably ineffective because of the tightness and orientation of the defect. Consequently, the manufacturer did not detect this flaw as it passed through the fabrication process.

The USAF established the Phase I "Recovery Program" in February 1970 to restore the fleet to 80 percent of design capability. This program included a proof test loading of each aircraft to design limit load (+7.33g and -2.4g) at a wing sweep of 56 degrees and a temperature of –40° C. A USAF Scientific Advisory Board Panel introduced the concept of cold proof test. C. Tiffany, a member of this panel, had previously successfully used cold proof testing in the space program for the inspection of steel fuel tanks. This is an extremely useful tool for the inspection of steel structures since cooling reduces the fracture toughness of steel. Consequently, the reduced fracture toughness reduces the critical crack size. Thus, if a structure has a crack of critical size at the reduced temperature, it will fail under the proof load condition. If it does not fail, then the structure does not have a crack of this size or greater. Consequently, when the operating temperature exceeds the proof test temperature, the number of failure-free flight hours may be easily determined.

The USAF funded the building of a proof test facility at General Dynamics in Fort Worth, Texas and at the Sacramento Air Logistics Center, California. The initial proof testing was completed in February 1972 on 340 F-111A/D/E and FB-111A aircraft.

The proof test process was repeated after approximately 1,500 flight hours. Since the proof test program started in 1970, there have been 11 major failures in proof testing. The defects that led to these failures could have caused an in-flight catastrophic failure. When failures occurred during proof testing, the USAF responded as if an in-flight failure had occurred. They initiated fleet-wide inspections and made modifications where appropriate. These proof test failures proved the lack of validity of the safe life methodology for the steel alloys used in this aircraft. Each of the aircraft suffering a proof test failure would have experienced an in-service failure in less time than that validated by previous fatigue testing.

There was some justification for believing the flaw found in the 1969 failure to be unique since the USAF could not reproduce the failure in the laboratory and did not see such a failure on another F-111 aircraft. In addition, if the F-111 incident had been the only failure of the safe life approach, the USAF would have likely rationalized the F-111 wing pivot flaw as a one-of-a-kind event and continued with this method. From other problems in the USAF aircraft, however, it had become evident that the safe life methodology had not precluded the use of low ductility materials operating at high stresses. The safe life also had not recognized that any single aircraft might have "rogue" manufacturing or service-induced defects that could lead to premature failure.

The recovery program for the F-111 was the first major application of damage tolerance concepts to aircraft.

The USAF significantly changed its structural integrity program because of the failure of an F-111 in 1969. Up to that time, the USAF had been concentrating on improvement in reliability methods to maintain flight safety of their aircraft. In addition, they were examining the potential for using the fracture mechanics based approach called damage tolerance [7]. The F-111 failure provided the basis for the decision on which path they would take. It ushered in the era of damage tolerance in the USAF [8]. The first assessments performed on the C-5A and the B-1A in 1971 and 1972 helped derive the original damage tolerance assessment (DTA) requirements for the USAF. These requirements were derived for monolithic (i.e., slow crack growth) structures. The failure of an F-4 wing, previously mentioned, also strongly influenced the damage tolerance requirements as initially established in MIL-A-83444. The technology for the analysis of fail-safe designs has evolved slowly, primarily because of the need for extensive finite element programs supported by expensive test programs. The change to a damage tolerance approach prompted considerable research and development in the area of fracture mechanics. At that time Air Force Flight Dynamics Laboratory was the focal point for much of this research. In the sixties and seventies, they developed much of the fracture technology that is still in use today. In addition, since the damage tolerance approach forced the engineer to better understand the stresses in the structure, finite element techniques emerged as the method of choice for the stress analysis. These capabilities permitted the USAF to perform a DTA of all the major weapon systems in the inventory in the seventies and eighties [9]. This effort required over one million man-hours to complete and every major manufacturer was involved with this activity. Because of this activity, industry was able to develop the technology required for this type of analysis. This technology is also suitable for application to new aircraft developments. Consequently, the USAF was able to include damage tolerance requirements in the specification for new aircraft procurement.

The level of safety afforded by the DTA process has proven through service experience to be high. However, it is evident that the probability of having an initial defect larger than the rogue flaw is considerably less than probability of missing a significant crack in the structure through an inspection. Therefore, for aircraft that are approaching a state of general cracking or their economic life, it is prudent to re-examine the damage tolerance derived inspection intervals to determine if they should be reduced. This may be done by means of a risk assessment [10]. Extensive cracking is not expected to be a problem for recently developed USAF aircraft since they were designed for two lifetimes of slow crack growth from a 1.27 mm initial flaw. This two lifetime slow crack growth guidance is included in the JSSG 2006 specification guide for structures.

A number of benefits have been derived from the DTA process. One of the more significant benefits is that it has placed the USAF in an active mode of problem identification rather than the reactive mode that prevailed in the sixties and early seventies. This has permitted the Air Logistics Centers to make long range plans for inspections and modifications of their aircraft. Further, the Air Logistics Centers now have specific inspection requirements (i.e., the how, when and where to inspect). Consequently, the logistics community has enthusiastically embraced the DTA process. This acceptance has significantly enhanced its success.

4 The KC-135

The KC-135, a tanker aircraft that followed the KC-97, was designed to refuel the B-52 fleet. The KC-135 was derived from an aircraft designated as the 367-80 developed by the Boeing Company with their own funds. The Boeing 707 and 720 were also derived from the 367-80 design. The first flight of a KC-135 occurred 31 August 1956. Five other configurations of this aircraft were delivered before production was terminated in 1965 after 820 aircraft had been manufactured. Thirty-seven different designations of the -135 aircraft now exist. The maximum takeoff mass is 146,285 kilograms and the fuel transfer load is 90,719 kilograms. It is powered by four CFM-International F108-CF-100 turbofan engines in the KC-135R configuration and four Pratt & Whitney TF-33-PW-102 turbofan engines in the KC-135E configuration. The thrust for the CFM engine is 996,233 Newtons and 80,068 Newtons for the TF-33 engine.

The KC-135 did not have a design service life originally specified. In 1962, the USAF made a decision to perform a fatigue test to better quantify the expected life. This test was conducted to failure of the wing at 55,080 test hours of a tanker mission of 5.1 hours. Based on this test, it was believed that a service life of 13,000 hours could be achieved if certain modifications were performed. These modifications consisted of reworking approximately 2,000 fastener holes.

Contradictory to the 1962 fatigue test results, the KC-135 aircraft experienced service problems early in its life. The 7178-T6 lower wing skins were designed with stresses approximately fifty percent higher than the 707 aircraft which had the higher toughness 2024-T3 alloy for the lower wing skins. Consequently, aircraft operating between 1,800 and 5,000 flight hours had experienced fourteen cases of unstable cracking in the lower wing skins. In all, there have been approximately thirty cases on unstable cracking in the range of 1,800 to 17,000 flight hours. The longest of these cracks was approximately 1.1 meters.

In addition to these problems, by 1968, it had become evident that a service life of 13,000 hours would not be adequate for this aircraft. Therefore, in 1972, another fatigue test was performed to determine the actions required for extending the life beyond 13,000 flight hours. This test was significantly more sophisticated than the earlier 1962 test and was more representative of actual force usage. One of the main differences was in the application of high loads. The 1962 test included an application of ninety percent of limit load every 200 flights. The 1972 test included the application of sixty-two percent of limit load every 200 flights. Because of the retardation effect of the ninety percent limit load application, the 1972 test exhibited earlier and more cracking than exhibited by the 1962 test. In fact, 367 cracks were found in the 1962 test article and 1060 cracks were found in the 1972 test article. However, as with the 1962 test, there was poor time correlation of the cracks occurring on the fatigue test article and those occurring in operational aircraft. The wing of the 1972 test article failed catastrophically at 55,505 cyclic test hours. Based on the results of the 1972 test, the USAF determined that the lower wing surface would need to be replaced at 13,000 equivalent tanker hours. The results of this test also alerted them to the possibility that the fail safety of the wing structure could be degraded by the presence of WFD [11].

Since some aircraft were near or already over the threshold of 13,000 equivalent tanker hours, a decision was made to replace the lower wing surface on these aircraft. Further, since the only available design at that time was the original 7178-T6 wing, that material was used for the replacement. This was done for twenty-nine wings. This replacement was questionable because it was accomplished with the same brittle material. However, it is likely that it did enhance the safety of the aircraft and, in addition, provided wings for teardown inspections.

The teardown inspections of six wings removed for a wing skin replacement served as the basis for an assessment of the influence of crack pairs. A crack pair was defined as a primary and secondary crack located such that an unstable primary crack could cause the secondary crack to go unstable and therefore precipitate catastrophic failure of the wing. In other words, the assessment was made for determining the degradation of the fail safety of the wing because of WFD. Finite element analyses of the wing have shown that approximately twenty fastener holes are subjected to significantly higher stresses in the event of failed skin element. If there were a crack of one millimeter in one of these fastener holes then the residual strength would have been reduced to a level considerably below limit load. A risk analysis for the wing was conducted by R. Meadows from the Oklahoma Air Logistics Center located at Tinker Air Force Base, Oklahoma. The database included 245 cracks, each of which were 1.27 millimeters or longer in length. In addition, 29 crack pairs were found which for the purposes of this study were defined as 1.27 millimeters in length for the primary crack and 0.254 millimeters in length for the secondary crack. Meadows used the most critical crack pair from each aircraft for his evaluation. The results of this examination showed that the mean time for a crack pair to develop was 10,709 to 15,441 flight hours with ninety five percent confidence. Based on these results, Meadows performed a risk assessment and found that by the time the fleet of aircraft had reached a life of 13,000 flight hours, he expected, at best, one loss and, at worst, fourteen losses. He also concluded that the degradation of fail safety started at about 11,000 flight hours.

Another teardown inspection of an aircraft with 11,558 flight hours indicated multiple crack alignment at the Wing Station 360 splice. Boeing determined that this splice, which is a known area of high stress, would have failed catastrophically with the application of seventy percent of limit load. In December 1976, a VC-135B was in the depot for replacement of the lower wing surface when it was found that the rear spar chord in the center section was severed and there were adjacent wing skin cracks. This aircraft had 12,400 flight hours. Further teardown inspections of wings that had been in service 8,000 to 10,000 hours revealed that there were aircraft in the fleet that were exhibiting more WFD than that found in the population examined by Meadows. Consequently, the recommendation was made that the lower wing surface should be replaced between 8,000 and 9,000 flight hours and restrictions should be placed on aircraft that were operating above 8,500 flight hours. The replacement was successfully performed without loss of an aircraft.

WFD is a major concern in aircraft since it can destroy the fail-safety that the operator relies on to maintain structural integrity. This phenomenon has occurred on many aircraft, both military and commercial. The USAF has learned WFD can degrade the fail-safe capability of a structure with cracks that are of the order of one millimeter in length [12].

The onset of WFD in a structure is characterized by the simultaneous presence of cracks at multiple structural details, which are of sufficient size and density whereby the structure will no longer meet its damage tolerance requirement. This means that the structure is incapable of maintaining its required residual strength after partial structural failure. In many cases, this deterministic definition is difficult to apply because of the complex cracking scenarios. Further, this definition may lead to an excessively conservative determination of the time of onset of WFD. An alternate definition that removes these problems is the following: The onset of widespread fatigue cracking is that point in the operational life of an aircraft when the damage tolerance or fail-safe capability of a structure has been degraded. The degradation is such that after partial structural failure the probability of failure of the structure falls below the thresholds specified by the procuring (or certification) agency.

For the USAF, the threshold single flight probability of failure for the intact structure is 10^{-7}. The USAF has determined the threshold for the conditional single flight probability of failure through their perception of the discrete source damage threat. In the case of the C-5A, they assumed the probability of discrete source damage was 10^{-3} [12]. For the case of the 707 they assumed it was 10^{-4} [11].

One of the primary inputs to the risk assessment approach to determine the onset of the time to WFD is the distribution of cracks in the structure. The USAF has determined this distribution through teardown inspections of full-scale fatigue test articles or operational aircraft. They believe this is the best method currently available to obtain the data required to derive the probability distribution function for equivalent initial cracks in the critical areas of the structure. The word "critical" here refers to an area that could significantly contribute to the probability of failure.

The probabilistic approach also requires the stress density function for each critical area is determined. The USAF derives this function from the available usage information generated by their individual aircraft tracking programs. The desired stress density function is the one for a single flight of an aircraft selected at random. The structural analyst can easily derive this function from the stress exceedance function developed as a part of the deterministic damage tolerance analysis. One can then compute the joint probability distribution of cracks and stress and integrate this function over the point set where the crack size has reached critical length. The result of this calculation is the single flight probability of failure. The time at which the probability of failure is unacceptable is the onset of WFD.

Therefore, the USAF considers the cracks in the structure and the stresses at the critical locations as random number sets. The crack growth function and the residual strength function are also random functions because of the intrinsic variability of the material properties. Fortunately, the variability of the crack growth and residual strength functions does not appear to have a major impact on the risk of failure. Therefore, the analyst uses his best estimate of the mean of these functions in the risk assessment.

The damage scenarios in an airplane that could constitute WFD differ depending on location in the aircraft. However, typically, they fall into two categories. The first of these is multiple site damage - characterized by cracks in multiple details in the same structural element. The second is multiple element damage where there are cracks in multiple structural elements.

Previous efforts have shown the analyst can readily apply this type of analysis to the structures where the concern is multiple element damage. This was the case, for example, for the KC-135 and the C-5A. The application of the risk assessment technology to the case of multiple site damage is very much the same as it is for the case of multiple element damage. In the case of multiple site damage there will typically be a "boundary" that will determine if the cracking has the potential to become catastrophic. For example in the case of the fuselage lap splice, the boundary would be the crack stopper built into the structure at the frame or between the frames and its surrounding structure. This crack arrest feature protects the integrity of the structure. The condition of the crack stopper and its surrounding structure (that is, the boundary) will determine if the damage could propagate to catastrophic failure. Therefore, the interest is primarily in the degradation of the boundary with time and not the growth of the holes in the lap splice to link-up. Therefore, multiple site damage may be evaluated in the same manner as multiple element damage. Lockheed [13] demonstrated an example this of this in their risk assessment on of the inner to outer wing joint of the C-141 aircraft.

Emphasis must be placed on the detection, through nondestructive evaluation, of cracks that could be significant for determination of the onset of WFD. As indicated above, there is a need to make an estimate of this onset based on probabilistic assessment of cracking data derived from the teardown inspection of fatigue test articles or operational aircraft. The analyst must recognize, however, that this is only an estimate. It is not realistic to expect analyses to determine this time with great accuracy even with the most sophisticated fracture mechanics programs. The actual time may be either somewhat earlier or later than this estimate. It is important, therefore to be able to validate this prediction with nondestructive evaluation. This is difficult because the size of defect the inspector must find is quite small. As indicated above, the experimental evidence to date indicates cracks of the order of one millimeter can significantly lower the fail-safety capability of certain structural configurations.

5 Conclusions

The USAF ASIP as originally conceived after the B-47 failures did not accomplish its goal of ensuring safety of operational aircraft. They learned that the use of the reliability approach they adopted, led to catastrophic loss of aircraft and was a large economic burden to correct mistakes made. In hindsight, they realized that an accurate determination of the reliability could not be ascertained through this approach. In response to the problems with probabilistic methods for ensuring safety, they moved into a more deterministic approach with the adoption of damage tolerance. However, damage tolerance is not completely removed from probabilistic methods. The reason is that the approach requires that determination of the initial flaw size for the damage tolerance calculations and, in addition, the inspection capability is based on a probability of detection of 0.9 with a 95 percent confidence. Experience has shown that damage tolerance inspections may not preclude widespread fatigue damage in an aircraft. The most logical approach to determine when in the life of an aircraft when this problem occurs is to use probabilistic methods. Consequently, probabilistic methods have an extremely important role to play in achieving safe and economic operation of an aircraft. However, their limitations must be clearly understood so that they will be used appropriately.

On the positive side, one could conclude from the examples that the USAF does learn from its mistakes. On the negative side, one could conclude that it takes a major catastrophic failure or an economic disaster to make the USAF act. A common thread among all of the failures is that structural integrity is very fragile, and mistakes can compromise it quite easily. Usually, the reason for mistakes is that the designer is unaware of the threats to structural integrity or does not consider them in the design process. The USAF is fortunate to have formalized the current version of the ASIP. This process, when followed, has been the key to virtual elimination of failures from strength, stiffness, or fatigue causes.

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