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Forty Years of Structural Durability and Damage Tolerance at Boeing Commercial Airplanes

Steven A. Chisholm, Antonio C. Rufin, Brandon D. Chapman and Quentin J. Benson

Abstract – **As The Boeing Company enters its second century in commercial aviation, this paper reviews the progress made over the past forty years in the development and application of durability and damage tolerance methods across its commercial airplane product line, and ventures forward to look at future challenges and opportunities. The company's pioneering efforts, which saw significant advances in the 1970's and 1980's with the development of comprehensive internal technology standards have been evolving, driven by shifts in the regulatory and competitive environments and significant technological developments. Successes are evident in terms of significant safety improvements and considerable reductions in service actions on airplanes designed since that time. Managing an aging fleet and the adoption of new regulations affecting maintenance planning have been, and remain some of the greatest challenges. New materials (composite and metallic) and assembly methods, and the ever-constant quest for productivity gains in the factory and in design and analysis processes are all further shaping the way durability and damage tolerance are being assessed and implemented.**

I. Introduction

In his 1993 International Committee on Aeronautical Fatigue and Structural Integrity (ICAF) Plantema Memorial Award lecture [1], Dr. Ulf Goranson described damage tolerance (and indirectly, durability) as the result of an evolutionary process built on decades of experience and lessons learned, made possible only through diligent attention to detail design, manufacturing, and maintenance and inspection procedures. The lecture touched on basic aspects of the damage tolerance philosophy in its application at The Boeing Company, and almost presciently, on the many challenges that the industry was beginning to face, particularly as it addressed widespread fatigue damage (WFD) and the application of damage tolerance to major structural repairs and alterations. Both of these topics have since become the subject of major regulatory actions and significant industry focus.

During these years, technology has evolved. Although the basic analysis principles employed today are not substantively different from the methods described in Ref. [1 – 5], the tools and processes used have been refined over these past 20 years, extended to broader applications, seeking more efficient use of engineering resources, greater accuracy, and incorporating service experience acquired from an ever-expanding fleet. In that time, durability and damage tolerance (DaDT), moreover, have had to keep pace with major developments in materials and manufacturing processes. Composites, for example, saw their first primary structure application at Boeing in the early 1990's on the 777 empennage. Today, they are used on the new Boeing 787 and 777X in areas of the airframe that previously were the exclusive domain of metals.

The most important outcome from the early developments and their evolution in the more recent past is the unprecedented, steady advance in commercial aviation safety that has been achieved since the advent of commercial jet transports in the 1950's, despite a significant growth in the size of the worldwide fleet (Fig. 1). Although the number of hull loss accidents associated with structural issues in these statistics comprises only a small percentage of the totals depicted in Fig. 1, design for DaDT has certainly been a fundamental contributor to this remarkable safety record. Progress in commercial aviation safety has always been entwined with the evolution of design requirements that are mandated by the regulatory agencies (Fig. 2). The introduction of fail-safe design on the Boeing 707, design and testing for durability starting with the model 727, and incorporation of DaDT methods and more active corrosion prevention and control measures starting with the 757 and 767 airplanes, have all played a role in bringing about a significantly safer and more durable and therefore economically viable fleet.

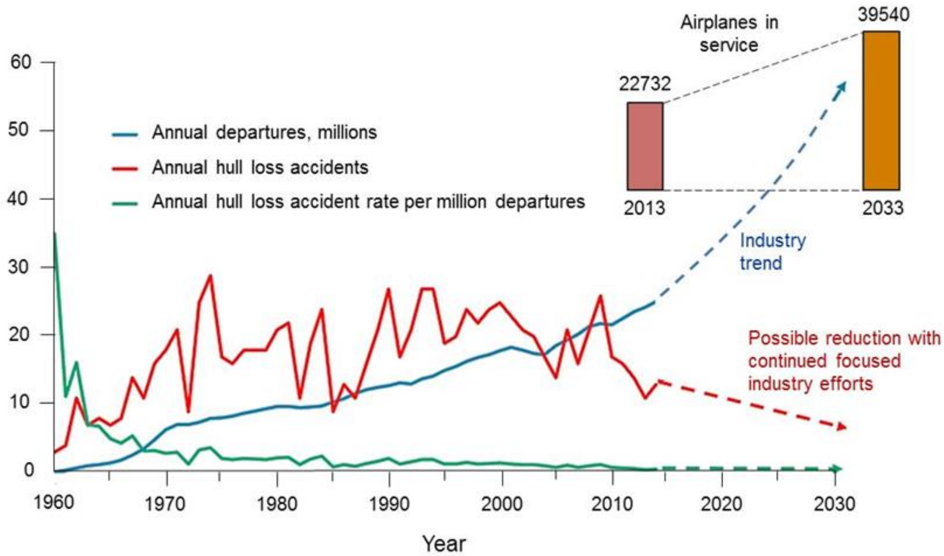


Fig. 1 – Annual hull loss accident totals and rates for the worldwide commercial jet fleet [6].

In another major accomplishment, and supporting these developments, Boeing has created industry-leading technology methods and standards that provide a Boeing commercial airplane product-wide uniform approach to design and analysis for fatigue, damage tolerance and corrosion prevention and control (Fig. 3). The standards capture lessons learned and are periodically updated to incorporate new methods and to gather in further testing and fleet experience, which can then be adopted in new designs or as design improvements. Furthermore, since their inception, these standards have been built for ease of use both in terms of prerequisite analysis skills and tools, and reliance on a manageable number of variables. This paper reviews the origins and evolution of durability and damage tolerance of commercial airplane structures, focusing on basic principles and their current application at Boeing, and their validation. Recent developments on composite damage tolerance are discussed next. The paper concludes with a discussion of new technologies and challenges, and how they are being addressed.

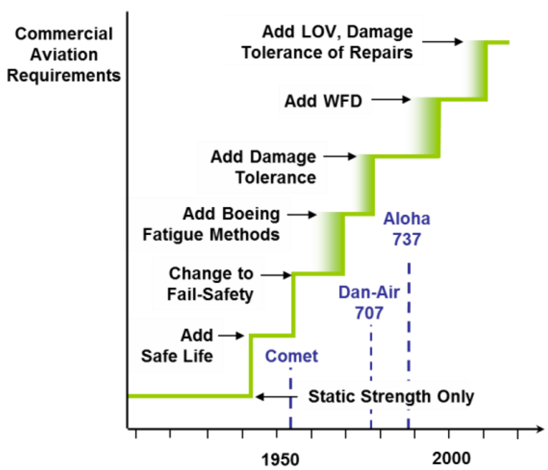


Fig. 2 – Evolution of design requirements for commercial transports, and some of the events that shaped them.

II. The Boeing Durability Method

Initially developed in the early 1970's and first implemented with the 757 and 767 models, the Boeing durability analysis standard, commonly referred to as "Book 2", along with its design companion, "Book 1", were first introduced with the overall objective of assuring a competitive economic life for the structure. This goal was specifically defined as an absence of significant fatigue cracking in the first 20 years of service (defined as a probability of cracking of less than 1 percent with 95 percent confidence) and at least 30 years of service before fatigue related maintenance begins to measurably escalate (defined as a nominal probability of cracking of 5 percent with 95 percent confidence), meaning that the airframe truly remains economically viable for a minimum of 30 years. This objective, and the means to attain it, has since been validated by 737 Next Generation (NG) / 757 / 767 / 777 / 787 full-scale fatigue test performance and by direct comparisons of service data with earlier airplane experience. In more recent times, the method has been effectively extended from its original economic life focus to also support damage tolerance threshold calculations and WFD analyses.

A. Durability method fundamentals

For metallic structure, the Book 2 method at its core balances a structural capability – characterized by the Detail Fatigue Rating (DFR) – against a set of requirements (stresses, minimum design service objective) to arrive at a Fatigue Margin analogous to a static strength Margin of Safety, in that it is likewise defined by a ratio of stresses or a ratio of parameters in units of stress (Fig. 4). Statistical factors are used to attain prescribed reliability levels for each component [1 – 3].

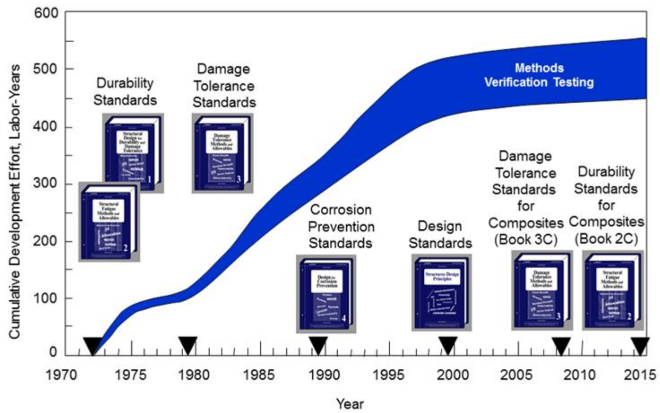


Fig. 3 – Evolution timeline and resources required for Boeing structural DaDT technology standards.

DFR is a component-level fatigue quality index expressed in units of stress. With a set of standardized stress-life (S-N) curves, it can be used to determine 95 percent reliability, 95 percent confidence fatigue lives at any given stress or allowable fatigue stresses for a particular target reliable life. DFR values can be applied to a wide range of structural features, from the simplest of stress concentrations to complex joints. For joints, the method can capture load transfer effects, material, surface finish, hole fill, hole treatments such as coldworking, and clamp-up. DFR values are determined based on combinations of empirical data and analysis. They can be drawn from test as well as service data, as Figure 5 shows. Service data can be especially useful in developing or improving fleet maintenance planning, or to handle unexpected service findings; for example, as a basis for fatigue-related service bulletin thresholds. Book 2 provides a comprehensive summary of DFR values as well as the procedures and constituent factors used to determine them. Complementing Book 2, Book 1 provides specific design and analysis guidance collected from service and full-scale fatigue test data to aid designers and analysts avoid known issues and adopt best practices.

The requirements aspect (lower branch in the Fig. 4 diagram) of the analysis begins with the definition of design service objectives based on a range of anticipated airplane operational usage rates. Loads are structured as standard mission profile segments, covering all applicable conditions encountered in a typical flight.

The method can accommodate conditions represented by step, constant-amplitude (discrete-cycle), and spectrum loading. The latter rely on standard exceedance curves substantiated by flight test or other operational data. Variable-amplitude profiles are rendered as equivalent once-per-flight constant-amplitude "GAG" (ground-air-ground) loads or stresses, the relationship between flights and the GAG cycle being handled by a proportionality constant termed "GAG Damage Ratio." The GAG Damage Ratio is determined by a complex procedure combining cycle counting and Palmgren-Miner's Rule over large numbers of flights to arrive at a cumulative fatigue damage that is then compared to the GAG cycle damage. This enables transforming design service objectives, which are expressed in numbers of flights into equivalent constant-amplitude GAG cycles that can then be used in conjunction with constant-amplitude S-N curves to calculate stress-based fatigue margins for any given structural component. The entire process can be accomplished seamlessly using specialized software.

B. A legacy of durability

Recent indications from the worldwide commercial airplane fleet are that average airplane ages may be peaking, but much of the current pool remains quite advanced in age. The size and longevity of the Boeing active commercial fleet, as illustrated by Table I, is a testament to a quality achieved only as a result of decades of development and experience. These statistics are also a reminder of the continuing challenges associated with aging structures and the value of design for durability.

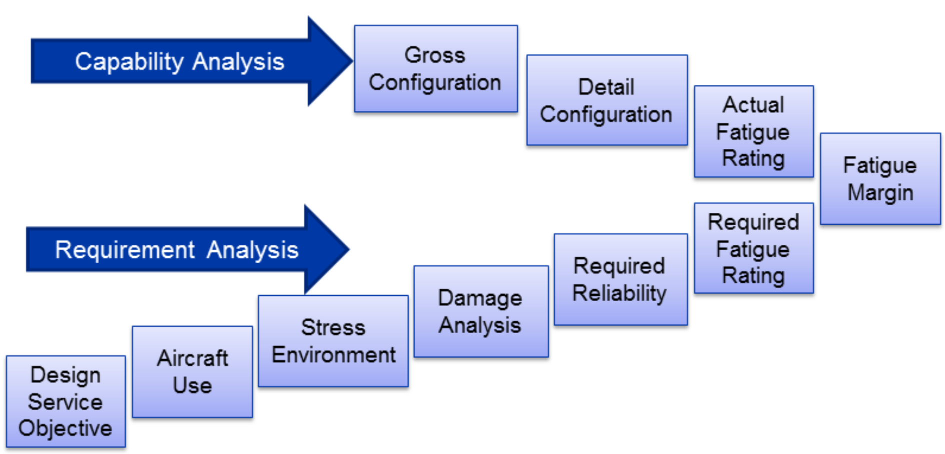


Fig. 4 – The Book 2 fatigue analysis process.

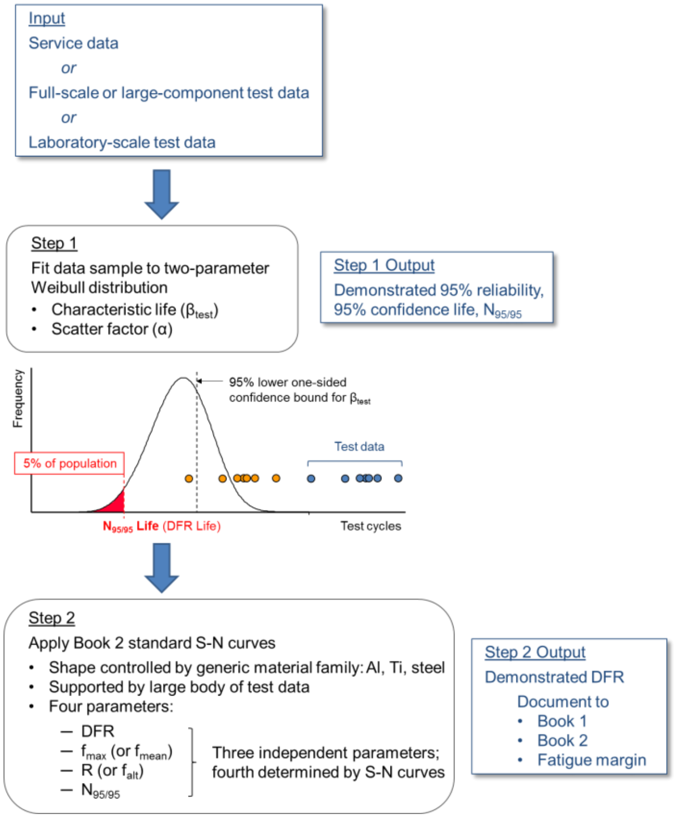


Fig. 5 – Determining DFR values from empirical data.

Table I. Boeing commercial airplane usage statistics, based on May 2016 data.

Airplane Model	EIS† Date	Airplanes (Active Commercial Fleet Only): Delivered	In service	Average age, yrs.	High Time Airplanes: Flights‡	Hours‡
727	1964	1,827	24	37.2	87.7	93.2
737-100/200/300/400/500	1968	3,098	695	22.8	97.3	102.2
737-600/700/800/900	1993	5,659	5,384	7.6	46.7	64.4
747	1970	1,489	501	14.8	39.1	136.5
757	1983	1,043	715	21.4	45.6	104.2
767	1982	1,061	738	17.9	50.0	125.0
777	1995	1,396	1,333	8.8	38.7	92.4
787	2011	378	378	1.7	6.4	17.9
DC-9	1965	927	2	36.0	108.1	109.3
MD-80	1980	1,191	351	25.2	60.3	90.0
MD-90	1995	116	71	19.0	40.9	56.9
717	1998	155	151	14.2	57.1	46.6
DC-10	1971	386	48	38.4	46.7	131.0
MD-11	1990	198	117	21.8	19.3	109.4

† Entry into service
‡ In thousands

The introduction of the durability method as well as active corrosion prevention programs has had a dramatic and almost immediate effect on fleet maintenance, as Fig. 6 shows. The 757 and 767 were the first models to take advantage of these efforts, resulting in a drastic reduction of maintenance service hours, a trend that has continued on newer airplanes, the 737 NG and 777, further enhanced by improvements to the methods and the continuous learning process afforded by fleet observations. This trend is also clearly evidenced in the many full-scale fatigue tests successfully completed or underway since then, which are discussed later in this paper.

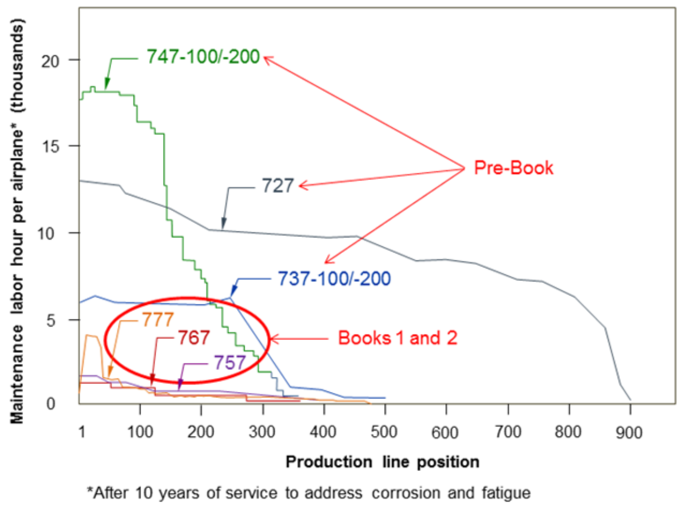


Fig. 6 – Impact of new design and analysis practices on fleet maintenance hours.

Despite the successes, there have been a few durability challenges as well, none perhaps as great for Boeing as the 737 fuselage skin longitudinal lap splices. Design, processing, and a complex stress environment required a number of iterations to achieve a lap splice design with a high level of durability. The evolution of some of the key lap splice design characteristics is illustrated in Fig. 7. A cold bonding process, which proved to lack the necessary robustness, was eliminated early on. Further tests led to the addition of hot bonded doublers and changes to the tear strap design and rivet diameters on later 737 Classic variants, significantly improving the durability and damage tolerance properties of the laps. The lap splices on the 737 NG airplanes represent a further improvement over earlier designs. Changes include increased skin gages (lower basic stress), widened tear straps (stiffening ratio increased), reduced eccentricity between upper and lower skin (lower local bending stresses), and machine-installed rivets (consistently better hole fill). The net results of these changes are illustrated in Fig. 8.

737 NG fatigue performance was successfully demonstrated on a 737-800 full-scale fatigue test to 225,000 simulated flights, or three times the 75,000-flight design service objective. High-frequency eddy current inspections of the rivet holes during teardown revealed very few crack indications, except at the manually assembled Stringer 14 lap splices (just below the window belt), where some small incipient cracks were found. Upper and lower row fatigue cracking was otherwise eliminated.

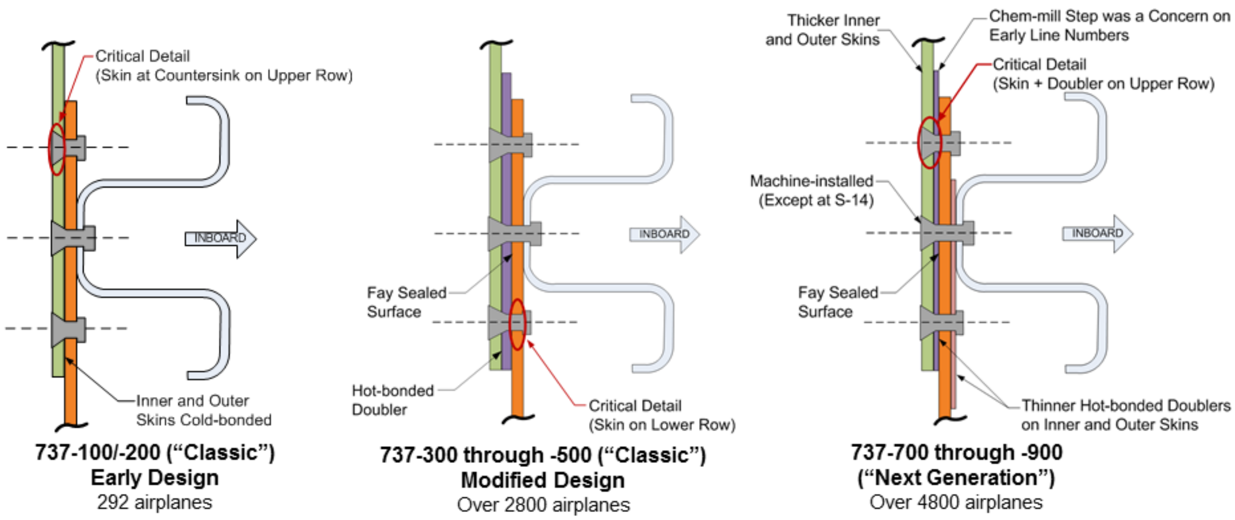


Fig. 7 – Boeing 737 fuselage skin lap splice design evolution.

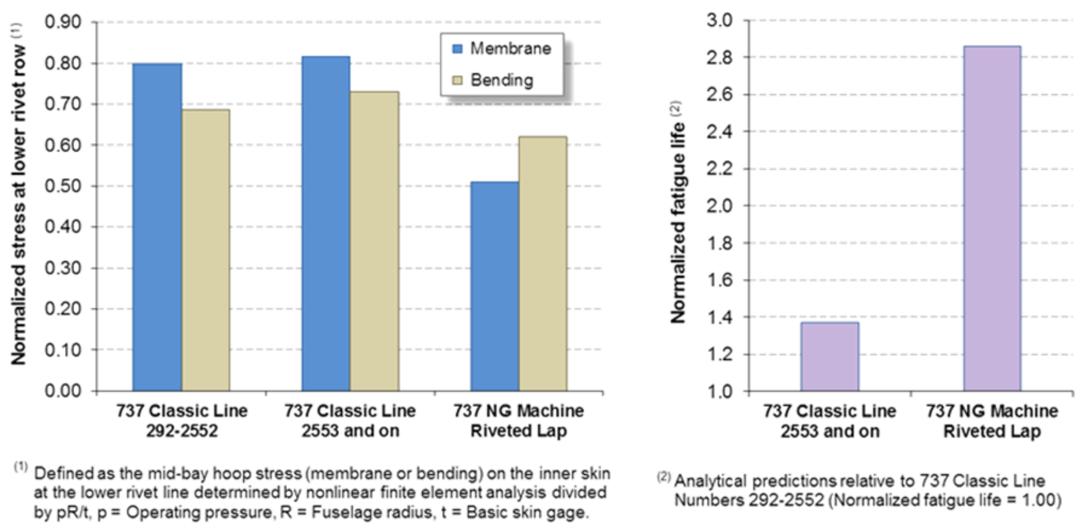


Fig. 8 – Fuselage crown longitudinal lap splice performance, Boeing 737 Classic vs. 737 NG (NOTE: For the 737 NG, the reference operating pressure is 7 percent higher than for the 737 Classics).

III. The Boeing Damage Tolerance Method

The accident at Lusaka in 1977 involving a Boeing 707 helped stimulate a 1978 revision of the regulatory requirements (14 CFR 25.571) [7]. That airplane, which had been designed and certified as fail-safe, was lost due to a horizontal stabilizer failure caused by fatigue cracking. The event demonstrated the limiting value of maintenance programs based solely on fail-safe design principles, which do not specify where to look and do not quantify how often to look for cracking. The accident occurred at a time when the United States Air Force (USAF) had already implemented damage tolerance requirements [8], and the commercial airplane industry was beginning to recognize that safe-life, fail-safe, and damage tolerance principles each have some inadequacies and that a combination of all three philosophies is needed in some cases. The subsequent changes in the civil regulations mandated the use of damage tolerant principles in all instances unless it imposes an unreasonable penalty. The Boeing damage tolerance analysis standard, commonly referred to as "Book 3" was first introduced in 1979 in response to the new requirements. Book 3 provides a method and design data for damage tolerance analysis of metallic structure. Similarly to the Boeing durability method, the Boeing damage tolerance approach was first implemented for new design with the 757 and 767 models. Book 3 was also used concurrently for supplemental structural inspection programs on the 727, 737 and 747 models.

A. Boeing damage tolerance method fundamentals

Damage tolerance is the attribute of the structure that permits it to retain its required residual strength for a period of use after the structure has sustained a given level of fatigue, corrosion, or accidental or discrete source damage. The basic requirements for damage tolerance are set by 14 CFR 25.571, which mandates damage tolerant design for structurally significant items (SSIs) or principal structural elements (PSEs). To clarify what is required, Boeing classifies aircraft structure into one of four categories, summarized in Table II. Any structural detail element or assembly that contributes significantly to carry flight, ground, pressure or control loads, and whose failure could affect the structural integrity necessary for the safety of the airplane, is characterized as a PSE. The changes in the regulations also mandated the development of inspection programs for new airplanes, the assessment of structural maintenance programs for existing airplanes, and initiated the development of supplemental structural inspection documents (SSID) in the late 1970's and early 1980's. Boeing retained the fail-safe requirement even though it is no longer required for certification.

Table II. Boeing structural classification for damage tolerance.

		Structural Category	Technique of Ensuring Safety	Technology Control Method	Structural Classification Examples
Other Structure		**Category 1:** Secondary Structure	Design for safe separation or loss of function	Continued safe flight	Wing spoiler segment (safe separation or safe loss of function)
Primary Structure	Damage Tolerant Design†	**Category 2:** Damage Obvious or Malfunction Evident	Adequate residual strength with extensive damage	Residual strength	Typical wing skin/stringer surface (fuel leak)
Primary Structure	Damage Tolerant Design†	**Category 3:** Damage Detection by Planned Inspection	Inspection program matched to structural characteristics	Residual strength and crack growth inspection program	All primary structure not included in Cat. 2 and 4
Primary Structure	Safe-Life Design	**Category 4:** Safe-Life	Conservative fatigue life	Fatigue	Landing gear structure

†Fail-safety required for Cat. 2 and 3. Additional WFD analyses for certain Cat. 3 structure

The key objective for airplane structure designed to be damage tolerant has always been to carry minimum regulatory loads until detection and repair of any fatigue cracks, corrosion, or accidental or discrete source damage occurring in service (Fig. 9). The best design has an allowable damage size that is obvious or malfunction evident to flight or ground crew personnel during routine activities around the airplane (Category 2). When this is not possible, some level of inspection is required to assure timely damage detection. Category 3 is primary structure that requires a planned inspection program to maintain structural integrity. Large allowable damage and/or slow crack growth, coupled with an easy inspection access, minimize operator inspection cost.

The fundamental elements of a classic damage tolerance analysis are residual strength, crack growth, and damage detection. An analytical approach to damage tolerance is complex even in its simplest form. In Book 3, Boeing has developed a method for damage tolerance analysis of metallic structure that is suitable for use by analysts having varying levels of familiarity with fracture mechanics concepts. Book 3 provides a frequently updated database of empirically-derived properties that determine the relationship between crack growth rate and stress intensity for various materials and environments [4, 5]. This document also provides a library of factors for a large number of frequently encountered structural configurations. Each of these factors is obtained by compounding individual component effects (e.g., fastener hole, free edge, fuselage curvature). Other factors are used to account for the effects of internal load redistribution caused by cracking of adjacent members as well as spectrum effects on specific configurations [5].

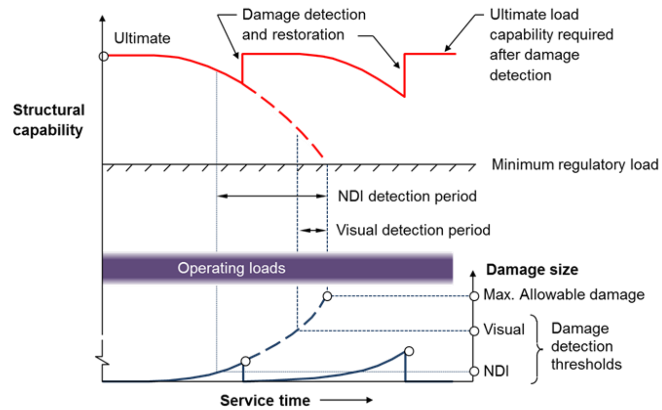


Fig. 9 – Damage evolution, detection, and repair as a function of service time in damage tolerant structure.

For the purpose of defining and managing the supplemental inspection program, Boeing uses a system referred to as Damage Tolerance Rating (DTR), which is built into the Book 3 method (Fig. 10). The DTR system is used to determine the adequacy of the baseline structural inspection program and if needed, to support the development of a supplemental program for detecting fatigue damage in all Category 3 SSIs/PSEs. This system is used to assess the probability of detecting fatigue damage assumed to have occurred in the fleet before the residual strength of the first cracked airplane falls below allowable limits. The adequacy of the structural inspection program for each individual component is determined by comparing the probability of detection ("Total DTR") to a predetermined acceptable level ("Required DTR"). By establishing damage detection probabilities as a function of inspection methods and crack size, it is possible to significantly increase the damage detection period [9, 10]. In contrast, the commonly used practice of setting inspection intervals to one-half of the damage detection period fails to provide quantitative damage detection reliability and does not capture the combined benefits of visual inspections performed during normal maintenance programs with targeted non-destructive inspection (NDI) methods [9, 10].

All damage-tolerant structure requires an inspection plan where inspection methods, thresholds, and frequencies are defined consistent with the stress levels and crack growth propagation characteristics. Normal maintenance inspections should allow the damage (developing at single or multiple sites) to be found before the crack reduces the residual strength capability of the structure below regulatory requirements. If the normal (or baseline) maintenance inspection program is not sufficient to meet the required

DTR, supplemental inspections and repeat intervals are established for the airframe structure by the airlines and approved by the regulatory agency. For this purpose, Boeing provides the DTR forms to the operators for each PSE. Operators use these DTR forms to develop their own repeat inspection interval for each PSE with the goal to achieve or exceed the required DTR established by the regulatory authorities. A DTR Check Form allows operators the flexibility to customize their supplemental inspection programs. Also, Boeing establishes inspection thresholds by crack growth analysis, assuming there is an appropriate initial manufacturing flaw, or by fatigue analysis, depending on the compliance approach set by the applicable airworthiness standard. Traditionally, a single threshold for the supplemental inspections is used for each group of PSEs of each model.

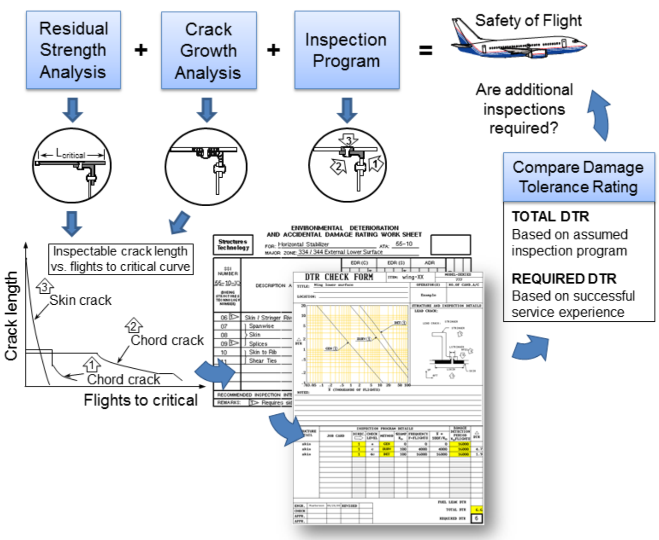


Fig. 10 – Fundamental elements of the Boeing damage tolerance analysis method.

Maintenance programs at Boeing are based on product testing, long fleet experience, and standard industry practices. All known forms of structural degradation are considered, providing the necessary inspections for damage tolerance, allowing for a timely detection of critical forms of damage including environmental (e.g., corrosion and stress corrosion) as well as accidental damage. Comparison with past successful practice is the primary means by which maintenance inspections or other procedures for accidental and environmental damage are substantiated. The process for establishing the baseline program takes into consideration an environmental deterioration rating (EDR) and accidental damage rating (ADR). The EDR/ADR rating systems are used to develop the initial structural inspection program for new Boeing airplanes.

B. Widespread fatigue damage

The Boeing 737 Aloha accident and a number of findings on other aircraft show that under certain conditions and in certain types of structure, multiple adjacent cracks can nucleate independently and eventually coalesce, resulting in faster crack propagation and a loss of residual strength that would not be accounted for in a classic damage tolerance analysis. This could in turn potentially invalidate a conventionally developed airframe inspection program. The limiting factors in the classic damage tolerant design philosophy are the initial assumptions concerning crack sizes and locations; the analyst needs to assume some form of dependent damage in the structure and in effect construct a potential failure scenario. With WFD, the analysis has to account for possible crack coalescence or other forms of interacting damage nucleating from multiple independent sources (Fig. 11).

Federal Aviation Regulation 14 CFR 25.571 was amended (25-96) in 1998 to include a requirement that sufficient full scale test evidence exists demonstrating that WFD will not occur within the design service life of the structure. In addition, Advisory Circular AC 91-56A was released at the time to provide guidance to airplane manufacturers and operators in the development of a WFD program that would preclude operation in the presence of WFD. The Boeing method for WFD evaluation, accepted by the FAA, makes full use of current industry practices and related FAA Advisory Circular material. The approach utilizes a comprehensive, multistage analysis procedure that combines statistics, fatigue, crack growth, residual strength, and test/service data to provide a simple analysis tool aimed at precluding the occurrence of WFD in the fleet.

In the Boeing process, the Inspection Start Point (ISP) is set at a level that will provide damage detection prior to a certain subset (typically 1%) of the airplanes in the fleet reaching a state where a particular amount of similar structural details are cracked on a number (typically the first) of similar component on the airplane. The Structural Modification Point (SMP) represents a conservative, lower bound estimate of possible WFD occurrence and is the point by which susceptible structure should be modified. The SMP is meant to provide an equivalent reliability level of a two-lifetime full-scale fatigue test, and provide enough opportunities to find damage on the lower bound airplane, typically the worst 5% of the airplanes in the fleet. Collectively, ISP and SMP define the beginning and end points of the damage detection period, sometimes referred to as the monitoring period for multi-site damage/multi-element damage (MSD/MED) detection (Fig. 12). Once the ISP and SMP are determined, the full inspection plan for the WFD program can be developed. The appropriate inspection methods and intervals are tailored to ensure timely crack detection with high confidence during the monitoring period, relying on the use of the Boeing DTR system. The Boeing WFD approach has been discussed previously in some depth [11], so no further elaboration is necessary here.

C. Limit of validity

In a continuing effort to address aging aircraft issues involving WFD, the FAA issued a new rule, finalized in January, 2011 designed to protect most of today's commercial planes and those designed in the future from WFD as they age [12]. Since then, all airlines that operate airplanes under 14 CFR Parts 121 or 129 have been required to take action to revise their U.S. FAA-approved structural maintenance program to bring it into compliance with the new rule. This final rule amends FAA regulations pertaining to certification and operation of transport category airplanes to prevent WFD in those airplanes.

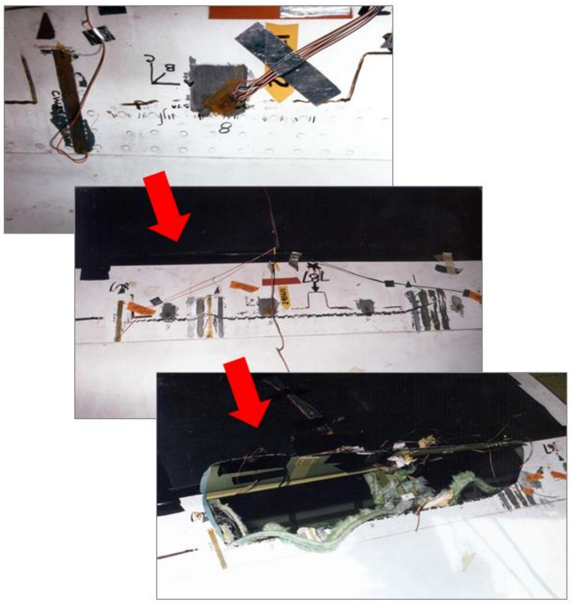


Fig. 11 – The classic WFD stages on a fuselage skin lap splice: Crack nucleation –> Coalescence –> Flapping.

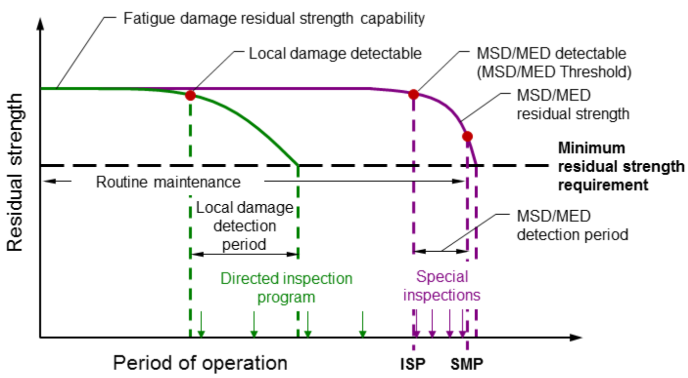


Fig. 12 – WFD typical implementation schematic.

The limit of validity (LOV) is a point (measured in flight cycles or flight hours) in the structural life of an airplane beyond which there is significantly increased risk of uncertainties in structural performance and the probable development of WFD. It represents an operational limit based on engineering data that supports the maintenance program. Therefore, all identified service actions are required for operation up to LOV. Any LOV extension requires additional fatigue test or service evidence and validation of the maintenance program for efficacy against WFD. The FAA defines the LOV as an airplane-level number, not a limit applicable to any particular part or component. When an airplane reaches its LOV, it must be retired from 14 CFR 121/129 operation; however, serviceable parts and components may be transferred to other airplanes provided the operator has complied with all existing continuing airworthiness requirements. With LOV now added as an absolute operational life limit, safety by retirement is reintroduced as a fundamental element of the safety process.

For certain existing airplanes, the new rule requires design approval holders to evaluate their airplanes to establish a LOV of the engineering data that supports the structural maintenance program. For future airplanes, the rule requires all applicants for type certificates, after the effective date of the rule, to establish a LOV. With the new rule, design approval holders and applicants must demonstrate that the airplane will be free from WFD up to the LOV. The rule furthermore requires that operators of any affected airplane incorporate the LOV into the maintenance program for that airplane and prohibits airplane operation beyond its LOV unless an extended LOV is approved. These requirements can drive a variety of maintenance program outcomes, as Fig. 13 shows.

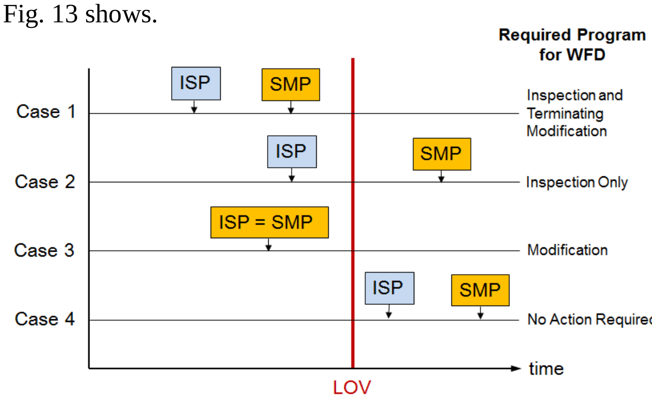


Fig. 13 – WFD Assessment outcomes as a function of the limit of validity (LOV).

The rule has been incorporated into 14 CFR regulation amendments 25-132, 26-5, 121-351, and 129-48. Additionally, the FAA has issued AC 120-104, which offers guidance on compliance with the new rule. Guidelines are provided for (1) design approval holders on establishing a LOV and how to address maintenance actions that have been determined necessary to support it, (2) operators on how to incorporate the LOV into their maintenance programs, and (3) anyone considering a LOV extension.

Nominally the LOV is assessed in terms of fatigue test evidence divided by a factor. FAA AC 120-104 describes the LOV as one-half of the fatigue test life representing the WFD average behavior. Boeing's approach to determine this value varies based on the amount of fatigue testing conducted for development of the final design requirements. Even though fatigue testing for certification purposes was not required for airplanes certified prior to 14 CFR 25 Amendment 96 (see Table III), in many cases such fatigue testing was actually accomplished. For those models certified prior to Amendment 96, fatigue test evidence is based not only on fatigue tests conducted at time of design, but also relies on post certification testing and in-service experience.

Table III. Part 25 Certification basis for Boeing and Douglas Heritage airplanes.

Model Definitions	
Pre Amendment 45 Airplanes	
Boeing Heritage Models	727, 737CL, 747CL, 747-400
Douglas Heritage Models	DC-8, DC-9, MD-80, DC-10/MD-10
Amendment 25-45 to 25-95 Airplanes	
Boeing Heritage Models	737 NG, 757, 767, 777-200/-300, 747-8
Douglas Heritage Models	MD-11, MD-90, 717
Amendment 25-96+ Airplanes	
Boeing Heritage Models	777-200LR/-300ER/F, 787

737CL: 737 Classic models are 737-100, -200, -300, -400, -500

737 NG: 737 Next Gen models are 737-600, -700, -700C, -800, -900, -900ER

747CL: 747 Classic models are 747-100, -100B, -100B SUD, -200B, -200C, -200F, -300, 747SP, 747SR

Table IV. Currently approved limits of validity.

Airplane Model	FAA Approved LOV Cycles	FAA Approved LOV Hours
717 ALL	110,000	110,000
727-100 L/N 1-47	50,000	50,000
727-100/100C/200/200F L/N 48 and on	85,000	95,000
737-100/200/200C L/N 1-291	34,000	34,000
737-200/200C L/N 292-1585	75,000	100,000
737-300/400/500 L/N 1001-2565	75,000	100,000
737-300/400/500 L/N 2566-3132	85,000	100,000
737-600/700/700IGW/700C/800/900/900ER	100,000	150,000
747-100/100B/100B SUD/200B/200C/200F/300, 747SP, 747SR	35,000	135,000
747-400/400D/400F, 747-400BCF(SF), 747-400LCF	35,000	165,000
757 ALL	75,000	150,000
767-200/200SF/300	75,000	180,000
767-300F/400ER	60,000	180,000
777-200/300	60,000	180,000
DC-8 ALL	56,000	125,000
DC-9 ALL	110,000	110,000
MD-80 ALL	110,000	150,000
MD-90 ALL	110,000	150,000
DC-10 ALL	60,000	160,000
MD-10 ALL	60,000	150,000
MD-11 ALL	40,000	150,000

LOV rule implementation is taking place in stages that are a function of the airframe certification basis. Table IV shows the currently approved LOV values for Boeing and Douglas Heritage airplanes. The LOV values for the pre-Amendment 45 airplanes (collectively known as Group 1) were submitted to the FAA in July 2012, and were approved by the FAA at various dates in August 2012. The 737 NG, 747-400, 757, 767, 777-200/300, MD-10, MD-11, MD-90 and 717 models (collectively known as Group 2) were approved by the FAA in July 2015. The FAA is expected to approve LOV values for the 777-200LR/-300ER/F, 747-8 and 787 models by 2017. These LOVs have been published in Airworthiness Limitations (AWLs) and Certification Maintenance Requirements (CMRs) documents.

D. Damage tolerance of repairs and alterations

On December 12, 2007, the FAA issued 14 CFR 26 Subpart E (26.41 – 26.49) entitled "Aging Airplane Safety – Damage Tolerance Data for Repairs and Alterations." Under the provisions of 14 CFR 26, Subpart E, any repair, master change service bulletin, or Supplemental Type Certificate (STC) affecting fatigue-critical structure approved by Boeing after January 11, 2008 is required to have a damage tolerance evaluation. Operators are also required to have adopted the damage tolerance provisions provided in the approval documentation for these repairs and alterations. After December 20, 2010, it became the operators' responsibility to ensure that all new repairs to fatigue-critical structure receive a damage tolerance evaluation and are properly documented in the damage tolerance-based maintenance program.

The objective of the rule is to support operator compliance with the damage tolerance requirements of the Aging Airplane Safety Rule (AASR), with respect to repairs and alterations. The rule requires identification of Fatigue Critical Baseline Structure (FCBS) and Fatigue Critical Alteration Structure (FCAS) on all Boeing models. Fatigue-critical structure is a new FAA concept introduced with this rule, defined as airplane structure that is susceptible to fatigue cracking that could contribute to a catastrophic failure, as determined in accordance with 14 CFR 25.571. A damage tolerance evaluation should be performed on repairs or alterations to FCBS or FCAS to determine if supplemental inspections are required. Existing repairs and alterations made before December 20, 2010 that affect fatigue-critical structure and that do not have approvals indicating a damage tolerance evaluation require further action.

The new rule had a significant impact on structural maintenance programs for all Boeing 7-series, as well as heritage Douglas transport-category airplanes. The requirements created a need for new criteria and methods for damage tolerance analysis of new structural configurations. Boeing has since published new and updated material, including lists of FCBS and FCAS for all affected models, making these materials available to assist airlines operating these models in complying with the new AASR [13].

Boeing has also developed a "Low Stress Criterion" for structural repairs that define stress levels below which no damage tolerance analyses and inspections need to be conducted on repairs on FCBS [13]. Often, in certain fatigue prone areas, stresses are so low that the fatigue lives could be several multiples of the Design Service Objective (DSO). Figure 14 shows an example set of criteria taken from Ref. [14]. Boeing has been able to develop these criteria, with a focus on reducing the scope of the effort, without compromising the structural safety of the fleet. This has been of considerable benefit to the industry, enabling operators to focus their attention and resources on the more critical structures that are above the stress levels defined by this criterion.

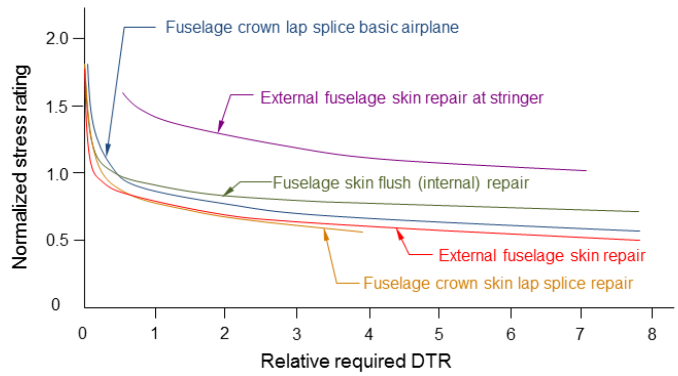


Fig. 14 – Normalized Stress Rating versus the multiple of required DTR for D/4C surveillance inspection for Pressure Critical Structure (reproduced from Ref. [14]).

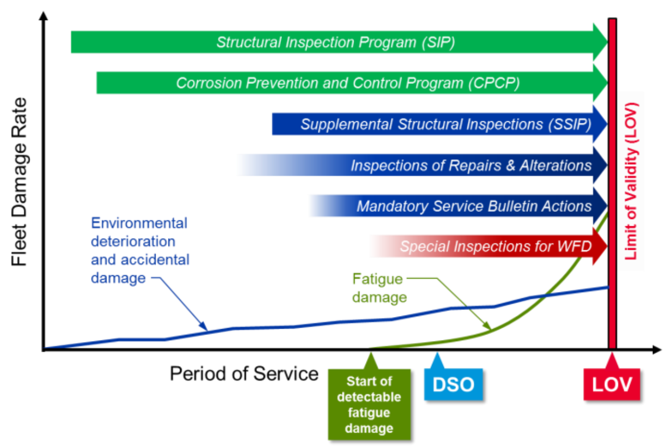


Fig. 15 – Strategies for ensuring fleet safety in the presence of fatigue, environmental deterioration, and accidental damage.

Figure 15 summarizes the inspection and maintenance philosophy discussed in this section for ensuring safety of the fleet in the presence of fatigue, environmental deterioration and accident damage. Boeing works with customers and regulators to understand any safety issue, then takes appropriate action to mitigate the risk and communicate changes to the fleet. New findings are documented and evaluated across all Boeing airplanes. Continued airworthiness is therefore assured by: (i) development of a baseline maintenance program, (ii) continued support of the baseline maintenance program, (iii) development of a supplemental maintenance program, and (iv) timely addressing of in-service findings.

IV. Durability and Damage Tolerance Methods Validation

The accumulation of test and service data over past decades has repeatedly shown that the effectiveness of Boeing DaDT methods is reflected in the quality and safety of Boeing airplanes. In this section, some of these trends and findings are highlighted, drawing specifically upon full-scale fatigue test evidence and teardown inspections. For brevity, the Boeing aging fleet survey program (initiated in 1986) and service demonstrated reliability methods based on fleet utilization/sampling are not reviewed in this paper. The interested reader may explore these topics in Ref. 1.

A. Full-scale fatigue test evidence

A full-scale fatigue test exposes a structurally complete airframe to the typical operating loads experienced by an airplane model fleet. Full-scale fatigue testing has long been a major part of Boeing structural performance data development, for both new models and airplanes retired from service. Though not the only evidence, such testing is primarily used to confirm the DaDT characteristics of the primary airframe structure, supporting a verification of the DSO and the proposed inspection and maintenance program. Additionally, it provides full-scale test evidence demonstrating the effectiveness of the design to preclude the possibility of WFD occurring within the DSO of the airplane, as required by the FAA since 1998 and by other regulatory agencies worldwide. Any fatigue cracking found in a full-scale fatigue test is assessed by the airplane program to determine whether design or manufacturing changes are needed. Internal guidelines are in place at Boeing to help make this determination. Figure 16 summarizes the current fleet data relative to DSO and historical full-scale fatigue test data for all Boeing twin and single aisle-category airplanes.

The effectiveness of the Boeing durability system, as introduced with Book 2 in the 1970's, was first evident from the 757 and 767 major airframe fatigue test findings. These airframes were fatigue tested to 2×DSO, with crack-findings leading to no more than a few dozen major design changes. This can be compared to the previous 747 major airframe fatigue test, which had roughly four times more design changes implemented after being tested to just one DSO. Later, full-scale fatigue testing of the 777 demonstrated further improvements relative to the 767 high-performance testing a decade prior with far fewer damage findings. This positive trend can be explained in part by the continued evolution, development, and consistent application of the Boeing DaDT technology standards.

The success thus far on the Boeing 787 full-scale fatigue test further demonstrates the effectiveness and continued maturation of Boeing durability methods. For the 787 airframe, Boeing set an objective to test to 165,000 flight cycles (3.75 × Short Mission DSO, equivalent to 75 years of service) in an equivalent 5×5 fatigue spectrum. This objective was reached on September 28, 2015.

The 787 test data compiled thus far has demonstrated quite clearly that the 787 outperformed the 767 and 777 in terms of fatigue performance. Figure 17 shows a comparison of fatigue damage findings from the 767, 777 and 787 full-scale fatigue tests. The relatively low number of 787 fatigue damage findings might be expected, to some extent, due to the fact that there is less metallic structure on the 787 airframe. However, one could argue that much of the critical, difficult-to-analyze primary structure remains metallic across these three airplane models. Although there can obviously be no contributions from crack findings in lap joints on the 787, many of the crack findings revealed in the

787 test have occurred in metallic structures within the fuselage.

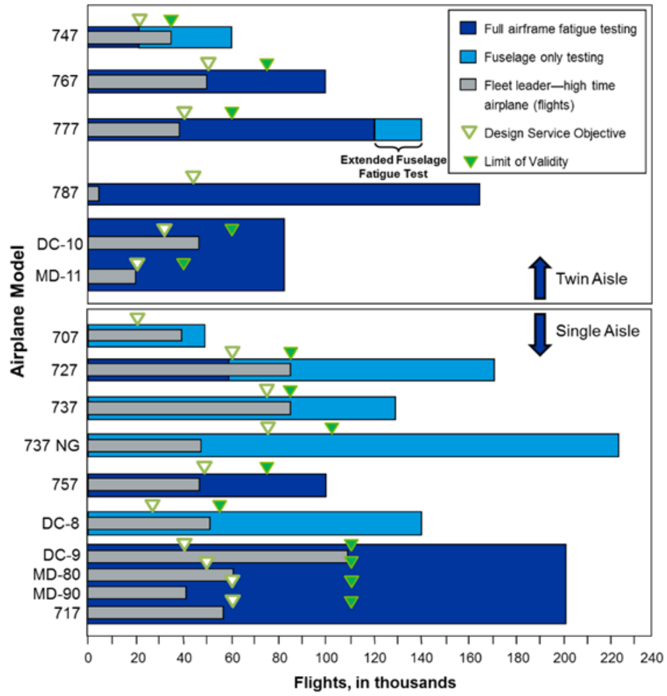


Fig. 16 – Boeing fleet data (as of May 2016) relative to DSO and full-scale fatigue test for twin aisle and single aisle-category airplanes.

Figures 18 and 19 show examples of two such structural findings, which are characteristic of the types of fatigue cracking observed beyond the first DSO. Another notable example, discovered early in the testing after 24,000 test cycles, was the failure of pins on the bearing pads of the main landing gear trunnion upper housing. A service bulletin was subsequently released in June 2013, and the bearing pad design was changed.

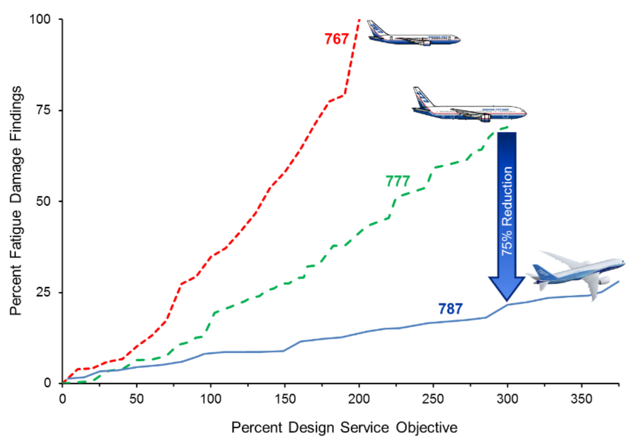


Fig. 17 – Comparison of findings from the 767, 777 and 787 major airframe fatigue tests.

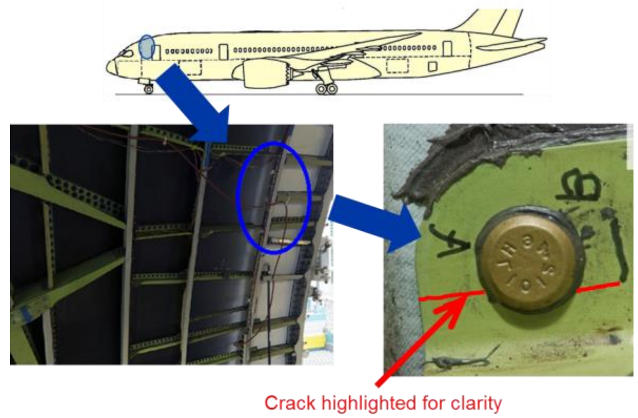


Fig. 18 – Example of a 787 full-scale fatigue test crack finding. Cracked fuselage door surround intercostal clips, found at the 72,000-flight inspection. In this case, the corrective action was to remove and replace the cracked parts.

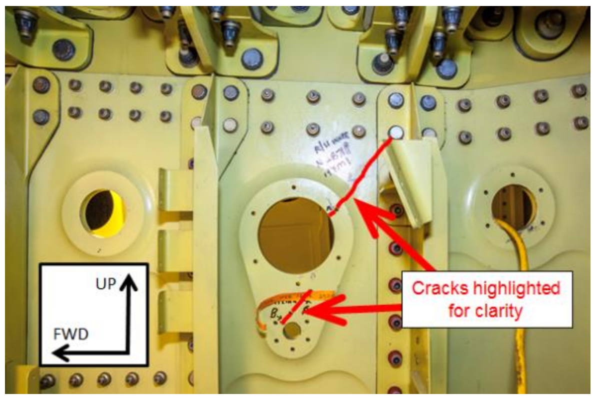
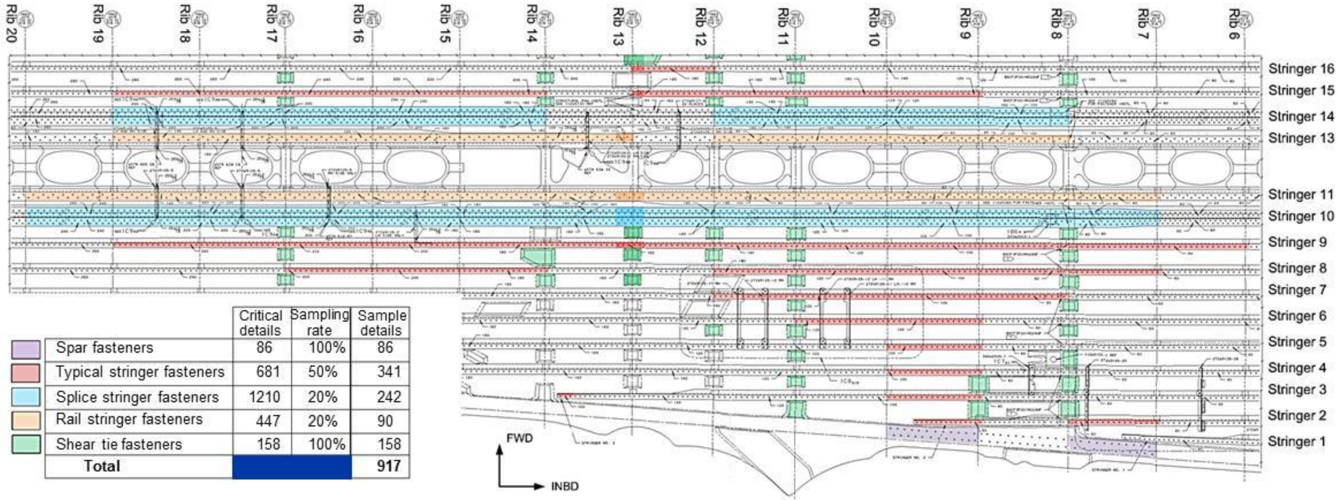


Fig. 19 – Example of a 787 full-scale fatigue test crack finding. Cracked satellite holes in the left and right (depicted here) side-of-body ribs immediately forward of the rear spar, found at the 132,000-flight inspection. This cracking was deemed acceptable for continuing cycling with increased surveillance.

B. Teardown inspections

Full-scale airframe testing is followed by extensive teardown inspections to locate any obvious problem areas. Since the introduction of the 707, Boeing has conducted several teardown inspections and evaluation of high-time airplanes. Major teardown efforts supplementing the 707, 727, 737 and 747 airplanes have been described elsewhere in much detail [15]. In this paper, we focus on more recent developments with the Boeing 777 teardown activities. The 787 full-scale fatigue test teardowns were completed in late 2016.



	Critical details	Sampling rate	Sample details
Spar fasteners	86	100%	86
Typical stringer fasteners	681	50%	341
Splice stringer fasteners	1210	20%	242
Rail stringer fasteners	447	20%	90
Shear tie fasteners	158	100%	158
Total			917

Fig. 20 – Boeing 777 full-scale fatigue test teardown activities on the outboard wing lower surface.

The 777 airframe was originally tested to 120,000 cycles, or 3×DSO, demonstrating excellent durability performance. Later, an extension of the full-scale fatigue testing was conducted on the 777 fuselage structure only, reaching as high as 3.5×DSO [16]. The primary objective of this extended test was to obtain additional crack growth data in the fuselage to support the structural maintenance plans for future aging fleet programs. Another objective was to develop and validate analytical procedures for calculating parameters that characterize WFD. The test demonstrated outstanding fatigue performance with the successful test completion to 140,000 cycles, simulating 70 years of service, well above regulatory requirements. The test showed excellent correlation of crack growth data with analysis, validating analysis methods, and verified the damage tolerance capability of the 777 using intentional damage (saw cuts) introduced earlier at 100,000 cycles.

To assist in establishing an airplane LOV, the new rule recommends the Type Certificate holder conduct effective teardown inspections of fleet or fatigue test articles to detect and characterize any MSD or MED that could result in a WFD condition. At the conclusion of the 777 full-scale airframe fatigue test, limited (sampling) teardown inspections were conducted for several WFD-susceptible items for evaluation. More recently, extensive teardown inspections were conducted for several WFD-susceptible items including the aft pressure dome, circumferential body joints, chemically milled steps in skin panels, fuselage skin to stringer attachments, and the outboard wing lower surface. Fasteners were removed from the outboard wing lower surface for open-hole high-frequency eddy current (HFEC) inspections (Fig. 20). Fastener locations inspected included skin attachments to splice stringers, shear ties, and the rear spar. No detectable cracking was found among the nearly 1,500 fastener locations inspected, demonstrating excellent fatigue performance and supporting the development of the 777 LOV.

Using the limited teardown inspections of the 777 aft pressure dome completed at the conclusion of testing, a preliminary WFD evaluation was conducted to conservatively estimate the required ISP and SMP based on the data from the cracked samples excised from the dome. This analysis demonstrated that the airplane would be free from WFD over its 20-year DSO. To better understand the WFD performance of the 777 pressure dome relative to the proposed 777 LOV, more extensive teardown inspections were recently conducted by removing nearly 2,000 fasteners for open hole HFEC inspection. Based on the inspection results, several segments of the radial lap splices were excised and disassembled for laboratory evaluations of crack indications. Using these observations, the WFD behavior of the aft pressure dome lap splices has been re-evaluated to fully characterize the structure in order to meet the certification requirements and ensure safe operation up to the LOV.

In addition, several fuselage lap splice repairs installed during the 777 full-scale fatigue test were also recently excised in an effort to determine whether repairs are potentially susceptible to WFD. Over fourteen longitudinal bays of fuselage skin repairs accumulated between 40,000 and 58,000 cycles of fatigue testing. The majority of the repairs performed well, with no visible cracking observed during the test. Partial teardown showed that some small cracks had nucleated at a small percentage of the inspected fastener holes. The results were compared to Book 2 predictions, which matched quite well. The findings were also used to develop large (greater than five-bay) lap splice repairs.

Similar teardown efforts were conducted in other areas of the 777 as well to support the WFD and LOV assessments. Supplemental teardowns over and above the original teardowns were also recently conducted for the 737 NG, 757, 767 and 777.

V. Durability and Damage Tolerance of Composites

After many years of limited application on commercial airplane structures, composite materials, especially carbon fiber-reinforced polymer composites (CFRP) are now being used in broad areas of the airframe. The Boeing 787 airplane is a prime example of this technological shift, with composites adding up to roughly one-half of the structure by weight (Fig. 21). Among the benefits composites realize are (1) the ability to more easily integrate large components in production, (2) lighter structure, and (3) greater resistance to fatigue and corrosion. For the 787, the expectation is that the latter attribute in particular will enable a doubling of the standard maintenance intervals, which for a heavy maintenance check will now become 12 years rather than the current 6 used on legacy metallic airplanes, translating into a substantial reduction in operating costs.

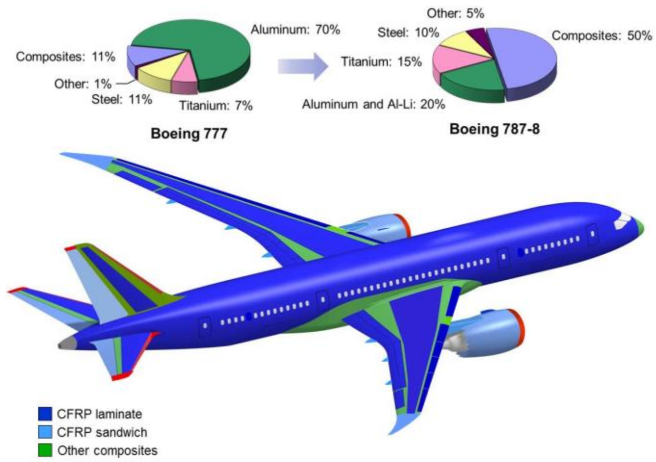


Fig. 21 – Boeing 787 Composite materials usage.

As Fig. 22 illustrates, expansion of composites technology has been made possible to a significant extent by the introduction of large-scale automated manufacturing and assembly processes [17], allowing—

- Construction of larger assemblies
- Significant reductions in tooling
- 30-40 percent shorter assembly flow
- Repeatable processes that create consistent first-time quality
- Sizable reductions in hazardous chemicals and waste

Incorporation of composites into primary structure on Boeing airplanes to the extent that they are used today on the 787 has been the culmination of a process that began in earnest in the late 1970’s and saw its first major production application on the 777 empennage. As part of the NASA Aircraft Energy Efficiency (ACEE) program, five 737-200 CFRP stabilizer shipsets were designed in 1978 and placed into service in 1984. After several years of service, two of the stabilizers underwent partial teardowns and a third stabilizer had a more thorough teardown inspection, which included mechanical property testing and nondestructive inspections (Fig. 23). Inspections found little deterioration due to wear, fatigue, or environmental factors. Test specimens excised from the stabilizers had residual strengths comparable to values measured more than 20 years earlier [18, 19].

Fig. 22 – Large-scale composite structure manufacturing development for the Boeing 787 program. Numerous fuselage test barrels were constructed at Boeing and at 787 Partner facilities. The first composite fuselage section (bottom) was unveiled in January 2005.

Other prototyping and technology development efforts followed, including a full-scale fatigue test of a prototype composite horizontal stabilizer based on the Boeing 767 planform, and the Boeing ATCAS (Advanced Technology Composite Aircraft Structures) program, whose goal was to support development of the technology required for cost- and weight-efficient use of composite materials in transport fuselage structure. This latter work was initiated in 1989 under a NASA contract as part of the Advanced Composites Technology (ACT) initiative [20].

The next major milestone was the successful development of the 777 composite empennage (Fig. 24), which has been in production since the early 1990’s. Certification of the empennage was supported by full-scale fatigue testing to three times the design life, following countless other mechanical tests ranging from small coupons to configured elements covering basic material properties, environmental effects, joints, durability, effects of defects, and repairs. In addition to satisfactory fatigue performance, the full-scale tests demonstrated greater than 150 percent of design limit load capability with barely visible impact damage (BVID) after fatigue loading as well as good correlation with analytical predictions [21].

First flight of the Boeing 787-8 was on December 15, 2009, followed by certification in August 2011 and first delivery to customer All Nippon Airways (ANA) on Sept. 25, 2011. Today, progress continues with the 787-9 and 787-10, the newest members of the 787 family. The 787-9 took flight on Sept. 17, 2013, launching a comprehensive flight-test program leading to certification and first delivery to launch customer Air New Zealand in June 2014. The third and longest 787, the 787-10, achieved firm configuration in April 2014 and is on track for delivery in 2018. Test and service experience on the composite structure has thus far been very positive.

Fig. 23 – Boeing 737 composite horizontal stabilizer technology demonstrator.

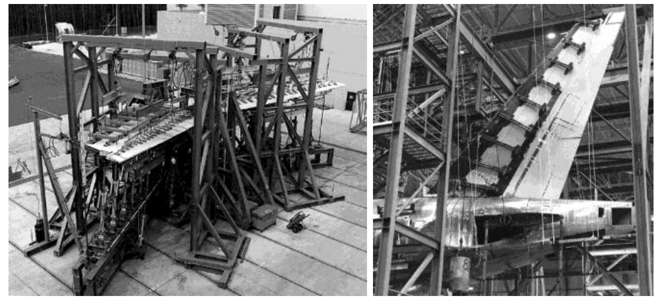


Fig. 24 – Boeing 777-200 composite horizontal stabilizer and vertical fin full-scale fatigue tests (1993-94).

A. Durability and damage tolerance of composites

The damage tolerance philosophy and regulatory requirements discussed earlier are applicable to all primary structure for the most part without differentiation between metals and composites. However, composite materials have some unique features that affect how damage tolerance principles are implemented.

Relative to metals, composites offer significant benefits in terms of specific strength and stiffness, tailorability, and resistance to fatigue and corrosion. Composites, however, do not possess the ability to blunt large stress concentrations or absorb damage (e.g., impact) by plastic deformation as metals do, and when built into laminates, they contain inherently weak interlaminar planes that can facilitate damage progression. Secondary loads, which are almost impossible to eliminate from complex built-up structure, often produce out-of-plane loading, resulting in interlaminar stresses that can cause delaminations. The diversification of fatigue damage (e.g., fiber breakage, matrix cracking, matrix crazing, fiber buckling, fiber-matrix interface failure and delamination), damage mode interactions, non-uniform damage development, and the inelastic behavior of composites during cyclic loading make analytical modeling difficult. Furthermore, linear damage accumulation hypotheses, such as the Palmgren-Miner Rule, which is often used for fatigue (crack nucleation) evaluations in metals, may not be adequate or even relevant for composites.

In assessing the types of damage posing the greatest risk to the integrity of composite structure, foreign object impact is usually among the chief concerns. Electrical discharge (including lightning strike) and major discrete source damage events such as ground collisions, engine rotor burst, bird strike, and fire/thermal are also considered. Figure 25 shows how design load levels vary in general terms as a function of damage severity, based on the guidance in [22]. Figure 26 takes this concept further using residual strength, illustrating how the guidance is typically applied, starting with barely visible impact damage (BVID, Fig. 27), which requires a demonstrated ultimate design strength capability and no detrimental damage growth during the design service life of the airframe with an appropriate factor on load, and at the opposite end, discrete source damage (DSD), which requires continued safe flight and landing loads. In between, maximum design damage (MDD) is established relative to a limit load requirement, which is also known as visible impact damage (VID, Fig. 27). Note that in general a strength evaluation for the structure in a pristine condition is of relatively little value as a result of this approach. This also means that any composite structure strength evaluation is in effect a damage tolerance assessment, because the analysis must consider the potential presence of undetectable damage such as manufacturing flaws and BVID.

As with other forms of discrete source damage, composite structures must meet the same lightning strike regulatory requirements as their aluminum counterparts. At Boeing, composite structures are designed to withstand significant strikes with only superficial damage, considered to be approximately 80-90th percentile strike energy levels. This damage will be generally expected to be within the Allowable Damage Limit (ADL), and the airplane can as a result be dispatched with a deferred structural repair. In similar aluminum structure, the strike would likely puncture the panel, requiring an immediate structural repair and thus putting the airplane temporarily out of operation.

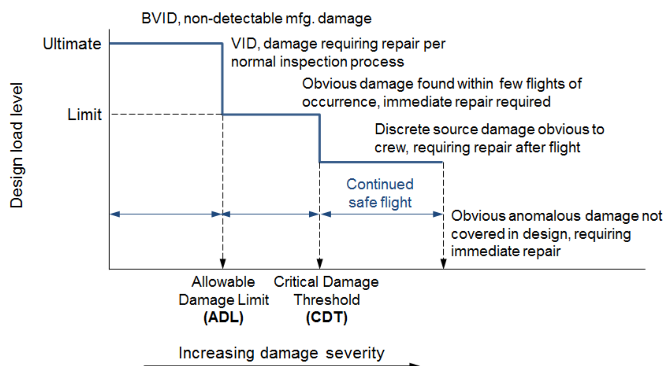


Fig. 25 – Design load dependence on damage severity (adapted from [22]).

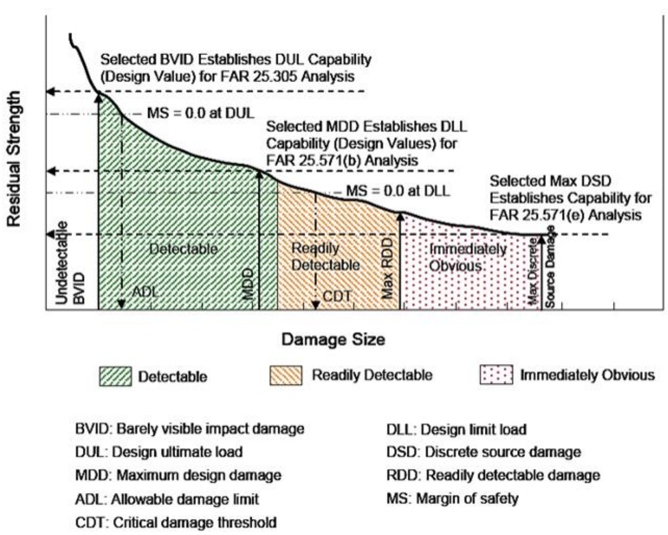


Fig. 26 – Design damage limits for composite structure.

Visual inspections in the field cannot reliably detect many of the small defects and impact damage that composite structure is typically sensitive to. There are effective NDI methods that may detect many of these types of defects or damage; however, the implementation has proven difficult in practice. Furthermore, damage progression analysis methods for composite structure are not mature enough today to enable development of a reliable inspection program for composite primary structure similarly to the way metallic structure is handled. For these reasons, evaluation for composite structure (according to [22]) used at Boeing since the NASA ACEE 737 program, and now including the 787, have thus far been based on a principle of no detrimental damage growth (NDDG). This approach assures no detrimental growth from undetectable defects and damage during the service life of the aircraft with a high reliability and confidence level as substantiated by fatigue test evidence.

Guidelines for demonstrating NDDG can be found in [22]. In keeping with that approach, structural details, elements, and subcomponents representing critical areas of the structure are tested in a manner that replicates anticipated service use and takes into account the effect of environment. Test articles feature damage representative of the types of conditions that may arise during fabrication, assembly, and in service consistent with the inspection techniques used. In these tests, residual strength is the primary test variable. The test goals are a function of whether the damage is categorized as undetectable (e.g., a small embedded manufacturing flaw, BVID) or detectable (from ADL to CDT). For the former, the main objective is usually to demonstrate a static ultimate load capability after cycling to a sufficient number of design lifetimes, taking into account reliability targets and application of an appropriate Load Enhancement Factor (LEF). For detectable damage, a successful demonstration of NDDG will show that following cyclic loading for at least two inspection intervals the damaged structure can sustain DLL with adequate reliability (refer to Fig. 26).

Although Boeing damage tolerance analysis methods cover a wide range of large damage types, large-notch analyses tend to be the most commonly performed evaluations. Experience gained in the design of the 787 shows that residual strength of fuselage structure with VID is enveloped by large-notch analysis. These analyses are usually based on parameters developed using various modeling strategies, including power-law (Mar-Lin [24]) fitting of test data on flat, featureless panels with center notches as the basic structural element. Damage nucleation from fatigue is not usually the norm for composite primary structure that is sized to meet the NDDG requirement. Instead, the environmental deterioration and accidental damage rating (EDR/ADR) system, which is also used for metallic structure, serves as a basis for a baseline structural inspection program for timely detection of environmental deterioration and accidental damage. Accordingly, there are generally no published fatigue thresholds for composite structures, and the baseline structural inspection program will carry on past the DSO and up to the airplane LOV.

Boeing has created companion technology standards for Books 2 and 3 specifically for DaDT of composites. Book 2C (Fatigue) and Book 3C (Damage Tolerance) for composites were released in 2014 and 2008, respectively. These documents are based primarily on experience gained in designing 777 empennage and 787 primary structures, with emphasis on methods and design values for CFRP solid laminates. Book 2C provides standard S-N curves for various structural details, treatment of thermal effects on fatigue, and statistical factors to account for the higher data scatter exhibited by composites relative to conventional metallic materials, including LEF determination criteria. Book 3C provides residual strength methods for composites, encompassing methods and allowables for structures with VID or large-scale damage.

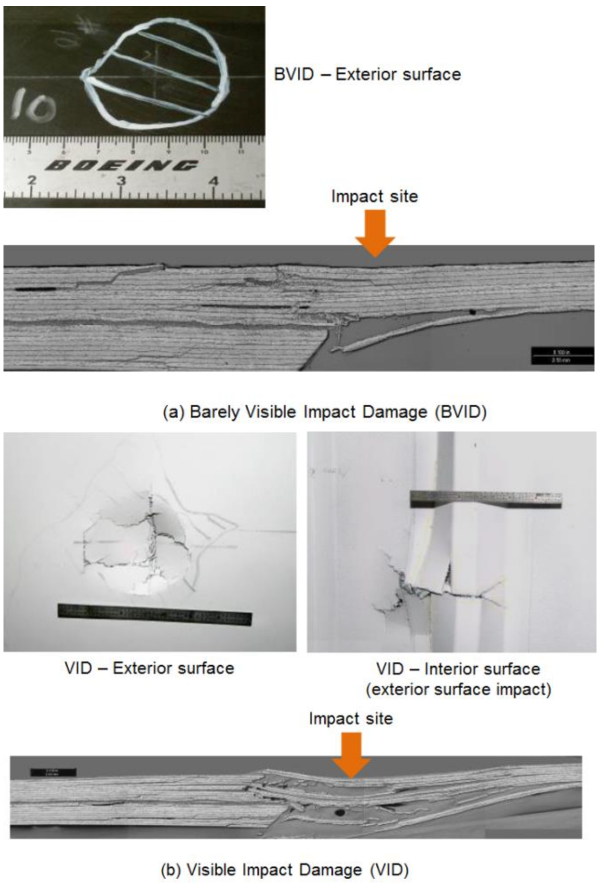


Fig. 27 – Typical damage: (a) BVID vs. (b) VID (from [23]).

B. Composite-metal structural interaction

While much attention in an airplane using composites to the extent that the Boeing 787 does is naturally focused on the composite structure, metallic elements cannot be ignored. Not only do they amount to a significant fraction of the structure (about 50 percent on the 787 –see Fig. 21), but they are also used in some of the most critical areas and experience some of the highest loads. One of the significant findings in hybrid structure is that when mated to metal parts, composites can create new conditions or exacerbate damage mechanisms in the metallic elements of the joint that can impact their durability (Table V). Other than perhaps corrosion and thermally induced stresses, this kind of adverse interaction is not always obvious or well understood [25]. The first two items in Table V have been the subject of significant work recently at Boeing and were deemed to be especially relevant to this paper, and are discussed below.

In mechanical joints, frictional load transfer has long been known to result in a lower rate of fatigue damage accumulation than when loads are predominantly transmitted by the fasteners in bearing [26]. Friction is promoted by joint clamp-up, which is driven mainly by fastener preload. When composite materials are present in the joint, some of the clamp-up can be lost due to the low transverse elastic properties of the material, thermal expansion differences, and creep. The latter can cause an irreversible loss in joint load transfer capability.

Fig. 28 shows some of the results of an internal Boeing study that collected bolt preload measurements over time in all-metal and CFRP-metal joints assembled with instrumented bolts. Comparing the two plots in the Figure, it is evident that the addition of a composite part causes the preload to drop by at least 10 percent in the first few hours following bolt installation, and that the loss is non-recoverable, contrasting with the stable behavior of the all-aluminum joint. In the Fig. 28 curves, the fluctuations between room temperature (RT) and cold values over time are the result of the differences in thermal expansion between the bolt and the composite, the latter usually having a much higher short-transverse expansion rate than the bolt material. A similar effect has been observed in joints cycled at high temperature, except the cyclic preload now peaks at temperature (rather than at ambient sea-level temperature) and the long-term loss of preload is magnified, the latter attributed to further softening of the composite. In airplane design, the loss of clamp-up can be addressed by either taking a reduction in joint fatigue capability (determined experimentally) of the metal details, by re-application of torque, or offset by specifying more capable fasteners [25].

Table V. Composite effect on the durability of joint metallic elements.

Composite attribute/feature	Possible impact to metal elements in hybrid joint
Short-transverse creep	Long-term reduction in joint clamp-up
Reduced electrical conductivity	Arcing damage induced by high electrical currents
Low in-plane thermal expansion	Thermal stresses
Low in-plane strain capability	Limits on hole treatments and fastener installation methods
Electrochemically passive carbon fibers	Susceptibility to corrosion when in contact with anodic materials (e.g., aluminum)

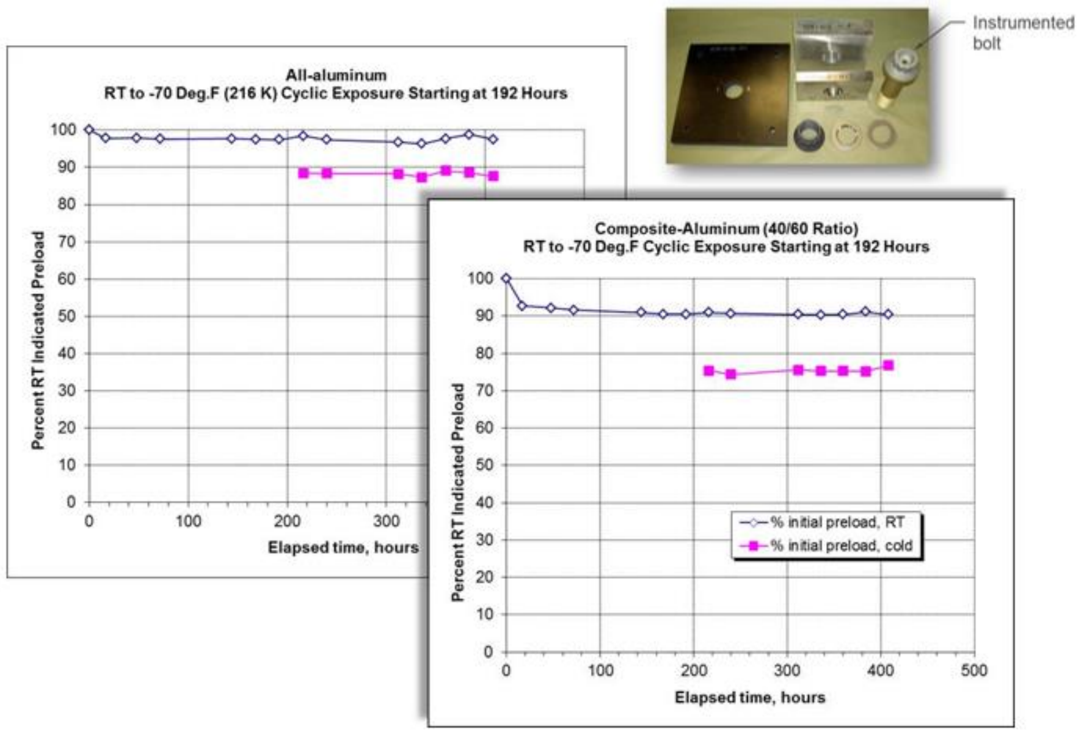


Fig. 28 – Half-inch (12.7 mm) nickel alloy 718 bolts in faying surface-sealed aluminum and hybrid joints aged at ambient temperature, then cycled cold [25].

Another item of particular relevance to this discussion is electrical discharge. Aircraft structures are exposed to a variety of electromagnetic effects threats: Lightning strike, high-intensity radiated fields/electromagnetic interference (HIRF/EMI), static discharge, and systems ground faults. With regard to lightning specifically, the design challenge is to provide current flow management to (1) protect passengers, crew, and systems, (2) prevent fuel ignition, and (3) minimize operator impact. New requirements, including some of the regulatory safety provisions in 14 CFR 25.981 concerning fuel ignition, make all joints and installations in fuel areas now potentially critical for lightning, and are driving significant changes to materials, finishes, and assembly processes. Additionally, greater composite usage results in a higher degree of sensitivity compared to conventional aluminum structure. Fatigue testing performed at Boeing on CFRP-metal joint specimens exposed to a range of electrical current levels has shown that under certain conditions, arcing can occur between the bolt and the hole as the current is transmitted across the joint, as evidenced by pitting and discoloration indicative of thermal damage in both the fastener and the hole on the metal part. These pits act as fatigue crack nucleation sites (Fig. 29, inset). The tests indicate that the severity of these conditions measured in terms of fatigue life degradation seems to depend on joint configuration, fastener finish (coated vs. bare), fastener fit, and current levels. An example is provided in the plot, illustrating the sensitivity to applied current of standard clearance-fit lockbolts installed in CFRP-aluminum joints.

New requirements and the potential for fatigue life reduction, along with the relatively common occurrence of lightning attachment on aircraft in service, make consideration of the phenomenon in the design process necessary. At joints, typical strategies involve reducing the current in the joint, minimizing the amount of energy discharged, and containing that energy. Energy discharge in particular appears to be best controlled with the help of fastener interference, usually by means of sleeve fasteners. For a given joint, the choice of fasteners and the level of shielding or containment required is usually a function of the location of the joint and the estimated magnitude of the threat.

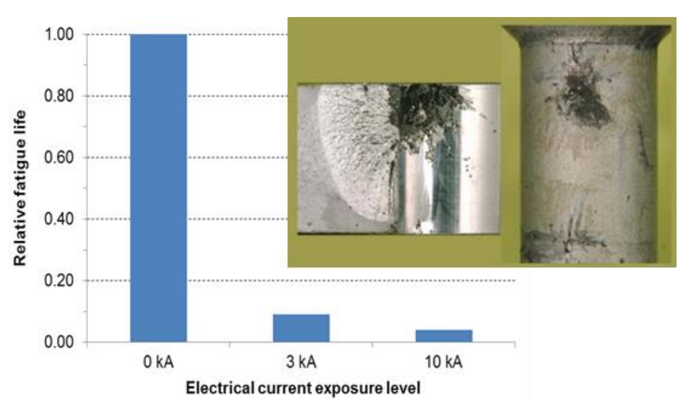


Fig. 29 – (Inset) Typical pitting and resulting fatigue crack resulting from a 10 kA per fastener step current through a hybrid CFRP-aluminum joint, standard titanium ¼-inch (6.4 mm) lockbolt installed in clearance and matching pin location. (Plot) CFRP-aluminum joint tests, standard titanium lockbolts installed in clearance [25].

C. Composite testing for the Boeing 787

Development of the 787 program has been made possible by significant advances in composite technology and data. For the 787, in addition to the above requirements, economic operation and maintenance of the airframe are being assured by application of the following criteria [23]:

- Allowable damage limits are based on damage that can be physically measured.
- Visual inspection techniques are the same as for current aluminum airplanes.
- Instrumented NDI will not be required for damage levels within published allowable damage limits.
- No new NDI techniques or equipment; planned inspections based on current Boeing 777 techniques and equipment modified to account for 787 structural configurations.
- Instrumented NDI may be required for damages that exceed published allowable damage limits.
- Methods validated by probability of detection studies and application to test articles.

Successful validation was accomplished thanks to a well-planned progression from laboratory-scale tests, configured elements, large panels (Fig. 30), subcomponents (Fig. 31), to full-scale fatigue testing (Fig. 32), culminating in a full-airframe fatigue test discussed earlier in this paper.

Repairs have also been the subject of a substantial amount of effort on this program. Numerous test articles ranging from coupons to components have featured repairs of the types planned for the Structural Repair Manuals, including bolted and bonded repairs. Tests have included static and fatigue with and without BVID, and including environmental tests.

Fig. 30 – Example of configured element testing of composites (In the photo: Residual strength testing of stiffened panel with BVID).

Fig. 31 – Subcomponent testing example: Co-bonded Boeing 787 composite "mini box" test article with VID.

Fig. 32 – Boeing 787 full-scale fatigue test examples: Top: Section 41 (forward fuselage). Bottom: Center wing and outboard wing stub box.

VI. Looking Ahead

A. Materials and basic part fabrication

Although much of the current focus in structural materials development is on composites, metals will continue to play a significant role in many areas of the airframe, and for the foreseeable future, will likely remain the dominant choice in short-haul, high-production rate airplanes. The industry's decades of familiarity with design, analysis, fabrication, and operation of metallic structures, and the considerable metals production and maintenance infrastructure are some of the reasons for that choice. In the meantime, metals technology has continued to evolve, taking advantage of new alloys, highly efficient basic fabrication techniques, and increased assembly process automation.

On the materials front, some of the best examples of this continued evolution are aluminum-lithium alloys. The so-called "third generation" aluminum-lithium alloys currently under development are characterized by reduced amounts of lithium (less than 2 percent by weight), but still provide 2 to

8 percent lower density than their conventional aluminum alloy counterparts [27], and slightly better or at least competitive mechanical properties (Figures 33 and 34). Parts produced from 2098 and 2099 sheet, plate, and extrusion are already in production in areas of the Boeing 787 fuselage, and newer alloys such as 2060 are being traded against other metals and composites on other models.

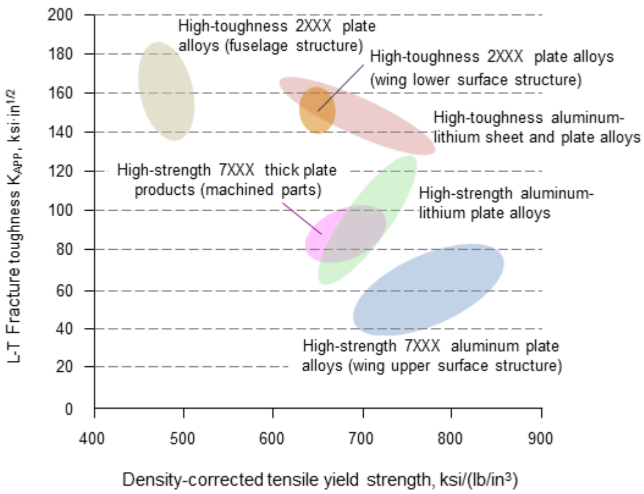


Fig. 33 – Fracture and strength properties of aerospace aluminum and third-generation aluminum-lithium alloys (Boeing data).

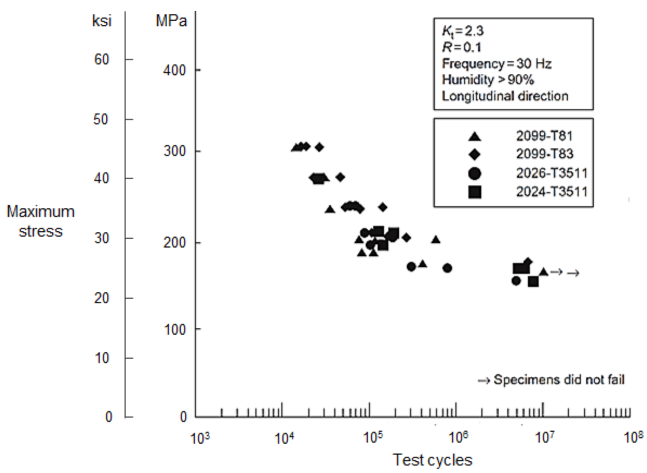


Fig. 34 – Typical constant-stress amplitude fatigue test data for notched aluminum (2024-T3511) and aluminum-lithium extrusions (reproduced from [27]).

Additional efficiencies can be gained by implementing advanced fabrication techniques. For metals, this is especially true for titanium, an impact that is compounded by the rising use of titanium alloys on new airplanes such as the 787. The high buy-to-fly ratios (the ratio of the weight of the raw material used for a component and the weight of the component itself) that are typical of machined titanium detailed parts (fittings, precision parts) are a measure of the significant recurring costs for these parts and high waste rates in the current fabrication process. Because of the intrinsic cost of titanium, even a moderate reduction of the buy-to-fly ratio can lead to significant cost savings. Welding, powder metallurgy and additive manufacturing are among the technologies being explored in pursuit of this goal (Fig. 35). Challenges from a DaDT perspective include development of design values in these uniquely configured, and often complex parts, non-destructive inspection, and characterization of the effects of highly process-dependent defects.

Lastly, the DaDT community is branching out to applications outside the traditional structures domain, into propulsion, systems, aircraft interiors, and non-traditional designs, where some of the biggest challenges are unique materials, designs, and construction methods (e.g., welding) that are distinct enough from primary structure that they require development of new or application-specific methods. Validation, and requirements that are often not as specific or as well controlled as in primary structure, add to the complexity of the task.

Fig. 35 – Sample advanced titanium fabrication processes. Left: Rough-machined titanium linear friction welded 787 floor beam fitting prototype. Right: Titanium additive manufacturing 787 side frame demonstration article.

B. Assembly processes

With fuel costs having doubled in the past decade as a percent of total operating costs and the new generation airplanes being able to deliver much improved fuel consumption, lower emissions, and lower environmental noise, orders for new and replacement aircraft will continue to grow at an extraordinary pace. The result is that a commercial airplane worldwide fleet of just under 21,000 airplanes at the end of 2013 is expected to increase to more than 42,000 aircraft by the end of 2033, of which roughly 37 percent will likely be replacements and 50 percent will represent new growth [28]. To meet that level of demand, production rates will need to grow to unprecedented levels. For example, the Boeing 737 model will see the current 42 airplane-per-month rate climb to as many as 52 airplanes per month by 2018, including both the Next Generation and MAX variants (Fig. 36).

On the 737 airplane family, the most significant production changes are impacting the wing line. The existing wing panel assembly process, which rivets wing skins and stringers while the wing is held in a stationary horizontal position is being replaced with a new high-precision automated system termed (vertical) PAL (Panel Assembly Line). PAL assembles the panels on a moving line at higher rates and in a vertical attitude, eliminating flow-intensive temporary fastening and overhead crane movement of parts (Fig. 37).

Qualification of the new PAL assembly system included a comprehensive fatigue test program, the goal of which was to show that for fatigue-critical areas of the wing, the new system meets or exceeds the fatigue performance of the older legacy processes. Laboratory-scale tests covered the range of fastener types and diameters and material stacks implemented on PAL for the 737 Next Generation and MAX models, using test specimens assembled on actual production equipment (Fig. 38). A similar fatigue test program has recently begun to evaluate and qualify the new 737 Spar Assembly Line (SAL).

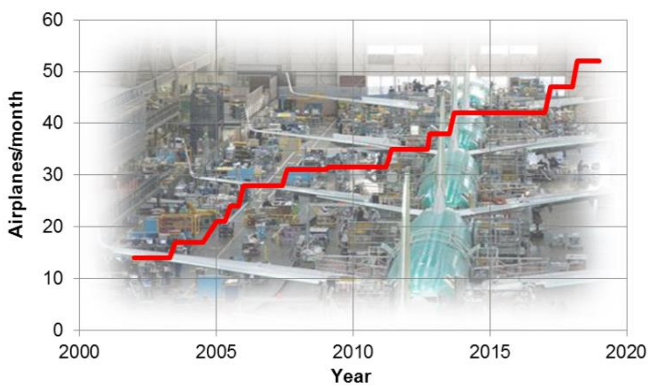


Fig. 36 – Production rate forecast through 2018 (left) for the Boeing 737 NG and MAX.

Fig. 37 – The 737 PAL wing assembly system. Automating the process of assembling the 737 wing panels is expected to reduce flow time and improve safety and quality.

Fig. 38 – Fatigue test specimen being assembled in a production 737 PAL gantry system.

On the Boeing 777 program, the focus is on a radically new fuselage one-up assembly process, FAUB, or Fuselage Automated Upright Build (Fig. 39). With FAUB, aluminum major fuselage sections are now built using pre-programmed, guided robot pairs that fasten the panels of the fuselage together with only minimal fixed tooling. Automated drilling operations will eventually install approximately 60,000 fasteners that are today installed by hand. In addition to benefiting production rates, FAUB will improve workplace safety and increase product quality.

Fig. 39 – The Boeing 777 FAUB fuselage assembly process.

Over the course of FAUB machine and process development, hundreds of specimens representative of various joints including riveted and bolted lap and circumferential fuselage splices were tested. Additionally, a 777 Freighter fuselage section featuring both the machine assembly process as well as manually assembled lap splices was built to allow a direct comparison of the current and new assembly methods. The 380-inch (9.65 m) long test article was assembled in Anacortes, WA and was transported by truck, barge and train 53 miles (85 kilometers) to the Everett Boeing facility for fatigue testing (Fig. 40). The test article was subjected to pressurization cycles equivalent to three times the design service objective. Subsequent non-destructive inspections and teardown evaluations demonstrated equivalent or better fatigue performance of the machine assembly process [29].

Fig. 40 – The Boeing 777 FAUB fuselage fatigue test article was subjected to pressurization cycles equivalent to three times the design service objective.

C. Analysis methods

The Boeing DaDT methods, as embodied in Book 1 through 3 (and their composite counterparts) have also undergone many changes over the years. These compendia are considered "living documents" that are periodically revised to adopt the latest improvements in analysis, loads, materials, and structural concepts, as well capturing new test and service data as it becomes available. As originally conceived, the methods were intended for a simpler, more conventional analytical treatment of the structure, loads, and stresses. As understanding of the operating loads environment and design data has expanded, methods have become more complex and a more efficient seamless integration of loads and stress data with the DaDT methods is becoming increasingly important.

One area that has been the object of considerable attention over the past decade has been the development of finite element analysis best practices in their application to both fatigue and damage tolerance assessments. Experience shows that the value and versatility of finite element analysis as an adjunct to DaDT evaluations can only be truly realized when the proper modeling strategy (both the type of analysis and physical model) is tailored to the specifics of the problem and evidence from either test or other previously accepted analyses can be used to validate the results. Figures 41 and 42 showcase elements of a particular initiative currently underway to improve the accuracy and consistency of fuselage lap splice fatigue life predictions, by developing and documenting a set of validated nonlinear finite element analysis standards and modeling techniques that can be easily used by analysts across airplane models as a complement to other existing methods [30]. Validation is being accomplished based on experimental data ranging from laboratory techniques such as optical digital image correlation (DIC) to airframe ground pressurization tests.

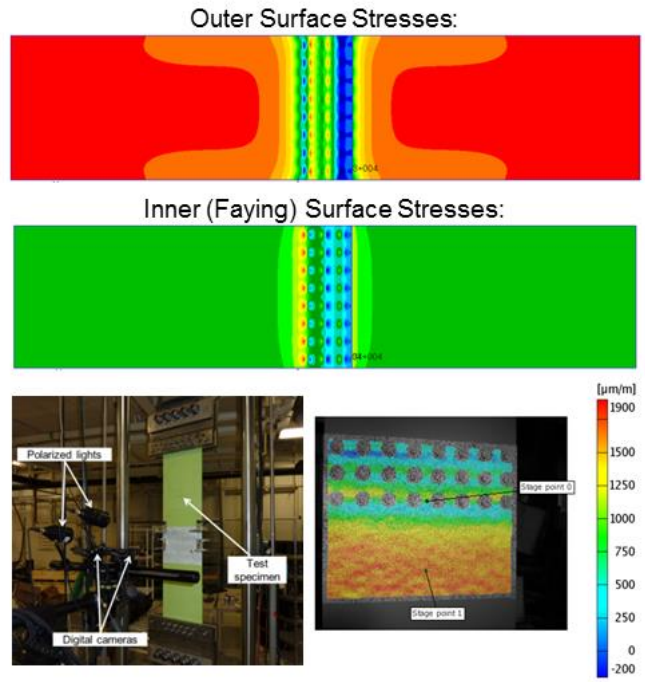


Fig. 41 – Fuselage lap splice analysis and laboratory-scale validation using optical DIC.

Similar efforts have been taking place on the damage tolerance side, the emphasis in most of these analyses being on development of stress intensity factors for complex assemblies. The example shown on Fig. 43 corresponds to a Boeing 747 passenger door cutout analysis, the goal of which was to generate Book 3 crack growth factors for the cracking scenarios depicted in the Figure. The model in this case represented a multi-bay section around the upper corner of the door cutout. Structural details included in the model were the bear strap reinforcement, edge frame outer chord, and the fasteners connecting the skin, bear strap, and outer chord. The analysis was handled as geometrically nonlinear problem. Stress intensity factors were calculated from the results of the finite element analysis using the Virtual Crack

Closure Technique (VCCT) [31]. Validation of the modeling strategy was accomplished by comparing strains drawn from an 'intact' version of the model with strain gage data taken from ground pressure test of an actual airplane.

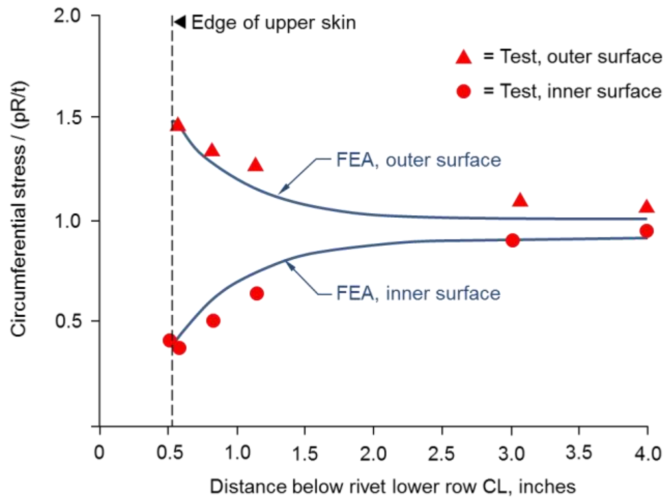


Fig. 42 – Finite element model correlation with ground pressurization test measurements on a Boeing 747 lower fuselage skin, below a longitudinal lap splice at mid-bay.

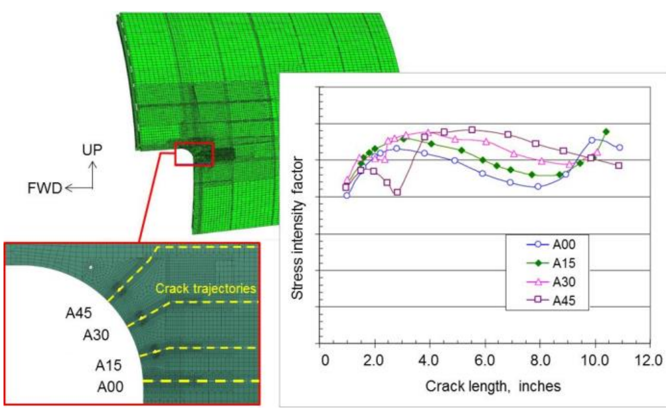


Fig. 43 – Boeing 747 Passenger door cutout crack growth analysis.

For composites, some of the greatest current challenges are in NDI, particularly in the application of the technology to large composite surfaces and in its ability to reliably detect degraded bondlines or interlaminar disbonds. There are NDI techniques that are capable of finding small delaminations, but they tend to also be time-consuming and require relatively high skill compared to other traditional inspection techniques. Boeing has been developing NDI methods that put energy into the structure transverse to the critical interfaces, scanning the structure as the energy pulse is applied. This technique has been shown to be capable of finding "weak bonds" or delaminations that are on the verge of damage nucleation and growth, but it is not yet ready for deployment in the field. The next issue is how to make use of that type of information once it becomes available in a damage progression or residual strength analysis. The analytical methods need to be quicker than existing VCCT-based damage propagation tools in order to support production-related rework and field repairs, and must be nimble and efficient enough to enable their use by non-experts.

Another area likely to receive some attention in the future is the evolution in composites from NDDG as an underpinning philosophy to a "predictable damage growth" approach. As composite usage is expanded, the desire to operate at higher strain levels in order to reduce weight will become an increasingly stronger motivation to steer away from the current application of AC20-107B toward the kind of safety-by-inspection approach that is commonplace with metallic structure. This will require methods and data capable of predicting the onset of delamination under combined cyclic loading and environment, which are currently very limited or unreliable.

D. The human connection

One challenge that is easy to overlook is the industry's ability to maintain skill continuity in this highly specialized (and sometimes subjective) technical discipline. Although U.S. Government projections indicate relatively low (less than 1 percent) annual employment growth rates in aerospace engineering over the next decade [32], the reality is that the industry is faced with significant turnover due to age demographics, globalization, shifting skill needs, and competition from other industries, at a time when commercial airplane demand is expected to surge. Boeing is projecting that about 50 percent of its top engineers and mechanics will be eligible to retire over roughly the next five years [33]. These are expected to become particularly acute issues in the fields of composite and stress engineering (including fatigue and damage tolerance), systems integration, and manufacturing/production engineering.

Boeing is actively engaged in efforts to maintain an experienced workforce through efforts in training and mentoring. In the DaDT field, training curricula have been developed and are being used across the company's design centers worldwide. Between 2006 and 2014, nearly 85,000 hours of specialized DaDT training were imparted to engineers supporting Boeing commercial airplane projects. Graduate certificate classes in partnership with local universities such as the University of Washington are being regularly offered. A further creative illustration of this commitment was the acquisition by Boeing of a retired 737 Classic (-300 L/N 1231) fuselage with slightly over 73,000 flights as a live training exhibit for structural engineers (see Fig. 44). Since acquiring the retired fuselage in late 2012, over 2,500 people have visited the hull. In addition, approximately 43 usage visits have been logged where the fuselage was used to either support a fleet investigation or testing of NDI equipment.

Fig. 44 – Retired Boeing 737 fuselage training aid.

VII. Conclusion

The DaDT methods whose foundations were laid down at Boeing decades ago have continued to prove themselves through service and test experience. As the regulatory environment has evolved and technology has progressed, these methods have had to keep pace or expand into new areas. Composites are a prime exponent of the latter. One other important success story has to be how Boeing developed supplemental structural inspections and integrated them into operator-applied maintenance programs, essentially converting fail-safe certified airplanes into damage tolerant ones. The success of these programs is evidenced in the ever-improving fleet safety record of the commercial airplane fleet, the majority of which consists of Boeing airplanes. Boeing has a monitored fleet in which events are evaluated using a common damage tolerance methodology, and maintenance programs are adjusted as necessary, while the airplanes are operated out to their 30-year life expectations and beyond.

Many challenges remain, not all of which are purely technical –for example, how experience and past lessons learned are preserved and built upon. It has been pointed out that with the introduction with LOV of a set of definitive structural life limits, DaDT regulatory requirements have come full circle back to the safe life philosophy of 60 years ago, and though the reality is of course more complicated, the industry would not have been able to progress without learning from experience that spans these many years. Maintaining that continuity in the midst of a major engineering generational transition will be one of our greatest future challenges yet.

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Biographies

Steve Chisholm is the BCA Director of Structures Engineering. In this capacity, Chisholm leads BCA Airplane Structures in support of Airplane Development, Airplane Programs, Product Development and Commercial Aviation Services. He also is responsible for driving functional excellence for all Structures Design and Stress skills across BCA and is the Structures Engineering process and skill owner for BCA. He is an Authorized Representative, he has long been involved in safety and compliance issues, and he was a member of the Boeing Technical Fellowship before entering management. He holds a Bachelor of Science in Mechanical Engineering from the University of Washington and a Master in Business Administration from Seattle University.

Antonio Rufin is a Senior Lead Engineer in the Structural Damage Technology (SDT) group for Durability and Structural Fatigue (Metals). He holds a Master of Science in Aeronautics & Astronautics from the University of Washington. In 2015 he was appointed Boeing Technical Fellow.

Brandon Chapman is a structural analyst in the SDT group, where he is primarily involved in the development of durability and damage tolerance analysis methods and allowables. He holds a Ph.D. in Physics and a Master of Aerospace Engineering in Composite Materials and Structures, both from the University of Washington.

Quentin Benson is a structural analyst in the 777 Aft Fuselage group. He holds a Master of Science in Mechanical Engineering from the University of Washington. In 2017 he was appointed Boeing Associate Technical Fellow.