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AGING AIRCRAFT AND FATIGUE FAILURE

P.F. PACKMAN*

I. INTRODUCTION

ALL AIRCRAFT are subjected to loads and environments that will reduce the structural strength of critical components of the airframe and engine over time. Recent inflight structural failures of commercial aircraft focused the public's attention on the technical problems of aging aircraft.

In June 1988, an accident involving an Aloha Airlines 737 jet¹ and several incidents in December 1988 involving Eastern Airlines 727 aircraft² prompted media and indus-

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¹ The Aloha Airlines 737 was flying at 24,000 feet when the entire upper fuselage ripped away. Ott & O'Lone, *737 Fuselage Separation Spurs Review of Safeguards*, AV. WK. & SPACE TECH., May 9, 1988, at 92. The accident occurred because of a failure in aging stringers, which are the stiffeners in the fuselage. *Id.* The Federal Aviation Administration (FAA) responded by issuing directives ordering increased inspection of the 737s. *Id.* More than 350 aging 737s are affected by the directives. *Id.* at 93. The only fatality in the Aloha Airlines incident was a veteran flight attendant who was swept out of the aircraft when the fuselage failed. *Id.* at 94. Since the accident, Aloha Airlines retired all four of its Boeing 737s, even though the company believes that the aircraft, which have 70,000 to 90,000 flight miles, are airworthy. *Safety of Aging Aircraft Undergoes Reassessment*, AV. WK. & SPACE TECH., May 16, 1988, at 16. Apparently the Aloha 737s are more susceptible to problems because they are used for short, over-water routes, which cause more corrosion on the aircraft. *Id.* For further details of the accident, see Shapiro, "The Plane was Disintegrating", TIME, May 9, 1988, at 38.

² One of the Eastern Airlines accidents involved a 727 that lost pressure at 31,000 feet after a hole opened in the upper rear part of the fuselage. *Investigators Suspect Faulty Repair as Cause of 727 Fuselage Failure*, AV. WK. & SPACE TECH., Jan. 2, 1989, at 107. The crack in the 727 developed at a lap splice where the metal portions of the fuselage are riveted together. *Id.* The older aircraft in the Boeing

try-wide reassessments of the issues relating to the safety of aging commercial aircraft. The Aloha and Eastern Airlines incidents involved cracking of the fuselage structure, which resulted in rapid decompression and structural damage.³ In the Aloha Airlines case, a portion of the top half of the fuselage section detached from the aircraft at 24,000 feet.⁴ The aircraft was an early model 737-200 that had accumulated approximately 88,000 takeoff and landing cycles in its twenty-year history.⁵ The Eastern Airlines 727s, one of which developed a fourteen-inch crack that forced an emergency descent from 31,000 feet, were also approximately twenty years old.⁶

The Aloha Airlines jet was an early model that combined a "cold adhesive bond" with three rows of countersunk rivets in the design of the fuselage splice.⁷ Figure 1 in Appendix A illustrates the areas of the aircraft involved.⁸ Boeing later changed from the cold adhesive

fleet used a cold-bonding process to lap splice the fuselage. *Id.* This process came under attack after the Aloha Airlines incident, prompting the FAA to issue directives to alert inspectors to carefully examine the process. *FAA Proposes Checks for Cracks on 727s*, AV. WK. & SPACE TECH., Jan. 16, 1989, at 63. The FAA proposed an extensive rivet replacement program for the Boeing 737 and considered proposals for the 727 and 747. *Id.*; see also *FAA Safety Directives Issued for Boeing 737s*, Dallas Morning News, Oct. 28, 1988, at 1A, col. 1 [hereinafter *FAA Safety Directives*].

³ See *supra* notes 1 and 2 for a discussion of the structural failures of these jets; see also *Incidents Spur Concern About Age of Planes*, Dallas Times Herald, Dec. 28, 1988, at A-3, col. 1 (providing further information regarding the structural failures).

⁴ See *supra* note 1 for a discussion of the Aloha Airlines accident.

⁵ See *supra* note 1 for further discussion of the Aloha Airlines accident.

⁶ See *supra* note 2. Eastern has had prior problems with its aircraft. See, e.g., *Eastern DC-9 Splits Open in Hard Landing at Pensacola*, AV. WK. & SPACE TECH., Jan. 4, 1988, at 68; *727 Emergency Exacerbates Beleaguered Eastern's Woes*, Dallas Morning News, Dec. 28, 1988, at 4A, col. 1.

⁷ *FAA Safety Directives*, *supra* note 2, at 12A, col. 5. The countersunk rivets are used to join the upper and lower aluminum skins of the aircraft shell in a manner similar to shingles on a roof. This design is especially susceptible to cracking when the rivet angle changes in flight. *Id.*

⁸ *FAA to Require Fuselage Repairs in Older 737 Jets*, N. Y. Times, Oct. 28, 1988, at A1, col. 3. The rivets indicated in Figure 1 join the upper skin to the lower skin near stringers, which are metal strips forming part of the shell of the airplane. The old countersunk rivet tends to cause cracks in the skin where the rivet angle changes, while the newly specified buttonhead rivet eliminates any changes in the rivet angle. Note that only the top row of rivets in the three existing rows is

bond to a hot adhesive bond after manufacturing its 291st aircraft. Boeing implemented an improved hot bonding after manufacturing its 464th jet, which apparently corrected the problem.⁹ No subsequent significant cracking or disbonding has been reported in the 737. The Federal Aviation Administration (FAA) issued rules in October 1988 to modify the structure of older 737s by replacing the first row of rivets with buttonhead-type rather than countersunk rivets.¹⁰

In its review, the FAA also stated that some early models of the Boeing 727 and 747 incorporated cold adhesive bonding techniques as well as countersunk rivets. The FAA is considering issuing airworthiness directives for these aircraft as well.¹¹ The Aloha Airlines accident prompted industry-wide reassessment of the technical, managerial, and economic issues relating to aging aircraft. The FAA sponsored an industry-wide conference in June

targeted for replacement. *Id.* Figure 1 in Appendix A illustrates the buttonhead rivet, which is not as susceptible to angle change as the countersunk rivet discussed in note 7.

⁹ See *FAA Safety Directives*, *supra* note 2, at 12A, col. 5.

¹⁰ Airworthiness Directives: Boeing Model 737 Series Airplanes, 53 Fed. Reg. 44,156 (1988) (to be codified at 14 C.F.R. § 39.13). The new directives include several modifications of procedures for external inspection of the fuselage skin at lap joints: (1) substitution of high frequency eddy current inspections for the current visual inspections for cracks; (2) paint stripping prior to inspection unless the rivet fastener is clearly visible through the paint and there are not more than two coats of paint on the airplane skin; (3) chemical stripping of paint rather than sandblasting; (4) retention of the threshold for external inspection of the lap joints at 40,000 landings for the first 291 Boeing 737s manufactured; (5) other modifications of timing for internal inspections; and (6) restriction of cabin pressures until initial inspections are completed to reduce stresses on fuselage skins. *Id.* at 44,156-58. These directives became effective November 21, 1988. *Id.* at 44,160. A similar directive concerning external inspection of circumferential fuselage splices and internal inspection of certain bonded doublers for delamination, cracking and corrosion was issued at the same time. See Airworthiness Directives: Boeing Model 737 Series Airplanes, 53 Fed. Reg. 44,160 (1988) (to be codified at 14 C.F.R. § 39.13).

¹¹ Airworthiness Directives: Boeing Model 747 Series Airplanes, 54 Fed. Reg. 7446 (1989). This proposal constitutes a new airworthiness directive requiring inspection of skin joints in the fuselage upper lobe for skin cracks and corrosion. This new proposal was prompted by service experience indicating that the cold adhesive bond used in the first 200 Boeing 747s had disbanded on some models. *Id.* Comments were received by the FAA through April 1, 1989, with codification later in 1989. *Id.*; see also *FAA Safety Directives*, *supra* note 2, at 1A, Col. 1.

1988 to discuss these issues.¹²

Figure 2 in Appendix A depicts the age of some of the older commercial aircraft presently in use.¹³ The average age of these older aircraft is about twenty years. Even with the introduction of recent aircraft designs, such as the Boeing 757 and 767 and the Airbus A320, the average age of the United States commercial fleet is more than 12.5 years. It is estimated that approximately 2,300 jetliners in service are over twenty years old.¹⁴ Aviation industry engineers and government regulators acknowledge that the commercial jet fleet is growing older. They further acknowledge that added precautions will be needed to monitor the cracking and other problems that can develop on these older jets. No one will agree or will comment, however, on real or potential loss of safety for these aging aircraft.¹⁵

The difficult technical problem is determining what can be done to improve or maintain the safety, reliability, and durability of aging airplanes. The cost of replacing them is extremely high. The waiting list for new aircraft is so long that current aircraft must continue to fly for at least five years. The cost of significantly increased maintenance can become prohibitive; yet the cost of failure is unacceptable. Two questions face aviation specialists. The first is whether there is a loss in safety for aging aircraft. The second is whether current technology, inspections, and maintenance programs can cope with the aging aircraft problem.

This paper presents a review of some of the technical problems associated with maintaining an adequate level of

¹² *FAA to Address Aloha Conference*, *AV. WK. & SPACE TECH.*, May 23, 1988, at 103.

¹³ Many early models of the Boeing 707 (471), McDonnell Douglas DC-8 (350), and Lockheed L-199 (53) are still in service and will be 30 years old in 1989. *Incidents Spur Concern About Age of Planes*, *Dallas Times Herald*, Dec. 28, 1988, at A-3, col. 1.

¹⁴ See generally Fischetti & Perry, *Our Burdened Skies*, *IEEE SPECTRUM*, Nov. 1986, at 36 (providing background information on design, aircraft maintenance, air traffic control systems, and wind shear detection advances, as well as design and manufacture standards for aircraft).

¹⁵ *Id.* at 73-74.

safety and reliability for aging aircraft. The review considers three major areas: (1) the safety inherent in the design of critical load-carrying aircraft structures; (2) the inspection maintenance schedules; and (3) the ability to increase the durability of the aging aircraft.

II. TECHNICAL BACKGROUND: DESIGN

A. *Fatigue Design for Airframes*

The primary load-carrying structure of an aircraft sees a large number of loads that vary during each flight. These loads can be classified roughly as (1) ground loads, (2) low- and high-speed taxi loads, (3) takeoff loads, (4) gust loads, (5) maneuver loads, and (6) landing loads. Each of these loads is representative of the particular phase of operation that the aircraft structure experiences. One complete sequence of loads during a single flight is called the ground-air-ground (GAG) cycle.

The design of an aircraft component is based on the ability to predict the time or number of load applications over which the component parts of the airframe will be able to resist these loads. Consider a simple example of a pressurized aircraft fuselage. During each flight the fuselage is pressurized with a pressure differential depending on the altitude of the aircraft. When it is on the ground with its doors open, the pressure differential is zero. The differential increases as the aircraft gains altitude. Thus, the fuselage sees at least one zero-maximum-zero pressure cycle for each flight.¹⁶

The fuselage is loaded with the GAG cycle for each flight. Each of the materials, joints, connections, and other parts of the structure is subjected to a phenomenon called *fatigue*. The fatigue of materials is a progressive degradation and loss in a structure's load-carrying capac-

¹⁶ This structure also experiences loads during takeoff and maneuvering that are introduced through the wing structure or the landing gear system. For the purposes of this analysis, however, the GAG cycle will be the only loading considered.

ity due to the initiation and growth of small crack-like flaws within the structure. Fatigue has been the subject of continuous and strenuous research. While a great deal of information is known about the phenomenon, at present many aspects of fatigue are not completely understood.¹⁷

The possibility of fatigue failure in a pressurized fuselage was first recognized by the loss of an early de Havilland Comet 1 aircraft.¹⁸ These types of failures were determined to be due to fatigue cracks growing in the skin originating out of a window cut-out. The cracks grew slowly for a period of time because of the then unprecedented cabin pressure differential of nearly nine pounds per square inch combined with the fatigue cycles imposed by the GAG cycles. When the crack reached a certain size, it propagated rapidly, destroying the fuselage structure and the aircraft. A similar problem occurred in a cargo transport aircraft, where pressure pulses from the tips of the propeller blades caused the fatigue cracking.¹⁹ Fatigue failure originating in an improper repair of the rear pressure bulkhead, and subsequent initiation and growth of a fatigue crack by the GAG pressurization cycle caused the recent crash of a Japan Airlines 747.

Fatigue failures are not limited to the fuselage structure. Wing structures may fail because of fatigue caused by maneuver loads or landing loads. In addition, landing loads or taxi loads may cause fatigue in landing gear structures. Fatigue failures of wing flap, aileron, elevator, and rudder hinges and other major structural compo-

¹⁷ See generally C. OSGOOD, *FATIGUE DESIGN*, (1982), for an explanation of the different technical approaches to determine the most appropriate way to measure individual characteristic responses to fatigue. The book presents a practical method for the prediction of fatigue life under different stress conditions and highlights a prediction of fatigue approach that gauges potential damage by the numerical crack-life standard.

¹⁸ Address by L.J. Hart-Smith, *Adhesive Bonding of Aircraft Primary Structures*, to the Society of Automotive Engineers, Inc., Aerospace Congress & Exposition in Los Angeles, Cal. (Oct. 13-16, 1980) [hereinafter Hart-Smith] (available from Professor Packman at Southern Methodist University).

¹⁹ *Id.* In the McDonnell Douglas C-133 Cargomaster transport aircraft, pressure pulses from the propeller blades, rather than structural stress concentration, caused the cracks. *Id.*

nents that are subjected to varying loads also have been reported. Furthermore, the turbine engine and other rotating components, blades, disks, and gears are subject to fatigue failure. Propellers, flight instruments, and other equipment may also see fatigue cycles. Fatigue combined with corrosion presents a more serious problem. It is estimated that more than seventy percent of the cracking that develops in aircraft is due to fatigue or fatigue-corrosion. Most structural failures in cyclically loaded structures, such as helicopters, automobiles, and power plant equipment, are due to fatigue, with or without corrosive environmental effects.

Figure 3 in Appendix A illustrates the general fatigue problem associated with aircraft. This curve plots a measure of strength, which is in pounds per square inch versus time or cycles of loading. The one-time-load or static strength of the typical structure is given as the ultimate load for that structure. For a large number of applied cycles, no decrease is observable in the residual or remaining static strength of the structure. As the number of applied cycles of load increases, the material begins to lose strength due to the cyclic loading. The residual static strength begins to decrease, slowly at first, then with increasing rapidity. When the value of the residual static strength falls below eighty percent of the structure's original unfatigued residual static strength of the structure, the structure is considered unsafe and must be withdrawn from service. Withdrawal is necessary because the component might experience a load equal to eighty percent of the original residual static strength of the structure sometime during a single flight. If this load were encountered, the structure now weakened by the cyclic loading would not be able to withstand the applied loading and would fail.

In Figure 4 of Appendix A, one can plot the growth of a defect that is responsible for the loss in residual static strength. The defect, whether a single crack or multiple cracks, may be either present in the material when the

component is introduced into service or initiated in the material after a period of usage. The crack size is plotted as a function of the time or cycles of applied load. For the initial portion of the life of the component, the cracks are either stable or growing very slowly. After a period of time the growth rate of the crack increases, and the increased rate corresponds to the decrease in residual static strength, resulting in the curve shown in Figure 3 of Appendix A. Thus, the determination of the life of the component due to cyclic loading is related to the initiation and growth of crack-like defects within the structure.

Three concepts provide the basis for the design philosophy used to minimize the potential for premature failure by fatigue of commercial, general aviation, and military aircraft: (1) safe life; (2) fail safe; and (3) damage tolerance.²⁰ Durability and economic life analysis are two additional concepts taken into account to complete the U.S. Air Force's approach to structural integrity.²¹ These concepts are important for understanding the problem of an aging commercial aircraft fleet, and the need for future aircraft designs.

1. *Safe Life*

The concept of safe life in fatigue design is predicated on the assumption that scatter exists in the fatigue life. An analysis of Figure 4 in Appendix A shows that predicting the increase in crack size over a period of time is

²⁰ See Ekvall, Burssat, Liu & Creager, *Preliminary Design of Aircraft Structures to Meet Structural Integrity Requirements*, 11 J. AIRCRAFT 136 (1974) (concluding that existing methodology has been demonstrated as sufficient to permit damage tolerance criteria to be formally considered in the design of primary aircraft structure); see *infra* notes 22-25 and accompanying text for a discussion of safe life, damage tolerant, fail safe, and slow crack growth design concepts.

²¹ See Gallagher, Grandt & Crane, *Tracking Potential Crack Growth Damage in the U.S. Air Force Aircraft*, 11 J. AIRCRAFT 435 (1978) (within the concept of economic life analysis is the "current U.S. Air Force ASIP [Aircraft Structural Integrity Program] Force Management policy [which] is based on a desire to anticipate and control cracking problems throughout the service life of the airframe structure."); Gallagher & Stalnaker, *Developing Normalized Crack Growth Curves for Tracking Damage in Aircraft*, 11 J. AIRCRAFT 114 (1978) (the Air Force evolved a design philosophy which assumes that cracks are present initially in airframes).

based on the knowledge of the cyclic loads that produce the incremental crack propagation. For a single structure in which the GAG pressurization cycle is well defined, such as a pressurized fuselage, this knowledge may be principally deterministic. The analysis would not be as simple for a wing structure or a landing gear structure whose actual loading and loading sequence is determined by random factors such as gust loads or rate of descent to touchdown. In many cases the magnitude, number, and sequencing of the fatigue cycles must be estimated before the aircraft is designed. This concept of a "mission load-exceedence profile," which is used to determine the fatigue cycles on a component of the aircraft, is a continuous design reiteration process that continues well into the aircraft's service life.²²

In such a random loading, the life or number of cycles to the point where the residual static strength has decreased would not result in a single value. Instead, the calculations would exhibit considerable scatter about some predicted mean life. Figure 5 in Appendix A plots the life, either in time or cycles, against the probability of failure. The mean life prediction is based on unflawed laboratory specimen data and a fatigue cumulative dam-

²² See C. Heikkinen, *Improved Aircraft Readiness Through Advanced Durability Analysis and Improved Material Quality*, (April 29, 1988) (unpublished report to Professor Paul Packman for CME 6362, available from Professor Packman at Southern Methodist University). Heikkinen's paper discusses durability analysis, which is a means of estimating crack growth damage for a population of structural details. See generally Ott, *Independent Research Urged for Inspection, Maintenance*, *Av. Wk. & SPACE TECH.*, June 20, 1988, at 112 [hereinafter *Independent Research Urged*]. Ott relates a discussion with Ray Valeika, vice president of maintenance and engineering at Pan American World Airways. Valeika believes that aircraft have the potential of operating safely within a system of government and industry regulation. *Id.* at 113. Although many believe that aircraft maintenance is reactionary in nature, the present system is actually a continual process of inspection, with the key being finding the first crack. *Id.* at 112. Ott describes some of the different levels of inspection that Pan American implemented in an effort to ensure that its aircraft operate safely. *Id.* at 113. Although Valeika believes that safe operations can be achieved through government regulation, he thinks that an agency other than the FAA should undertake the necessary research and development programs. *Id.* He also believes that one agency "shouldn't do the testing and make the rules." *Id.* (emphasis added).

age analysis called Miner's Rule. To ensure that the actual component will not fail, the useful design life is considered to be one-fourth of the predicted mean life. Thus, if we want an aircraft fuselage to withstand 10,000 GAG cycles, the structure should be designed to withstand 40,000 GAG cycles. To verify the design, the structural components are tested in a controlled laboratory environment to verify the 40,000 GAG cycle life.

A scatter factor multiple of four is assumed to account for the effect of variations in: (1) the initial quality of the manufactured components and materials; (2) the effects of the environment; (3) variations in material properties due to production and passage of time; and (4) variations in the load levels and load sequences encountered by the aircraft itself.²³

The test article for acceptance testing is an actual aircraft, whose acceptance criterion will be no failure in four lifetimes. The structure tested for acceptance is monitored carefully during the test period, and often well beyond the initial acceptance test period. The time of appearance and location of any cracks or defects in the exemplar structure is noted. These problem areas are scheduled for maintenance inspections at regular intervals, which usually are equal to one-fourth of the time the cracking in the exemplar test structure is noted. The length of these inspection intervals is maintained, or decreased, for the remainder of the aircraft service life. Repair procedures are designed for each specific area. If the problem appears widespread, a modification or redesign is carried out. Current FAA requirements for the redesign and repair of the 737 splice structure involved in the Aloha Airlines incident are adaptations of such a procedure.²⁴

²³ See, C. Heikkenen, *supra* note 22, at 8-12 (discussing mathematical calculations and formulas necessary for the prediction of crack growth and fatigue). The extent of crack damage can be estimated quantitatively at any service time by using advanced durability analysis. *Id.* at 13.

²⁴ See Ott, *Airlines, Manufacturers Propose Plan to Ensure Safety of Aging Fleet*, Av. Wk. & SPACE TECH., June 6, 1988, at 88 (discussing the task-oriented program pro-

The major shortcomings of the safe life design concept are significant. A test article can exhibit a life considerably longer than any individual aircraft in the fleet. Figure 6 in Appendix A demonstrates an example of poor correlation between test article life and actual service life. Fleet structures are designed so that a test article has a life of 400,000 hours. The acceptable design life for the fleet itself is 100,000 hours of operation, which amounts to about fifteen years of operation with a scatter factor of four on design life. A single structural component, however, may contain a small flaw that can rapidly grow to a size capable of causing early failure well before the end of the 100,000-hour design life. For example, in several military aircraft, a number of catastrophic structural failures that were caused by initial crack-like defects were not detected during manufacturing or service inspections. Figure 7 in Appendix A summarizes examples of early service life failures.

2. *Damage-Tolerant Design*

The safe life approach is still the primary design process for the sizing and analysis of aircraft components. The next step in the development of safe designs for aircraft incorporates the concept of "a damage-tolerant design." This concept assumes that the production component contains crack-like defects when it is initially introduced into service. The design is predicated on the structure's tolerance to the presence of damage, which is its ability to withstand the presence of these defects.

posed by the airlines and manufacturers at an FAA conference on the safety of aging commercial aircraft); *see generally* Ott, and O'Lone, *supra* note 1, at 92 (discussing the Aloha Airlines accident); *see also* FAA Safety Directives, *supra* note 2, at 1A, col. 1, (discussing the FAA directives that resulted from the Aloha Airlines accident, which "call for an altitude restriction of 26,000 feet for all Boeing 737s that have completed more than 40,000 landings until the planes pass inspection . . ."). The Aloha Airlines accident caused the government to reassess maintenance and inspection programs for aging aircraft. *See supra* note 1 for further discussion of the Aloha Airlines accident.

Damage-tolerant design is divided roughly into two general areas: (1) fail-safe and (2) slow crack growth.

a. *Fail-Safe*

Figure 8 in Appendix A illustrates the basic concept of a fail-safe structure with four tension-loaded panels. The structure is designed to contain a single member failure without resulting loss of the aircraft. In this case, the structure is designed to survive a crack that grows completely across one of the four panels. Lower wing surface panels, fuselage stringer skin structures, and horizontal and vertical fin attachment structures are all designed to be fail-safe. Fail-safe structure allows for crack growth up to a certain size, as illustrated in Figure 8 in Appendix A. These cracks should be detected and repaired before further cracking occurs. If some crack growth remains undetected, there should be sufficient backup or redundant structure to carry the total load. Certain aircraft components obviously cannot be made fail-safe, including hinges, landing gear systems, and flap tracks. These must be designed by alternate procedures.

Full-scale and component fatigue testing for fail-safe design includes both constant amplitude and spectrum loading. This results in the best comparison of the estimated life to the actual life of components, as well as verification of fail-safe performance. Many representational structural components are tested against a flight-by-flight spectrum, usually prior to aircraft certification. Any cracks discovered during these tests are allowed to progress in order to obtain crack growth rates and to verify that the structure is fail-safe. Repairs for damage incurred during the test are designed to restore the structure's fatigue and static strength and are usually incorporated into the repair manual for the structure.

b. *Slow Crack Growth*

Slow crack growth design assumes that *initial defects are*

present in the structure at the inception of service.²⁵ It further assumes that fatigue created during the service GAG cycles will grow to a point where these defects might eventually cause failure or require replacement of the component. The structure is designed and inspected to prevent the maximum anticipated initial damage from growing into a critical size during the service life. The slow crack growth design is illustrated in Figure 9 in Appendix A. This design is typical of components that cannot be designed to fail safe criteria.

The critical size that would cause the failure of the component at eighty percent of the residual strength load is calculated. The required design life is calculated to be approximately eighty percent of the estimated time to failure. The fatigue mission spectrum is used to calculate the rate of crack growth per hour of flight, using the fatigue crack propagation properties of the material. The number of flight hours for each individual aircraft is tracked, and the aircraft is repaired or withdrawn from service at the designated design life.

The structure's safety depends upon the assumption that no flaws larger than the maximum initial size used in the calculations will form. Figure 10 in Appendix A shows values that are used for the size of the initial damage used in the design. Several different levels of initial damage are allowed, depending upon the structure's level of criticality. If at the onset of service, a flaw in a part is larger than the flaw sizes assumed in the initial design, that flaw may cause the part to fail before the required design life expires. To ensure that no parts with flaws larger than

²⁵ See Gallagher & Stalnaker, *supra* note 21, at 114 (discussing the Air Force design philosophy which assumes that cracks in an airframe are present initially). See generally *Independent Research Urged*, *supra* note 22, at 122 (discussing mathematical calculations and formulas necessary for the prediction of crack growth and fatigue). Airlines are implementing new inspection, maintenance, and repair procedures. For example, a B-Level inspection for corrosion and fatigue damage takes place every 1,000 hours. *Id.* at 113. A more detailed D-Level inspection, which involves replacement of aging parts, occurs approximately every four years. *Id.* Knowing that initial defects are present in a structure allows for incorporation of the service involved in the B-Level and D-Level inspections. *Id.*

the initial maximum design flaw size are utilized, a fracture control program is used to inspect each part through nondestructive inspection (NDI). A careful analysis ensures a high probability that any flaw larger than the initial flaw will be found. Figure 11 in Appendix A shows a curve determining the probability of detecting crack-like defects as a function of the defect size by NDI inspection during manufacturing. Obviously, larger flaws have a higher probability of being detected.

At best, the calculation of fatigue-induced crack growth behavior is complex. It must include a wide spectrum of variables, including loads, structural geometry, structural response, material properties, and level of manufacture. Each of these variables affects the initiation time and rate of crack propagation. The variability in the response of materials to a complicated stress-time-temperature-corrosion environment is particularly important. Scatter in the fatigue performance of materials exposed to a well-defined load environment is well documented. Because of the localized nature of fatigue damage and the difficulty of detecting damage in structural details, some degree of damage tolerance is usually built into the surrounding structure.

B. Conclusions Regarding Design

The significant design features developed in the previous sections are incorporated into most of the critical structural components of commercial and military aircraft flying today. Some of these features were available and incorporated into aircraft designed more than twenty years ago; these features have contributed significantly to the high level of safety of the structure. Newer aircraft designs follow the same philosophical concepts and procedures. Improvements to structural designs have resulted from more sophisticated computations and advances in materials and processes, rather than from major changes in design procedures.

In many cases, these design features are conservative.

Several incidents, such as the Aloha Airlines fuselage failure and failures due to heavy overstressing caused by turbulence, show that the airframe structure has more than an adequate margin of safety built into the design. Design and manufacturing features that make incremental improvements in strength and safety are usually incorporated into later production models of the same series of aircraft. These incorporations further refine and improve the initial quality of the materials and calculations for fatigue life. For example, Boeing engineers were able to estimate the possibility that an early design 737 fuselage would fail between 87,000 and 91,000 total cycles. The Aloha Airlines 737 had 89,680 cycles at the time of the accident.

Older aircraft are apparently designed as safely as modern aircraft. If one considers *only the design aspects*, there is no inherent decrease in safety as an aircraft ages. The difficulty lies in the fact that older aircraft have been subjected to more fatigue cycles, and there is a substantial increase in the probability that the older aircraft have sustained more fatigue damage and cracking. Thus, *the burden for continued safety shifts from the design to the maintenance and inspection process*. The structure's safety depends upon the detection and removal of defective structural components during routine maintenance. Structural design features prevent minor mishaps from turning into major disasters.

III. NONDESTRUCTIVE INSPECTION/MAINTENANCE INSPECTION

A series of nondestructive inspections must be performed on each critical part used in the aircraft to ensure that the component's life will comply with the anticipated design life. Nondestructive inspection (NDI) or nondestructive testing (NDT) can be defined as inspection processes that can determine the acceptability or fitness for purpose of a part without destruction of the part. In most cases the ability of the NDT to detect the presence

of a crack or crack-like defect must be assessed. The majority of NDI procedures fall within one of the following categories:

(1) visual or low magnification optical inspections. These may include specialized optical techniques used to enhance the image, or may include such procedures as liquid crystals, laser holography, interferometry, or computer enhancements;

(2) x-ray or penetrating radiation coupled with visual examination of the film;

(3) visible or fluorescent dye penetrant inspection in which surface defects reveal their presence during subsequent visual examination of the treated component;

(4) magnetic particle, visible, or fluorescent dye inspection. The part is magnetized and crack-like defects produce pseudo-magnetic poles, which attract magnetic or fluorescent coated particles. These reveal surface or slightly subsurface defects during visual inspection;

(5) ultrasonic inspection in which high frequency waves are reflected from surface or internal defects and produce an indication of the reflection on a cathode ray screen. This process requires information and knowledge of the types of reflections possible;

(6) eddy current inspection, in which a high frequency surface wave interaction with a surface or slightly subsurface defect results in perturbation of the electromagnetic field. This field perturbation is detected by a receiving coil and produces an indication. The process requires information about and knowledge of the types of electromagnetic perturbations that are possible in the material; and

(7) acoustic emission procedures, in which stressing the component containing the defect causes the defect to enlarge slightly. The defect then emits a stress wave pulse, which can be detected by pressure transducers placed on the part.

The primary purpose of these inspections is to ensure that defects of a size or criticality larger than a preselected design maximum will be detected by the NDT and that

the component containing these defects will be repaired or removed from service. The objective of the NDT is to achieve a level of competence and assurance consistent with the design requirements. Many levels of inspection take place during the initial manufacturing and subsequent service life. These inspections may include: (1) inspections performed on the raw materials prior to producing the finished part, which are usually conducted by the producer of the raw material; (2) inspection of the intermediate semifinished product, which is usually performed by the manufacturer or intermediate producers; (3) inspection of the product during various stages of its final manufacture, which is performed by subcontractors and the overall manufacturer; (4) acceptance inspection of the component after assembly into the aircraft structure, which is usually conducted by the producer and buyer; (5) periodic, routine inspection, which is usually carried out by the user or a subcontractor; (6) maintenance repair and A-, B-, C- or D-Level check and inspection, which is usually conducted by the user or subcontractor; and (7) special inspections to comply with Airworthiness Directives (AD) or service bulletins.

A. *Inspection Plans*

The primary manufacturer often produces a complete NDI plan covering the product cycle from design to service. Portions of the plan are provided to the user as a part of the purchase package. The complete inspection program usually covers specifications, research development, fabrication, NDI and field service manuals, field inspections, and teardown inspections.²⁶

Nondestructive inspection philosophy for military air-

²⁶ See D. Hagemaiier, *State-of-the-Art Inspection of Aircraft Structures* (1975) (unpublished manuscript) (available from Professor Packman at Southern Methodist University). The different phases of inspection are summarized briefly as follows:

(1) *Specifications*: Specifications contain requirements and procedures for qualifications, standardization, calibration, control of equipment, and personnel qualification requirements. These specifications are coordinated prior to release by

craft is different from that of commercial aircraft. In general, the military aircraft requirements are usually more demanding.²⁷ This usually is because the military structure is operated at higher stress levels and requires the early detection of considerably smaller flaws than commercial aircraft.

Most inspections in the field and in commercial acceptance areas are performed by certified inspectors in accordance with MIL-STD-41D or SNT-TC-1A. These inspection standards are limited to eddy current, magnetic particle, penetrant, radiography using radioactive isotopes, and x-ray. Vendor personnel are usually qualified by the vendor or an independent agent.²⁸

engineering, quality assurance, and other applicable departments. All NDI must be performed according to these specifications.

(2) *Research and Development*: During the development phase of the program, the capabilities of NDI to detect flaws in new materials are tested. Among the items considered in the effort to apply NDI correctly are the structural component to be inspected, critical area within each component, maximum allowable flaw size, critical flaw orientation, proper NDT method to be used, and the proper time in the manufacturing sequence to apply NDI.

(3) *Fabrication*: This is the process by which NDI is applied to either vendor-produced or in-house parts. Fabrication is a detailed written procedure that is performed by qualified personnel pursuant to guidelines set out in the applicable process specifications. This procedure includes an identification and recording system which requires that either the parts themselves or the records accompanying the parts be marked after inspection.

(4) *NDI Manual*: This manual sets forth NDI technique development, which must be applied whenever the NDI method (a) improves safe operation, (b) saves maintenance costs or manpower, or (c) increases operational effectiveness of the aircraft. The NDI Manual defines the area to be inspected, specific components to be inspected, location of the component, access to the area to be inspected, preparation of that area, methods and techniques of evaluating components for defects, equipment to be used, and reference standards for standardizing test sensitivity.

(5) *Field Service Manuals*: These are other manuals or technical orders, besides the NDI Manual, containing requirements for NDI. Although these manuals do not give NDI procedures, they may (a) give sufficient information for a mechanic to service a system, (b) call for inspection checks, (c) contain acceptance criteria for certain defects, or (d) give requirements to perform certain inspections at specified times. Some examples of these manuals include the Maintenance Instructions, Component Maintenance Manual, Structural Repair Instructions, Structural Repair Manual, Inspection Requirements, and Line and Dock Manual.

Id.

²⁷ *Id.* at 2.

²⁸ *See id.* at 4.

B. *Factors Influencing Defect Detectability*

The only way the NDT engineer and inspector can determine if an applicable instrumented NDT test method, such as eddy current or ultrasonics, will detect the designated minimum flaw is to have a reference standard with built-in defects of known dimensions. Calibrated notches or holes provide a minimum sensitivity level and indicate defect resolution. A reference standard based on geometric defects (holes, notches, or slots) is not necessarily identical to the tight crack-like defect or corrosion defect that often is found in aircraft having a large amount of flying hours. The proper selection of ultrasonic and eddy current testing permits the reliable detection of defects, provided that the calibration and sensitivity of the instrumentation are properly selected. If the sensitivity level is too low, harmful defects may not be detected. Conversely, if the level is too high, characteristics of the material that are either natural or not significant may be mistaken for defects.²⁹

The NDI procedure's ability to detect the presence of a crack in a part depends upon a number of factors. These factors can include:

(1) *The location of the crack* — surface cracks are more likely to be detected than cracks below the surface by simple techniques such as visual inspection, dye penetrant, magnetic particle, or eddy current. Interior defects are usually more difficult to detect.

(2) *The condition of the surface* — clean, well-prepared surfaces make the detection of cracks much easier. If the surface is roughened, dirty, oily, or covered with paint, the coupling of the NDT technique and the surface of the material containing the defect will be poor, greatly increasing the likelihood that a defect will be missed.

(3) *The adjoining structure geometry* — cracks usually develop in localized areas of high stress associated with changes in geometric shape. These changes in shape may

²⁹ *Id.* at 3.

decrease the ability of the NDT process to detect the crack because the adjacent geometry may shield the crack from the inspection. Typical examples include cracks that develop in the base of threads and cracks that develop under installed rivet or fastener holes.

(4) *The stress on the crack* — cracks under compressive loading are usually forced closed by externally applied loads. These cracks may be more difficult to find because it is the “openness” of a crack that aids many NDT procedures by producing a signal that indicates the crack’s presence. The fine crack-like lines formed by compressive stresses are not easily detected during visual inspections. Dye penetrant does not enter into tight cracks to produce visible indications. Similarly, ultrasonic stress waves do not reflect from tightly closed cracks.

(5) *The clarity and completeness of the inspection instructions* — before the production, service, or maintenance inspectors can inspect a part, NDI laboratory engineers must develop procedures to detect defects. Instrument sensitivity settings, specific procedures, and exemplar defects must be examined to ensure that the inspection process is adequate. These procedures must be agreed upon and be followed by the in-service inspectors.

(6) *The training, ability, and motivation of the inspector* — in the final analysis, the successful detection of defects depends upon the inspector’s ability and motivation. Most in-service inspections are difficult, demanding, tedious, and repetitious. To increase the effectiveness of inspections, more emphasis should be placed on adequately motivating and compensating inspectors.

The NDI’s capabilities for detecting flaws in materials should be defined during the developmental phase of the design. Quality Assurance NDT personnel in conjunction with field inspectors provide an effective transition from the laboratory to production and field inspections. NDT procedures called for in the inspection must be carried out correctly or defects that should be removed may be missed by the inspection and left in service.

C. *Failure to Detect a Defect During Maintenance*

A number of instances of inspectors failing to detect flaws during their inspections have been documented. For example, a flaw 4.525 inches long was detected in an area that had been inspected ultrasonically only 1,000 flight hours earlier.³⁰ It is unlikely that the flaw did not exist during the ultrasonic inspection. Furthermore, an inspection technique used at one installation may achieve a certain level of reliability, but may result in a different level of reliability at a different installation. For example, a technique that achieves a 90% reliability/95% confidence level (*i.e.*, a technique that is 90% reliable 95% of the time) at one installation may not achieve this same level at a different installation. In addition, an installation may not be able to maintain a specific level of accuracy for a certain inspection technique over an extended period of time. Thus, reliability of inspections that may vary significantly from installation to installation compounds the problem of potential low reliability of depot maintenance-level inspections.

D. *Airframe Inspection and Maintenance*

Standard aircraft inspection procedures detail the requirements for specific inspections at scheduled times. Although the manufacturer usually provides a commercial NDI manual, the Air Transport Association does not specify or require use of the manual. Many inspection programs, however, develop their own manuals. Field service manuals or technical orders often contain references to NDI. For example, when an aircraft experiences hard landings, excessive maneuvers, turbulence, lightning

³⁰ See Dornheim, *Boeing Methodology Faulted in Assessing Aircraft Corrosion*, Av. Wk. & SPACE TECH., July 18, 1988, at 91. Boeing estimated that "a crack could grow from a visually undetectable size to a 40-in. major failure within 3,000 flight cycles from the start of a crack" *Id.* Boeing, however, does not address the corrosion deterioration problem in its structural inspection document used to inspect aging aircraft because Boeing assumes that operators keep their aircraft corrosion-free. *Id.* at 91, 93.

strikes, or other unusual problems, a visual inspection must be performed, and perhaps an even more detailed NDT inspection may be required. Both structural repair instructions and manuals describe basic procedures for repair of cracked or damaged structures. The instructions and manuals normally contain acceptance criteria for cracks, corrosion, or inspections.

To avoid unwanted inspection costs, most parts are zoned with quality grades based on stress analysis or criticality of part function. Less critical parts require either less inspection or more time between inspections. Preparation of an NDI manual is the best way to approach the problems associated with on-condition maintenance and depot or base-level inspection.³¹

³¹ D. Hagemaijer, *supra* note 26, at 4-5. The NDI manual is an integral part of the testing and inspection plan:

An NDI manual is prepared for use during the operational phase of the aircraft. Techniques are developed to inspect critical areas of components for potential service damage (cracking, corrosion, or deformation). NDI technique development is required whenever one of the following criteria is met:

- (1) The NDI method improves safe operation or reliability of the system or sub-system.
- (2) A savings in maintenance costs or manpower will be realized by using NDI methods.
- (3) Operational effectiveness or life cycle costs will be favorably effective

The manual does not contain inspection level or frequency . . . acceptance/rejection limitations, or instructions for correcting defective conditions. The manual defines:

- (1) Area of aircraft to be inspected.
- (2) Specific component to be inspected.
- (3) Specific location on the component being inspected requiring special attention (defines defect location and orientation).
- (4) Access to area for inspection.
- (5) Preparation of area for inspection
- (6) Defines specific test methods or techniques required to evaluate component for particular defect or condition
- (7) Description of equipment required to perform the evaluation.
- (8) Description of reference standards (when required) for standardizing test sensitivity.

Each inspection will, when necessary, specify a verification (back-up) inspection procedure by another means to verify initial inspection results. The NDI manual evolves by a well-coordinated and planned sequence of steps.

An industrial task force assembled in 1977 attempted to develop standards for inspecting aging aircraft.³² The response at that time was to increase the frequency and scope of maintenance inspections. Normal procedures required that A-, B-, C-, and possibly D-Level checks on aircraft be performed within specified numbers of flying hours.³³

These inspection and maintenance programs are sensitive to the number of flying hours, but not necessarily sensitive to GAG cycles. The inspection programs and procedures, as well as time between inspections should include the difference in operations and loads used by different airlines. A program based on an average mixture of all types of operations should not be implemented. Short-haul flying, such as commuter flying, introduces the highest level of damage per hour of the aircraft's operation. The GAG cycle of twenty- to forty-minute operational flight produces more severe fatigue damage than that produced by a long overseas flight.

One airline has a standard 1,000-flight-hour inspection for corrosion, fatigue, and other types of cracking. This inspection occurs regardless of the type of operation the aircraft undertakes. Pan American Airlines requires a D-Level check on 747-100s every four to four and one-half years. In this procedure, essentially all critical moving parts are examined, and when necessary, are replaced with new or reconditioned parts. These procedures cost

³² See *Independent Research Urged*, *supra* note 22, at 112-13. Ray Valeika, vice president of maintenance and engineering with Pan American World Airways, was chairman of the task force. *Id.* at 112. Valeika advocates the position that aging aircraft can be operated safely under the system of inspection, surveillance, and repair developed by the industry and the federal government. *Id.* Valeika believes that airlines deserve more credit for the routine in-depth inspections that identify problems in early stages. *Id.*

³³ *Id.* at 113. The B-Level check is done every 1,000 flight hours for the purpose of detecting corrosion and fatigue damage in key areas such as behind the wing fairings and inside horizontal stabilizers. *Id.* A D-Level check is more extensive and detailed. On the average, the D-Level check is needed every four to four and one-half years. *Id.* A D-Level check may include reskinning the plane. *Id.* All movable parts are replaced with new or reconditioned parts. *Id.* Inspectors may require that additional work be completed. *Id.*

well over \$1,000,000 per inspection.³⁴

An airframe life extension program procedure used by Boeing-Wichita specifies the inspection and repair of preselected airframe structural areas. These procedures involve the inspection of specified portions of the airframe that are determined by test or calculation to be areas of high stress and potential service damage. Any damage found in these inspections must be repaired. The structure is assumed to receive a life extension because defective parts found during an inspection are supposed to be scrapped or repaired. The assumption a life extension is based upon is the belief that existing cracks will be discovered and repaired. Ideally, the inspected and repaired structure will be free of flaws and have a lifetime equal to the lifetime of the original flaw-free structure.

For example, in bolt hole inspections each hole is inspected for the presence of cracks. If a crack has a radial length of 0.03 inches or longer, the inspection process is likely to detect its presence. Figure 12 in Appendix A shows an eddy current inspection technique for detecting cracks under installed bushings. Figure 13 in Appendix A shows typical probabilities that eddy current inspection procedures will detect radial cracks in unfilled fastener holes.³⁵ If a crack is detected, removal by successive one-sixteenth inch reams until the crack indication disappears

³⁴ *Id.* The procedure is cost effective because replacement of the aircraft would cost approximately \$130 million. Pan American believes that the D-Level check produces an aircraft that is equivalent to a new one. *Id.*

³⁵ Hagemaiier, Bates & Steinberg, *On-Aircraft Eddy Current Subsurface Crack Inspection*, 46 MATERIALS EVALUATION 518, 518 (1988). Subsurface crack detection is accomplished using phase analysis eddy current (*i.e.*, current induced by an alternating magnetic field) instruments which produce impedance plane responses automatically on a CRT. *Id.* Figure 12 in Appendix A illustrates eddy current inspection for cracks under an installed bushing. To establish a reference standard, an eddy current probe is inserted into a bushed hole in metal of similar thickness and composition to that of the aircraft metal. *Id.* An operating frequency is selected that allows the eddy currents to penetrate the bushing to detect a notch precut in the metal to simulate a crack. *Id.* Using the generated CRT image as a reference standard, the bolt hole probe is inserted into bolt holes on the aircraft to detect existing cracks. *Id.* The depth of eddy current penetration is a function of operating frequency, material conductivity, and material magnetic permeability. *Id.* at 521-22. As illustrated in Figure 13 of Appendix A, the mini-

is allowed as long as the number of reams does not exceed a preselected number, for example, three. If an eddy current inspection determines that a hole is crack-free, a final oversized one-sixteenth inch diameter ream will remove any undetected cracks that are less than 0.03 inches in length.

E. *Conclusion: Airframe Inspection*

The technical problem with maintaining safety for aging airframes is knowing where to look for cracking, and developing reliable procedures for finding the initial small cracks. The present system of inspecting, maintaining, and repairing aircraft should be strengthened. Alternative methods should be considered for establishing limitations on the lives of components, inspection intervals, and some training standardizations for inspectors. The United States Navy uses full-scale fatigue tests to determine useful safe life. If the Navy anticipates additional use, it commissions new tests that go beyond the initial design life. It seems reasonable to require further fatigue-substantiation tests for commercial aircraft that will determine the life extension of critical structures. The evaluation of in-service hold times, environmental effects, and corrosion, which affect the validity of life extension estimates, make the application of these tests to real structures difficult.

Increased federal and industrial support needs to be made available for both applied and directed research and development (R&D) programs dealing with the technical aspects of in-service crack detection. Programs designed to examine the human factors and management processes involved in inspection and repair should also be emphasized. The FAA needs to involve more inspectors who are trained to make major maintenance checks. The FAA should also evaluate projects designed to produce more

imum detectable crack length is determined by the thickness of the aluminum structure under examination. *Id.* at 522.

reliable NDT procedures. In addition, the FAA needs to have more specialists who can conduct field visits to determine the human factors associated with problems of defect detection, particularly routine inspections. Moreover, nationwide attention should be focused on the need to educate and train more engineers, technicians, and maintenance personnel in the reliability of NDT and more advanced techniques. NDT, particularly in field inspection, is presently a lower level job that is mundane, boring, and one out of which people would like to be promoted.³⁶

An R&D program aimed at gathering and distributing inspection procedures, training procedures, and other factors needed to improve the reliability of NDT should be established. In the late 1970s, the United States Air Force and the Advanced Research Projects Agency of the Department of Defense established such a basic research program, but it is no longer in effect. The primary aim of the project was directed toward advanced NDT research that could be used in advanced military aircraft. The Electric Power Research Institute (EPRI) has also supported an NDT research and development program and has established an NDT center. The EPRI center, however, is dedicated to power plant and nuclear inspection procedures that apply to aircraft only marginally.

Industry does not appear to object to more stringent requirements if the requirements address and resolve safety problems. Hasty and ill-conceived requirements

³⁶ *Independent Research Urged*, *supra* note 22, at 112. Some experts maintain that research and development efforts in the inspection and maintenance of aircraft should be premised upon the preference for an engineering approach over a political solution. *Id.* An independent research effort would more likely appreciate and account for the different ways airlines operate aircraft and avoid problems inherent in establishing a single standard for all airlines. *Id.* Proponents contend that a systematic approach to safety is preferable to the reactionary method employed by the FAA. *Id.* at 113. Private industry or government agencies other than the FAA, such as NASA, are proposed as better suited to undertake research and development efforts because of their immunity from the rule-making process. *Id.*

would therefore significantly increase inspection costs, create confusion, and not truly increase safety.

IV. DURABILITY AND RELIABILITY

A. *Durability*

Aircraft structural systems must meet stringent strength demands, be resistant to cracking, and allow for sufficient structural back-up structure to ensure that fail-safe requirements are met. At the same time, the entire flying system must meet economic, range, and fuel cost requirements. It must also meet requirements for durability and structural integrity during its operational lifetime. The durability requirements for an aircraft are designed to estimate the inspection and repairs that may need to be performed on the structure as it ages.

In a durability analysis, the total economic costs of inspecting and repairing the aircraft during its design lifetime are important. For example, if an aircraft is to be flown for 100,000 hours of operation, a durable design is one that does not require major reworking or inspection of primary structural components, particularly fastener or rivet holes, until after the 100,000 hours. Thus, durability could be considered analytically to be a measure of the total number of repairs and the cost of those repairs that must be made on the aircraft structure within the assumed life of the aircraft.³⁷

³⁷ Yang, *Statistical Estimation of Economic Life for Aircraft Structures*, 17 J. AIRCRAFT 528 (1980). Statistical methods for predicting the economic life of critical components of aircraft structures have been developed. *Id.* at 534. The formulas account for various conditions, such as any type of initial fatigue quality, crack growth damage accumulation, loading spectra, material/structural properties, usage change, and inspection and repair maintenance. *Id.* at 528. The formulations allow for the determination of economic life using either of the following two criteria: (1) a rapid increase in the number of crack damages exceeding the economic repair crack size, or (2) a rapid increase in the maintenance cost. *Id.* It has been demonstrated numerically that the percentage of cracks exceeding the economic repair crack size increases rapidly after a certain time, thus determining the component's economic life. *Id.* While the inspection and repair maintenance procedure has a significant impact on the safety and reliability of aircraft structures, its effect on the economic life of a component is limited. *Id.*

A durable structure is defined as one for which the number of flight hours until repairs are required is greater than one design lifetime. This is illustrated in Figure 14 of Appendix A, in which a structural component's crack size is plotted against the component's lifetime. The initial flaw size is not a single value, as used in the slow crack growth design, but rather includes all of the defects assumed to be present in the structure at the inception of service. The scatter in flaw initiation sizes represents variability of materials as well as manufacturing variability throughout the structure. Some of these defects may grow faster than others, resulting in the spreading out of the scatter band that represents the distribution of defects at any time in the aircraft's history.

A durability analysis focuses on the number of cracks and the number of hours of operation after which the cracks will grow to exceed a given flaw size. This size is usually 0.03 inches, which would be the detectable flaw size growing from a fastener hole. As the initial defect distribution grows and disperses, the total distribution of defects in the component approaches the inspectable flaw size of 0.03 inches. If a large percentage of the fastener holes do not develop cracks larger than 0.03 inches until after one design life (100,000 hours), the design is considered to be a durable structure.³⁸

Attaining durability is a difficult task. Durability analysis is a complex design process and is difficult to verify prior to use. The actual component may be subject to un-

³⁸ See Rudd & Gray, *Qualification of Fastener Hole Quality*, 5 J. AIRCRAFT 143 (1978). Rudd and Gray give one example of a durability analysis, the Equivalent Initial Quality Method. This test assumes that defects or initiation of cracks resulting from imperfections in either the manufacturing of the structural component or in the material itself are present at the onset of the aircraft's life. The article gives a description of the test and applies it to fastener holes, which are the most prevalent source of cracking in aircraft structures. Specifically, it describes how a statistical distribution may be used to determine fastener hole quality resulting from certain manufacturing processes. The article then applies the test to two aircraft, the F-4 C/D and the A-7D. The conclusion is that this test may be used in the future to determine the acceptability of certain manufacturing procedures, required inspection intervals, and maintenance and modification schedules. *Id.*

expected and unidentified operational exposures that the analysis verification does not account for. The durability fatigue analysis process is not as well developed as the durability analysis for a static (one load application) case.

Typical approaches that compensate for fatigue damage variability in aircraft structures usually range from selection of arbitrary scatter factors such as two or four, to the use of computational cycle-by-cycle life analysis, which tracks each aircraft individually. The problem is one of characterizing the scatter in fatigue behavior, in expected load history, in the initial material properties, and in manufacturing processes by fitting a statistical distribution to the observed data. The likely or average performance and its associated safety factor, which will assure an adequate margin of safety for the structure, are then determined. Success is measured by the accuracy with which the selected distribution and parameters replicate the observed scatter of the actual fleet's fatigue performance.

B. *Reliability*

The concept of structural reliability as applied to older aircraft refers to the ability to predict the location and number of fatigue cycles that apply to each aircraft to produce a detectable flaw. The appearance of unexpected fatigue damage usually occurs first as a crack and is followed by subsequent growth, which causes the failure. If this fatigue damage remains hidden and is not removed, the function and safety of operational structures may be compromised and serious economic consequences may result. Fatigue damage can appear early in the structure's life. If it is shown that the damage results from a fleet-wide problem rather than an isolated incident, then obviously the component is not reliable as designed. The structure must be replaced or repaired, and the aircraft's use must be restricted.

As the aircraft ages, any a priori prediction of the structure's reliability becomes more difficult. The total structure undergoes a large number of cycles as well as

exposure to the environment. As the number of potential sites for fatigue and corrosion damage increases with age, predicting the location of the damage becomes more difficult. In addition, the varied usage of each individual aircraft increases the scatter in the distribution of flaw growth, which also decreases the accuracy of the prediction.³⁹

Remedial steps for fatigue repair are not simple because of the cumulative effects of the fatigue cycling. Repair of a fatigue defect in one location does not preclude the presence of another defect in an immediately adjacent location. Fatigue damage initiation and propagation continue over the total life of the system.

The major problem associated with a statistical approach to the fatigue damage-reliability problem is identifying the tail of the distribution of the damage curves. The central tendency properties, mean, median, standard deviation, and characteristic life can be estimated by examining a few experimental test values. The reliability, and associated safety, of a single aircraft is represented by the time-to-first failure rather than the time-to-average failure. Current FAA Advisory Directive procedures are based on the discovery and recognition of the first failure. When the FAA gathers sufficient information indicating that the failure is a reliability problem instead of a series of isolated incidents, remedial action for the total fleet is taken.

The current United States Air Force philosophy, which is based on the concept of initial flaws or preflaws in every critical component, assumes that each component is designed so that it will fail first. This philosophy is extremely conservative and often excessively expensive.

³⁹ See Hart-Smith, *supra* note 18 discussing examples of these problems of lack of structural reliability. One example of the scope of the problem is presented by the McDonnell Douglas C-133 Cargomaster transport aircraft. The cracks in this aircraft that caused an explosive decompression were not caused by structural stress concentration, but instead by pressure pulses from the tips of the propeller blades. Unfortunately, the result was the same: a long thin crack developed and caused the decompression. *Id.*

The question of whether it truly increases the airframe's reliability remains unanswered.

V. SUMMARY AND CONCLUSIONS

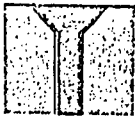
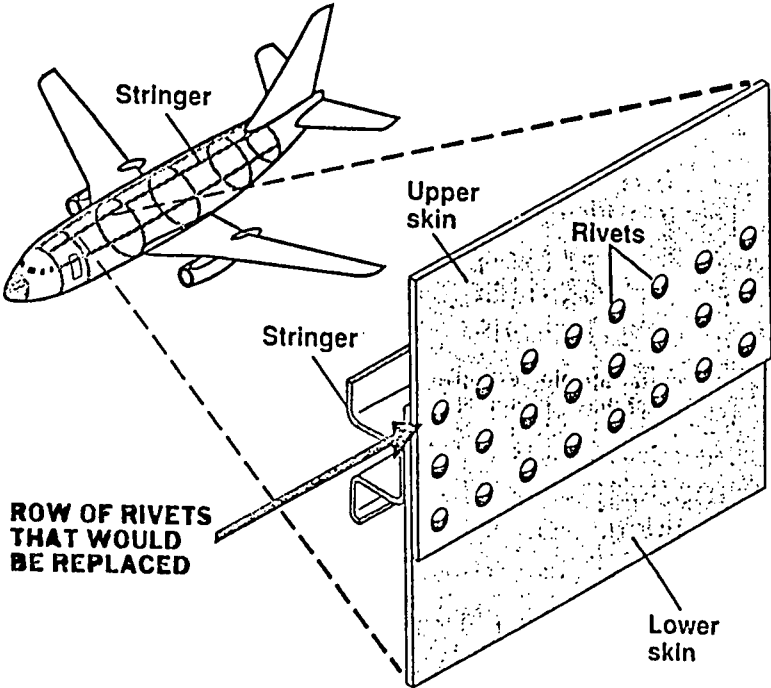
Aircraft structural systems are designed to be safe for a finite fatigue life, with the implied knowledge that fatigue cracking will occur over the aircraft's life. The continued safety of any aging aircraft can be maintained and ensured only by continued inspection. The safety of the aircraft thus depends upon the thoroughness, sensitivity, and success of the inspection procedure. The material properties of crack growth rate, material sensitivity to crack growth parameters, environmental effects such as corrosion, and residual strength in the presence of known crack-like flaws are important parameters in the formulation of any surveillance scheme. The interval between the time a crack is first formed and the time at which the crack has grown to a size detectable by overhaul inspection or inspections prior to the growth of the crack to a critical size provides the only opportunity to locate and remove the damage. The success of any of these procedures depends on the sensitivity and reliability of the inspection and the inspectors.⁴⁰ This places a high degree of responsibility on the inspection program, managers, and line inspectors. To ensure the safety of an aging aircraft fleet will require more resources and management concern directed toward this technical area.

⁴⁰ See *Independent Research Urged*, *supra* note 22, for further discussion of problems with the current system of inspection and maintenance.

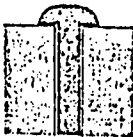
APPENDIX A

The Trouble-Prone Rivets on 737's

A row of rivets would be replaced on early-model Boeing jets near each of 10 stringers, the metal strips running front to back that form part of the shell of the plane. Sections of the aluminum skin are attached to the stringers and to each other in panels that overlap like shingles on a roof. The top row of rivets, targeted for replacement, helps join the upper and lower skins.



OLD RIVET
Countersunk type tends to cause cracks in the skin where the rivet angle changes.



NEW RIVET
Buttonhead type protrudes from skin, eliminating changes in the rivet angle.

Source: The Boeing Company

FIGURE 1 FAA REPAIR REQUIREMENTS ON OLDER BOEING 737 (New York Times Oct. 1988)

Figure 2
Maximum Age of Commercial Aircraft
Still in Operation

<u>TYPE OF AIRCRAFT</u>	<u>YEAR IN SER- VICE</u>	<u>MAX AGE</u>	<u>NUMBER OPERATING AS OF 1986</u>
Boeing 707	1958	30	284
Boeing 727	1963	25	1678
Boeing 737	1965	23	1135
Boeing 747	1970	18	597
Mc/Douglas DC-8	1959	29	244
Mc/Douglas DC-9	1965	23	921
Mc/Douglas MD 80	1980	8	328
Mc/Douglas DC-10	1971	17	356
Lockheed Electra	1959	29	76
Lockheed L1011	1972	16	217
Airbus A 300	1974	14	247
Airbus A 310	1983	5	79
Airbus A 320	1988	—	—
Concorde	1975	13	—
Martin 404	1951	37	—
Shorts 330	1976	12	—
Metro (Merlin)	1971	17	—
Fokker F27	1958	30	434
Fokker F28	1969	19	189

Jane's Aerospace Facts, Jane's 1986

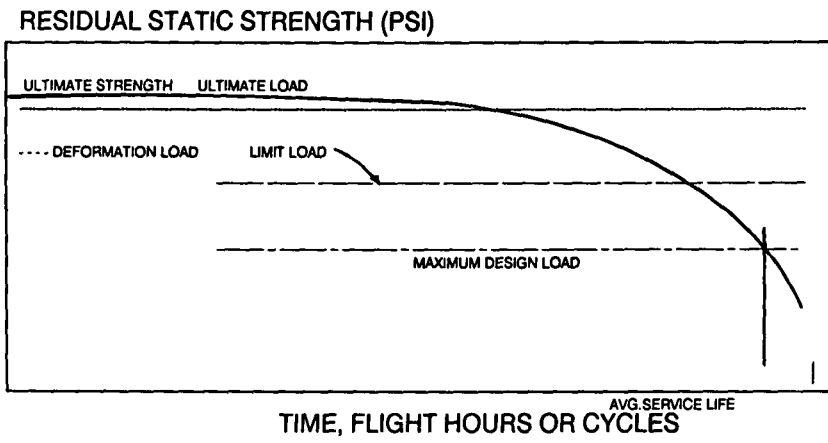


FIGURE 3.. FATIGUE LIFE OF AIRFRAME.
Residual Static strength decreases as a function of time or cycles on the aircraft.

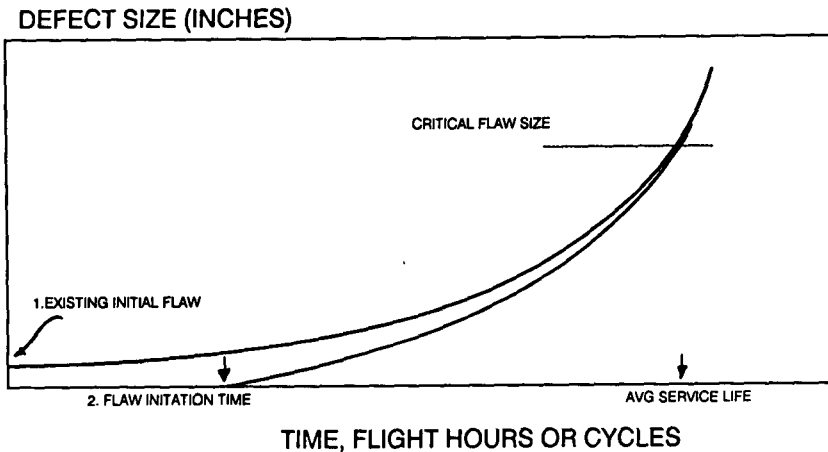


FIGURE 4..DEFECT SIZE VS FLIGHT HOURS.
Growth of the flaw that reduces the residual static strength.

PROBABILITY OF FAILURE

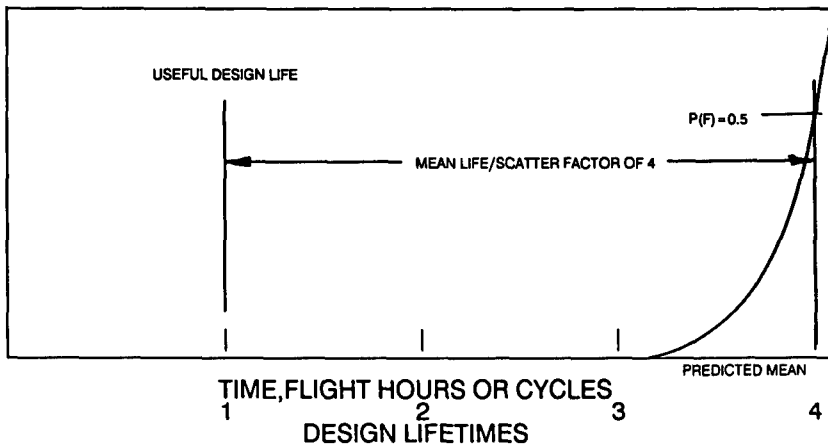


FIGURE 5.. SAFE LIFE FATIGUE DESIGN

Based on unflawed data and Miner's cumulative damage calculation for mean fatigue life.

FLAW SIZE (INCHES)

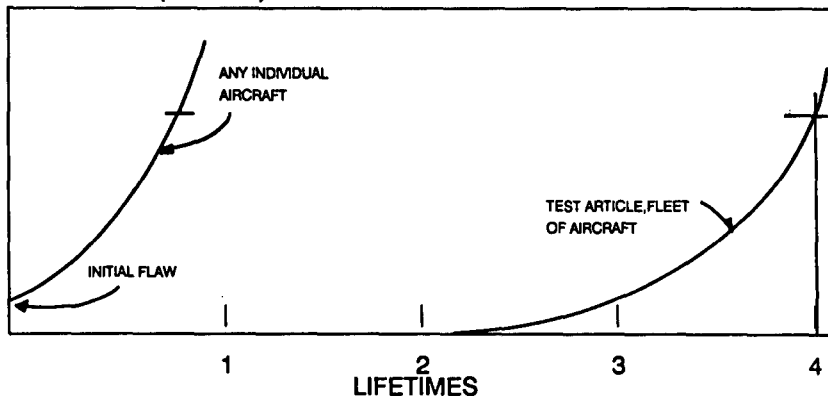


FIGURE 6..SHORTCOMINGS OF SAFE LIFE DESIGN

Any individual aircraft could exhibit poor correlation between test and service life experience

Figure 7
Examples of Early Service Life Failures
Shortcomings of Safe Life Design

<u>Structural Failure</u>	<u>Test Article Hours to Failure</u>	<u>Service Article Hours to Failure</u>
F-4 Wing failure	11,800	1200
F-111 Wing pivot	40,000	100
F-5 Wing	16,000	1000

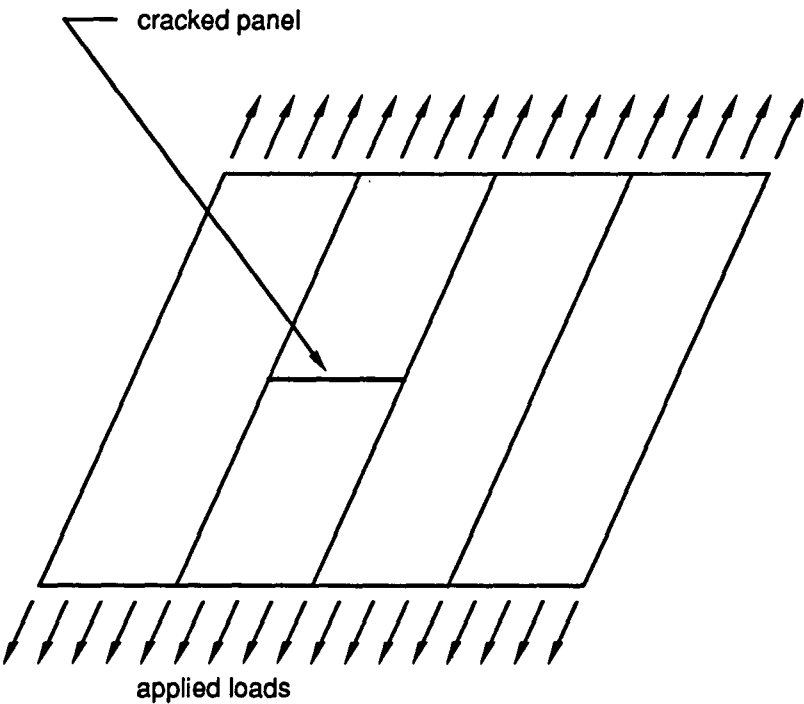


Figure 8. FAIL SAFE DESIGN - The structure is designed to contain a single member failure without loss of the aircraft

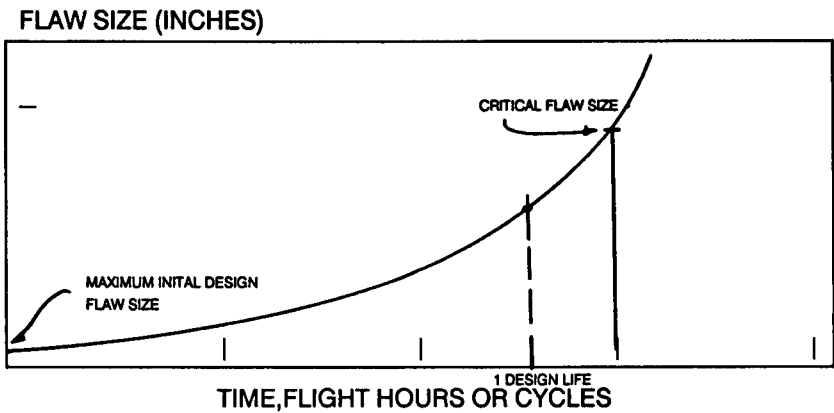
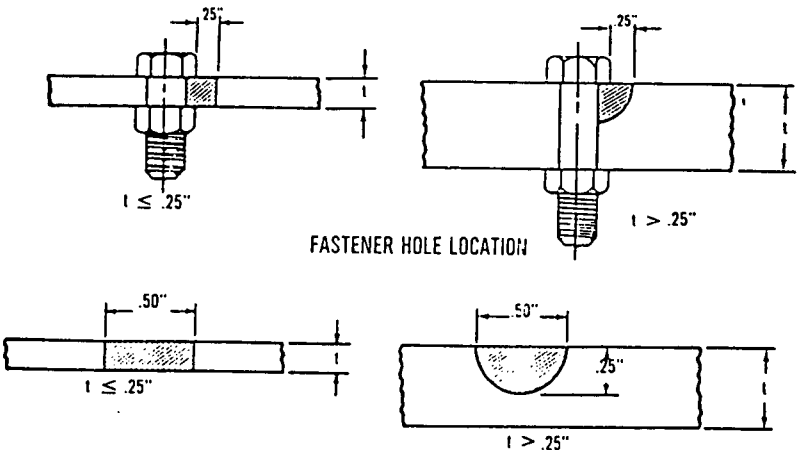


FIGURE 9.. SAFE LIFE, SLOW CRACK GROWTH DESIGN
Structure is designed and inspected so that maximum expected initial damage will not grow to critical size in 1 design life.

INITIAL PRIMARY DAMAGE FOR DEPOT OR BASE LEVEL
INSPECTABLE STRUCTURE WITHOUT COMPONENT REMOVAL



INITIAL PRIMARY DAMAGE FOR NONINSPECTABLE AND DEPOT
OR BASE LEVEL INSPECTABLE STRUCTURE WITH COMPONENT
REMOVAL (SLOW CRACK GROWTH STRUCTURE)

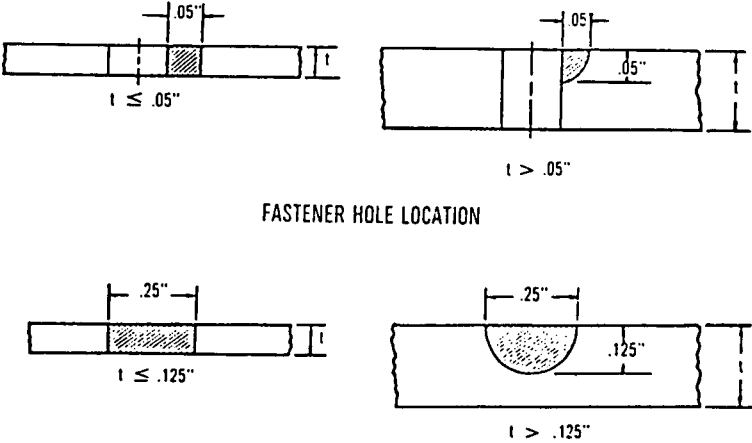


FIGURE 10 SIZE OF INITIAL DAMAGE USED IN CALCULATION FOR
SLOW CRACK GROWTH ANALYSIS. (MIL-A-83444; MIL A-
8866, see also Ref 14,15)

(SURFACE FLAWS)

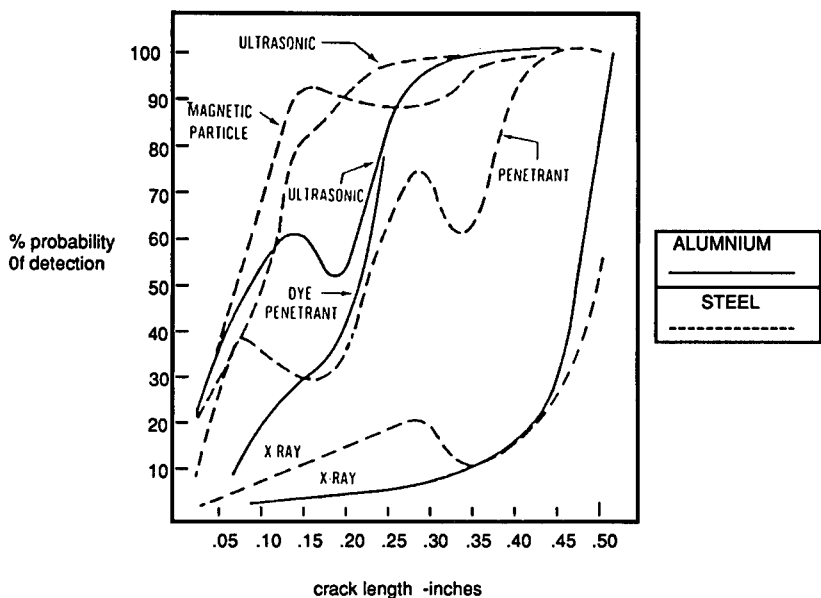


Figure 11. Reliability of Manufacturing NDI

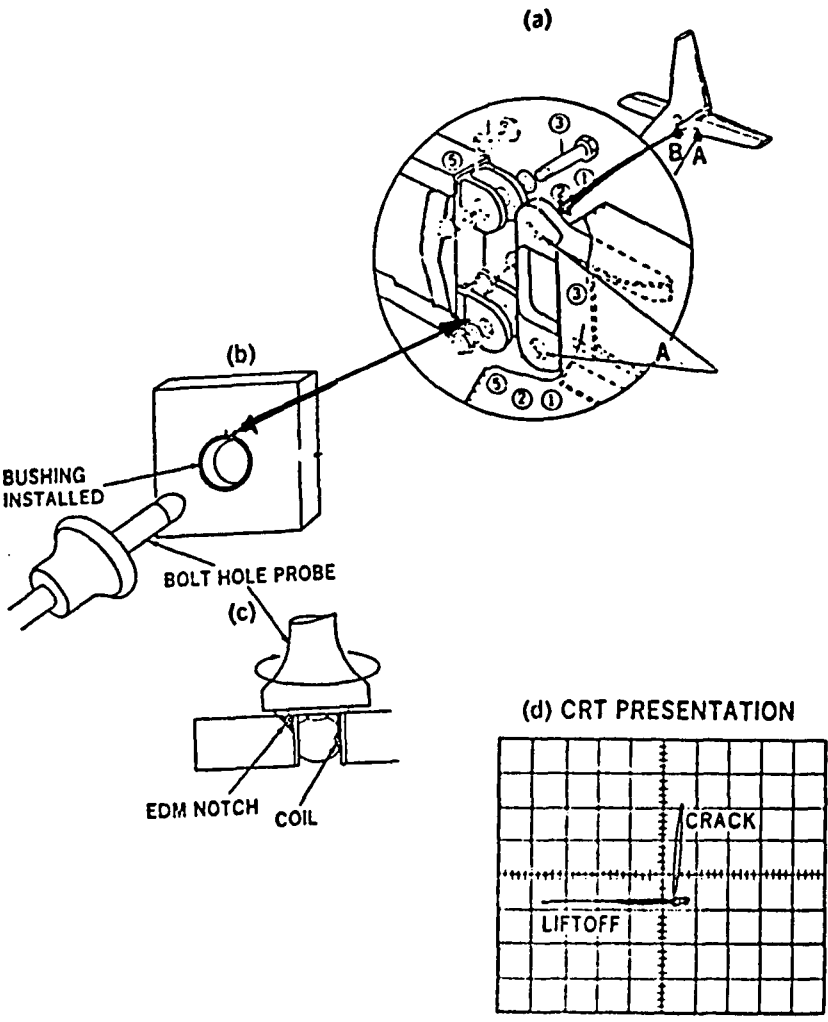


FIGURE 12 EDDY CURRENT INSPECTION FOR CRACKS UNDER INSTALLED BUSHING (A) TYPICAL LOCATION (B) BOLT HOLE (C) PROBE ROTATION (D) TYPICAL INDICATIONS

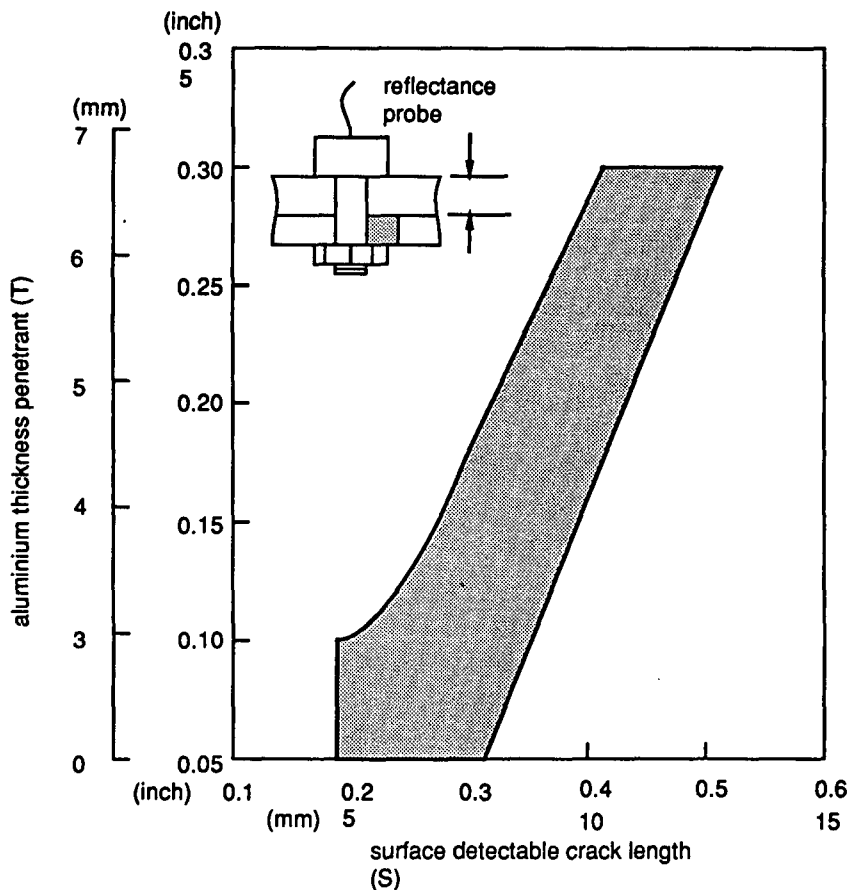


Figure 13. Detectable Crack Length for Eddy Current Inspection of Aluminium Airframes (Ref. Hagamier, etal.)

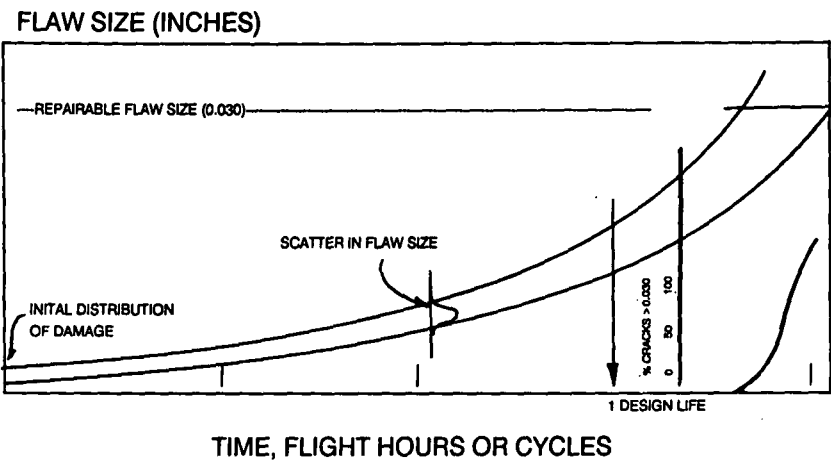


FIGURE 14.. DURABILITY ANALYSIS
Measure of the number of defects that will grow to require repair (0.030) or larger in one lifetime.

APPENDIX B

ADDITIONAL SOURCES OF INFORMATION

Abramson, *Defining the Design Defect in Aircraft Products Liability Cases*, 45 J. AIR L. & COM. 167 (1979).

Aging Aircraft Task Force Recommends Modify and Replace, Not Just Inspect, AIR SAFETY L. & TECH., Mar. 7, 1989, at 1.

Comment, *Crashworthiness Claims in Aviation Accidents*, 53 J. AIR L. & COM. 291 (1988).

Comment, *Liability of Independent Servicers and Repairers of Aircraft*, 54 J. AIR L. & COM. 181 (1988).

Comment, *Products Liability: Component Part Manufacturer's Liability for Design and Warning Defects*, 54 J. AIR L. & COM. 215 (1988).

Dombroff, *Certification and Inspection: An Overview of Government Liability*, 47 J. AIR L. & COM. 229 (1982).

Donnelly, *Aircraft Crashworthiness — Plaintiff's Viewpoint*, 42 J. AIR L. & COM. 57 (1976).

Eastern 727 Suffers Skin Tear: FAA Inspection Rules Tightened, AIR SAFETY L. & TECH., Jan. 10, 1989, at 3.

FAA Takes New Steps on Aging B-737s: Action Considered on B-727 and B-747, AIR SAFETY L. & TECH., Nov. 1, 1988, at 1.

Fleming, *The Duty of the Manufacturer to Recall Aircraft*, 45 J. AIR L. & COM. 581 (1979).

Galerstein, *A Review of Crashworthiness*, 45 J. AIR L. & COM. 187 (1979).

Harrington, *The Chicago DC-10 Crash and Passenger Confidence: Are There Any Implications for Airline Deregulation?*, AIR & SPACE L., Winter 1984, at 3.

Jorgensen, *The Defense of Aviation Mechanics and Repair Facilities from Enforcement Actions of the Federal Aviation Administration*, 54 J. AIR L. & COM. 349 (1988).

Kean, *Product Liability for Aircraft: A Conflict of Obligations*, J. BUS. L., Mar. 1986, at 242.

Knight, *Accident Investigation Procedures as Viewed by a Technician*, 9 AIR L. 30 (1984).

Krause & Cook, *The Liability of the United States for Negligent Inspection — 1983*, 48 J. AIR L. & COM. 725 (1983).

New Ads Issued for Aging DC-9 Aircraft Said to be Result of an Industry Effort, AIR SAFETY L. & TECH., Nov. 29, 1988, at 1.

Peck & Hoch, *Engineers' Liability: State of the Art Considerations in Defining Standard of Care*, TRIAL, Feb. 1987, at 42.

Peck & Hock, *Liability of Engineers for Structural Design Errors: State of the Art Considerations in Defining the Standard of Care*, 30 VILL. L. REV. 403 (1985).

Petroski, *Structural Failure and Metal Fatigue: Flaws Can Improve Design*, 5 J. PROD. LIAB. 401 (1982).

Ripling, Corsley & O'Donnell, *How Fracture Mechanics is Used to Determine Why Metal Parts Break*, 34 FED'N INS. COUNS. Q. 177 (1984).

Rosenfield, *Aviation Deregulation and Safety*, 9 AIR L. 105 (1984).

Saba, *Aircraft Crashworthiness in the United States: Some Legal and Technical Parameters*, 48 J. AIR L. & COM. 287 (1983).

Smith, *Defective Military Aircraft and the Government Contractor Defense: The Constitutional Difficulties that Arise Even After Boyle v. United Technologies Corp.*, 54 J. AIR L. & COM. 439 (1988).

Sterns, *IFR — The Liability of the Airframe and Component Manufacturer*, 44 J. AIR L. & COM. 367 (1978).

Stevens, *Government Liability Under the Federal Tort Claims Act for Negligent Inspection and Certification of Aircraft*, 8 AIR L. 230 (1983).

Tetelman & Redmon, *The Role of Metallurgy in Aircraft Accident Investigation and Litigation*, 44 J. AIR L. & COM. 401 (1978).

Turner & Satin, *The Government Contractor Defense: When are Manufacturers of Military Equipment Shielded from Liability for Design Defects*, 52 J. AIR L. & COM. 397 (1987).

Walkowiak, *Product Liability Litigation and the Concept of Defective Goods: "Reasonableness" Revisited?*, 44 J. AIR L. & COM. 705 (1979).

Whitehead, *Some Comments on Aircraft Crashworthiness*, 42 J. AIR L. & COM. 73 (1976).

Wilson, *Airworthiness Directives: Recovering the Cost of Compliance*, 49 J. AIR L. & COM. 1 (1983).

Comments

