JOURNAL OF AIRCRAFT
Vol. 37, No. 1, January–February 2000

Need for New Materials in Aging Aircraft Structures

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Post-Cold War political and economic considerations have resulted in efforts to extend the life of many aircraft that are the backbone of NATO operational forces. Although some are designated to be replaced with new aircraft, the replacement schedule for many often requires an unprecedented life span of up to 80 years before retirement. Aircraft within these older fleets have had, and continue to experience with growing frequency, fatigue and corrosion related cracking problems that are of concern to operators. To sustain their airworthiness and reduce the maintenance burden, structural components undergoing these aging problems will have to be repaired or replaced. Material development that has taken place since most of the older vintage military aircraft entered service has resulted in improved alloys and processes that can be used to upgrade life expectation, performance, and affordability of older systems. Some of these newer materials and the advantages they offer over their older counterparts are described.

Introduction

IN 1996 the U.S. Air Force requested the National Research Council (NRC) to identify research and development needs and opportunities to support the continued operations of their aging aircraft. The study was undertaken by a committee selected by the National Materials Advisory Board of the NRC, and the results published in the committee's final report.[1] Among the many committee recommendations, one was to develop guidelines for broadening the use of newer, improved materials as substitutes for incumbents with low damage tolerance and poor corrosion resistance. Such substitutions must also make good business sense to offset the up-front costs associated with change and to also assure materials will be available. Several examples of life-cycle cost benefits derived from applications of new materials to aging aircraft structure problems are given by Austin et al.[2]

The U.S. and other NATO-country air forces have many old aircraft that form the backbone of the total operational force structure. Many of these (e.g., the KC-135, the B-52, and the C-141) were introduced into service in the 1950s and 1960s. Even the F-15 air superiority fighter became operational 20–25 years ago, and the F-16 and A-10 combat aircraft at least 15 years ago. The old age of these aircraft, in conjunction with changes in usage and mission requirements, have increased downtime and repair costs associated with structural cracking and corrosion problems. Alloys developed since the 1980s have addressed these failure modes with better corrosion resistance and toughness improvement while also providing the necessary static strength and other properties.[3]

Structural (fatigue) cracking is a direct result of aircraft use (i.e., load or stress cycles) and will eventually occur in all aircraft. Corrosion results from the exposure of susceptible materials to various corrosive environments (e.g., humid air, saltwater, sump tank water, and latrine leakage) and to inadequate or deteriorated corrosion protection systems. In the case of aluminum primary structure, numerous service difficulties have been documented on components manufactured from alloys 2024-T3, 7075-T6, 7178-T6, and 7079-T6.

For example, to minimize structural weight and thus maximize payload capability, 7178-T6 was originally specified for the KC-135 Stratotanker wing (covers and stiffeners). Moreover, although predominantly 2024-T3, certain areas of the fuselage body skin were also designed as 7075-T6, as were portions of the fuselage stiffening structure. With the best technology available at the time, the aforementioned 7xxx series alloys were all designed to emphasize strength, which came at some cost to damage tolerance and corrosion performance. Concerned over 7178-T6 damage tolerance shortcomings, the U.S. Air Force in 1977 recommended redesign of the KC-135 inboard lower wing covers with alloy 2024-T3. To address the corrosion concern, newer and more corrosion-resisting 7xxx series alloys are now being routinely substituted for 7178-T6 on an attrition repair basis. Growing KC-135 maintenance issues have now reached a point where the U.S. Air Force is now recommending fleetwide replacement of the full outboard wing structure with new 7xxx series materials. More recently, the U.S. Navy and Lockheed Martin came to a similar recommendation and will apply new 7xxx series alloys to the rewinging of the P3 patrol aircraft fleet.

The potential threat of multisite damage (MSD) to integrity of older transport aircraft body splice joints is also a well-known concern[4,5] that (to be shown later) can be linked to a combination of age and the corrosion and damage tolerance performance limitations of the materials of construction.

Research since 1960 has led to the development of several new aluminum alloys, heat treatments, and processing methods that yield more damage-tolerant and corrosion-resistant alternatives for airframe components than those materials originally manufactured into the older aircraft.[3] The overaged T73 and T76 tempers were developed in the early 1960s to make 7075 more resistant to stress corrosion cracking and to exfoliation corrosion; however, the improvement obtained was at the expense of strength. In 1974 Cina obtained a patent[6] specifically targeted at 7075, for a heat treatment procedure to provide stress corrosion resistance equivalent to an overaged T73 temper while maintaining the peak-aged strength. Although the concept, called retrogression and reaging (RRA), seemed industrially impractical at the time, derivative tempers have been taken to practice, as discussed by Holt et al.[7] In the 1970s alloy 7050-T74 (formerly T736) was developed to fill the need for a material that would develop high strength in thick section products, good resistance to exfoliation and stress corrosion, and adequate fracture toughness and fatigue characteristics. Also, in the 1970s a derivative of 7075 (i.e., 7475) was developed that provided improved fracture toughness compared with 7075.

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Table 1 Longitudinal property comparisons for various 7xxx series aluminum alloy products

Alloy/temper	Ult. tens. strength, ksi (Mpa)	Tens. yld. strength, ksi (Mpa)	Compr. yld. strength, ksi (Mpa)	Elong, %	Kic, L-T toughness, ksi $(\text{in})^{1/2}$	Exco rating ASTM G34	SCC thresh. stress ST, ASTM G47, 20 days, ksi
			Plate, 1.00 in. (25.2 mm)				
7075-T651	79 (545)[a]	72 (497)[a]	70 (483)[a]	7	26 (typical)	ED (typical)	10 (typical)
7178-T651	84 (580)[a]	73 (504)[a]	73 (504)[a]	5	< 18 (typical)	ED	< 10 (typical)
7055-T7751	91 (628)[a]	88 (607)[a]	88 (607)[a]	7	26 (typical)	EB	15 (min)
7150-T7751	84 (579)[b]	78 (538)[b]	77 (531)[b]	8	27 (typical)	EB	25 (min)
7050-T7651	80 (552)[a]	71 (462)[a]	68 (441)[a]	9	31 (typical)	EB	20 (min)
7050-T7451	76 (524)[a]	67 (462)[a]	64 (442)[a]	10	32 (typical)	EB	35 (min)
7475-T7351	72 (497)[a]	62 (428)[a]	60 (414)[a]	10	50 (typical)	EA	40 (min)
			Extrusion, 0.500 in (12.7 mm)				
7075-T6511	85 (587)[a]	76 (524)[a]	76 (524)[a]	7	27 (typical)	ED (typical)	10 (typical)
7178-T6511	90 (621)[a]	81(559)[a]	79 (545)[a]	5	< 18 (typical)	ED (typical)	< 10 (typical)
7055-T77511	95 (656)[a]	93 (642)[a]	94 (649)[a]	9	30 (typical)	EB	15 (typical)
7150-T77511	88 (607)[b]	83 (572)[b]	83 (572)[b]	9	27 (typical)	EB	25 (min)
7050-T76511	79 (545)[b]	69 (476)[b]	69 (476)[b]	7	40 (typical)	EB	17 (min)
			Die-forging, 4.00 in (102 mm)				
7075-T6xx	73 (504)[b]	62 (428)[b]	——	7	29 (typical)	ED (typical)	10 (typical)
7055-T76xx	74 (511)[a]	65 (449)[a]	——	4	25 (typical)	——	35 (typical)
7055-T74xx	72 (497)[a]	62 (429)[a]	——	4	29 (typical)	——	35 (typical)
7050-T74xx	70 (483)[b]	60 (414)[b]	——	7	27 (typical)	EB	35 (min)
7175-T74xx	73 (504)[b]	63 (435)[b]	——	7	30 (typical)	——	35 (min)
7075-T73xx	64 (442)[b]	53 (366)[b]	——	7	——	——	42 (min)

[a]MIL-HDBK 5 Minimum "B" basis value. [b]MIL-HDBK 5 Minimum "S" basis value.

In the 1980s further derivative 2xxx series and 7xxx series alloys and temper combinations appeared, namely, 2524, 7150, 7055 and -T77 bringing enhanced combinations of strength, toughness, and corrosion performance. The 1980s also introduced a new generation of low-density Al–Li alloys (e.g., 2090, 8090, and 2091) developed to offer alternatives, other than increasing strength, for reducing structural weight. During the past decade other new alloy and process improvements have evolved (e.g., low porosity thick plate, improved stress relieved forgings) to address limitations found in pre-1980s aircraft materials. These newer alloy/process improvements offer root cause correction to many older aircraft maintenance problems while also providing the competitive durability advantages of new airplane maintenance. In addition, today's materials are available in forms and sizes that can accommodate parts consolidation to achieve significant reductions in joint counts, fastener counts, and installed cost.

The purpose of this paper is to review some recent advances in derivative alloys that have occurred primarily through application of a very large scientific knowledge base, tighter chemistries, and improved process controls. The newer alloys offer useful improvements in product performance, quality, and reliability that can be applied to aging aircraft problems and to thereby reduce ownership costs.

Recent Advances in Derivative Alloys and Tempers

Improvements in Strength, Corrosion Resistance, and Toughness

During aging aircraft retrofit, the substitution of alloys with equivalent strength but with higher corrosion resistance and fracture toughness will extend maintenance schedules, decrease down time, and reduce costs. As mentioned earlier, RRA demonstrated, in concept, that the microstructural factors determining strength and corrosion resistance can be decoupled to improve corrosion performance without substantial strength loss. In the 1980s work reported in Refs. 8–10 showed that beneficial RRA effects can be obtained in large components if the retrogression temperatures are below 200°C for 7075. Hepples et al.[11] showed that under commercially possible thermal process routes, RRA can be applied to achieve peak strength and high resistance to stress corrosion cracking (SCC) and exfoliation corrosion in the 7xxx series alloy system. Independently, Alcoa developed a patented aging process designed to overcome the commercial production impracticalities of RRA.[12] The new temper became known as T77, and although the details of the process are proprietary, Fig. 1 demonstrates the higher combination of strength

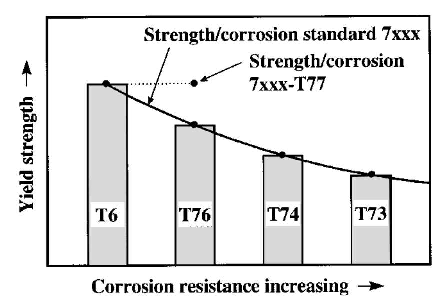


Fig. 1 Improvement in strength/corrosion combination due to the T77 temper.

and corrosion resistance made possible via implementation with the newly developed, controlled-toughness 7xxx series alloys, for example, 7050, 7150, and 7055. Some property comparisons for the new 7xxx series alloys are given in Table 1.

Alloy 7150-T77 has higher strength with durability and damage tolerance characteristics matching or exceeding those of 7050-T76. Boeing selected extrusions of 7150-T77 as fuselage stringers for the upper and lower lobes of the 777 jetliner because of the superior combination of strength, corrosion, and SCC characteristics and fracture toughness. Alloy 7150-T77 plate and extrusions are also being used on the new C17 cargo transport[3] and as a 7178-T6/7075-T6 replacement material in the U.S. Air Force KC-135 Stratotanker. Alloy 7150 is also being used to replace problem 7178-T6 and 7075-T6 materials in seat tracks and floors of older transport aircraft. Improved fracture toughness of 7150-T77 products is attributed to the controlled volume fraction of coarse intermetallic particles and unrecrystallized grain structure, whereas the combination of strength and corrosion characteristics is attributed to the size and spatial distribution and the copper content of the strengthening precipitates.[3] The improvement in properties using the new temper, relative to older alloys and tempers, is illustrated in Fig. 2.

Alloy 7055 was developed by Alcoa for compressively loaded structures.[13] Alloy 7055-T77 plate and extrusions offer a specific strength increase of about 10% relative to that of 7150-T6 (almost 30% higher than that of 7075-T76). These products provide high resistance to exfoliation corrosion, similar to that of 7075-T76, and toughness and fatigue crack growth resistance similar to that of 7150-T6. In contrast to the usual loss in toughness of 7xxx series products at low temperatures, fracture toughness of 7055-T77 at

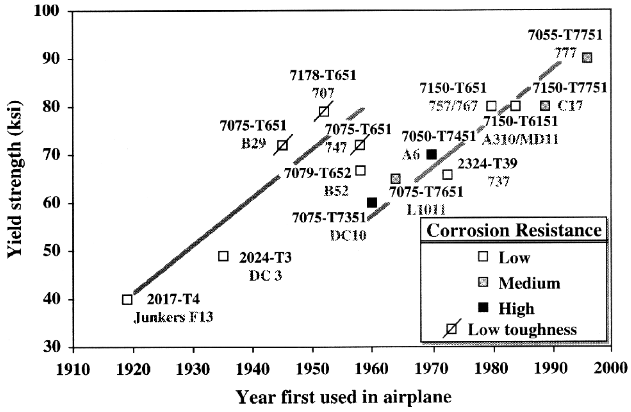


Fig. 2 Comparison of strength and corrosion resistance of various aluminum aerospace alloys as a function of year first used in airplane.

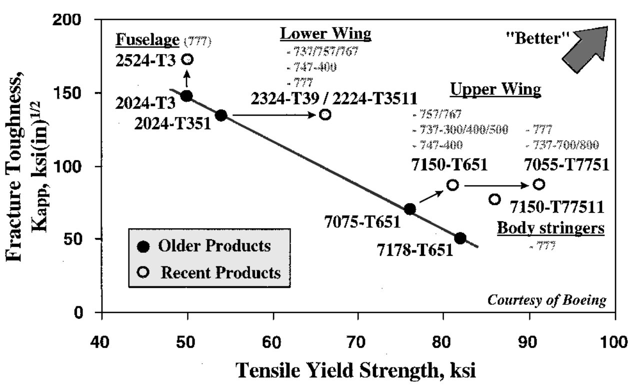


Fig. 3 Strength–toughness property combinations of older and newer aluminum alloy products.

-65°F (22 K) is similar to that at room temperature. Resistance to SCC is intermediate to those of 7075-T6 and 7150-T77 products (Table 1). The attractive combination of properties of 7055-T77 is attributed to its high ratio of Zn:Mg and Cu:Mg. When aged to -T77, this composition provides a microstructure at and near grain boundaries that is resistant to intergranular fracture and to intergranular corrosion. Additionally, the matrix microstructure is resistive to strain localization while producing a high strength. Alloy 7055-T7751 is used as the upper wing skin material for the Boeing 777 aircraft and more recently specified for the same on the A340-500/600 aircraft. The improved strength–toughness property combinations of newer alloys and tempers, relative to the older ones, are illustrated in Fig. 3.

Improvements in Material Durability/Damage Tolerance

Extension of airframe service beyond original life goals can potentially introduce MSD states, such as widespread fatigue or widespread corrosion that, if uncontrolled, may imperil the safety and economic life of the aircraft. For this case, the traditional damage tolerance inspection requirements directed at the presence of a single crack are inadequate. Moreover, analysis and testing of pristine structure may not assure all critical locations are known when corrosion is present.[1] This realization and the desire for reliable, longer-lasting aircraft has given rise to new requirements that in-service age degradation be accounted for in aircraft system design, maintenance, and airworthiness certification processes.[4,5] This philosophical shift creates the opportunity for affordable replacement materials that can not only resist the occurrence of MSD, but that also offer improved structural damage tolerance with undetected MSD or corrosion present.[14,15]

The occurrence of widespread damage sites can be associated with the intrinsic characteristics of the material microstructure.[16] Material microstructural sites prone to the development of cracklike

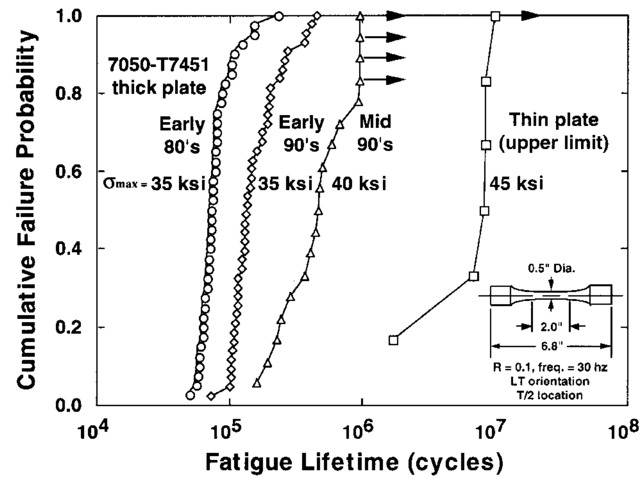


Fig. 4 Fatigue lifetime improvements in 7050-T7451 thick plate due to process refinement.

damage, attributable to corrosion or fatigue, can be associated with particles, inclusions, pores, and grain boundaries.[17] Although these features are necessarily a part of the material, the character of these features can be altered through modifications to material composition and processing while still meeting the material strength and damage tolerance capability goals.[18,19] The following are several notable examples.

Machined structures from plate thicker than 3 in. is often used to reduce part count and assembly costs associated with built-up components manufactured from thinner material. However, because the thicker plate undergoes less work than thin products there is a higher probability that porosity developed during the casting operation will not be sealed. Obviously, the higher porosity material has a poorer fatigue performance than lower porosity material. There has been continuous process refinement in the production of thick plate since the early 1980s that has reduced porosity as well as particle and inclusion size. Consequently, the improved fatigue quality of recent production plate materials, even in a one-to-one substitution, should retard the onset or spread of secondary fatigue cracking longer than products produced from pre-1980 production material. The effect of the process refinement on the fatigue lifetime of 7050-T7451 plate is illustrated in Fig. 4 (Ref. 18).

Other recent alloy developments have brought improvement to damage tolerance capabilities and fatigue strength in the presence of pitting corrosion. For example, alloy 2024-T3 sheet is often selected for fuselage skins because of its superior damage tolerance properties when compared to higher-strength 7xxx series products. An equivalent strength 2024-T3 derivative, 2524-T3, was recently developed by Alcoa[20] and, as shown in Fig. 5, offers substantial performance improvement over 2024-T3 wide panel toughness (R-curve) and fatigue crack growth ($da/dN - \Delta K$), the latter especially at high ΔK. The improvement was achieved by tightening control of composition and processing based on the knowledge that constituents associated with Fe and Si impurities reduce fracture toughness[21–24] and degrade resistance to both initiation[13] and growth[25] of fatigue cracks. Coarse primary phases formed when solubility limits are exceeded at the solution heat treatment temperature (or those formed during hot rolling and not redissolved during subsequent processing) have a similar effect.[26] Consequently, tight controls on chemistry (i.e., low levels of Fe and Si), balancing the Cu and Mg content to produce maximum strength without exceeding solubility limits at the solution heat treatment temperature,[27] and a controlled processing schedule are all necessary.[26] In controlling the Cu and Mg contents, the levels of Fe, Si, and Mn in the alloy have to be considered because the constituent phases in 2X24 are usually Al_7Cu_2Fe, $Al_{12}(Fe,Mn)_3Si$, $Al_6(Fe,Cu)$, and the dispersoid is $Al_{20}Cu_2Mn$.

The work of Chen et al.[28] demonstrated that pitting corrosion attack in 2024-T3 is highly localized to Fe and Si bearing

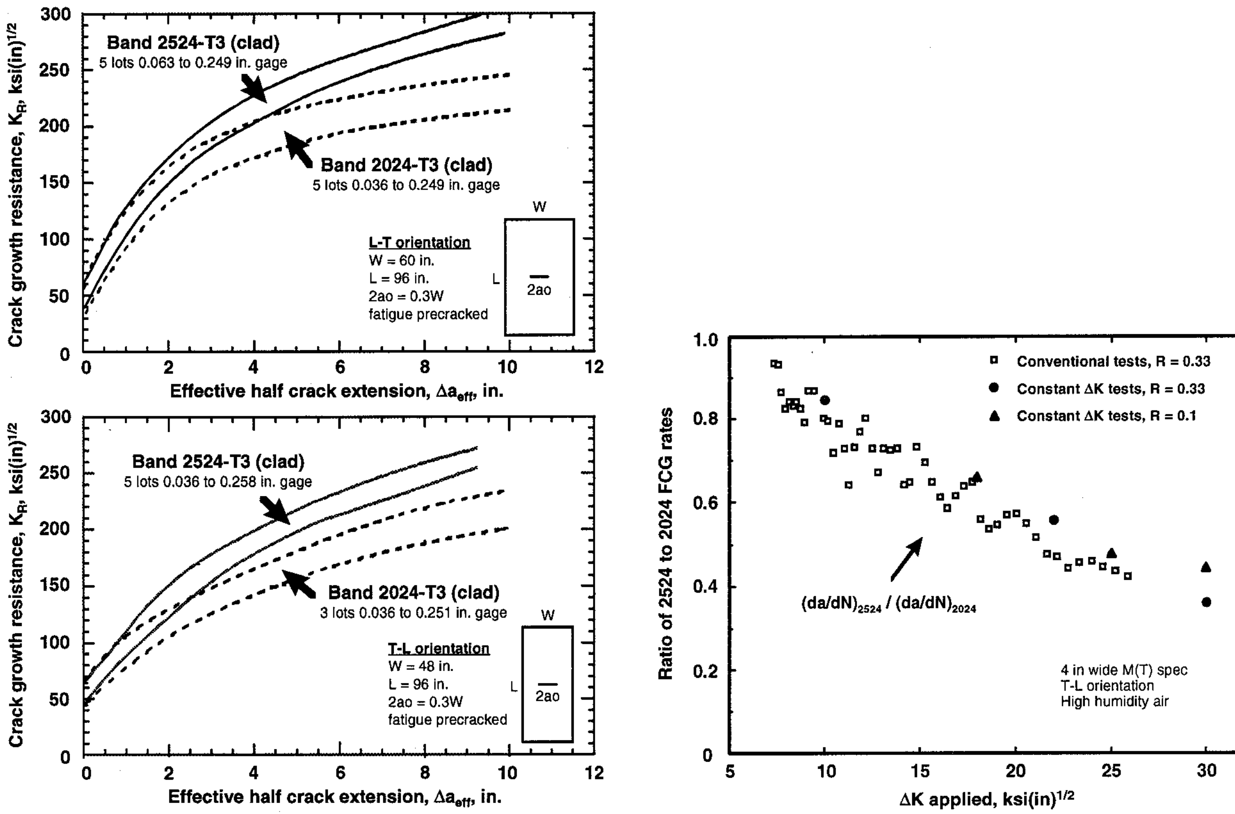


Fig. 5 Comparison of 2524 vs 2024 damage tolerance capabilities: wide panel K_R curves (*L-T*, top left and *T-L*, bottom left), and fatigue crack growth performance (right). Courtesy of Boeing.

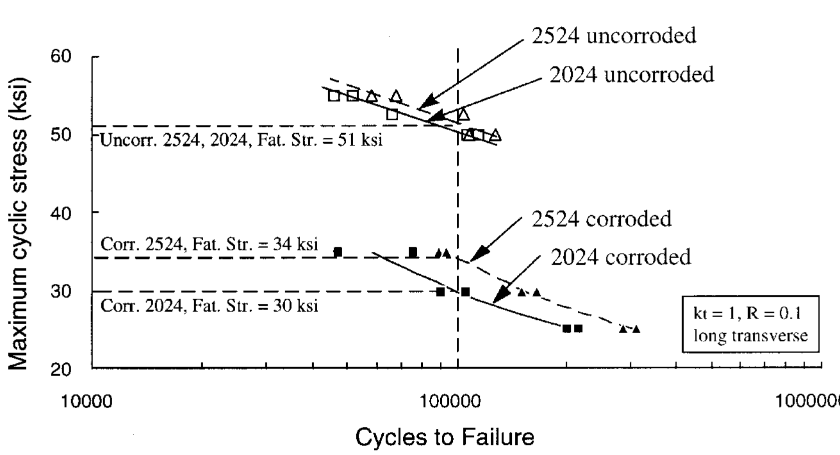
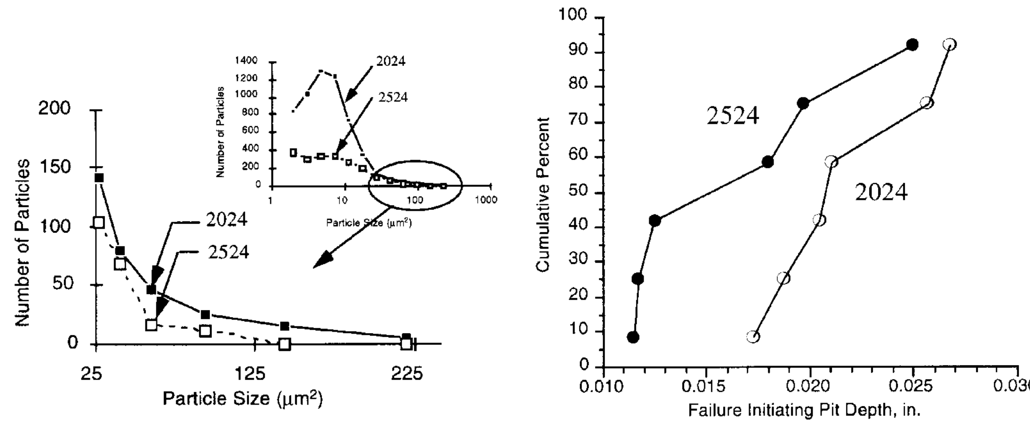


Fig. 6 Axial *S*/*N* fatigue performance of 2024-T3 and 2524-T3 bare sheet (0.125 in. thick) with and without prior corrosion.

second-phase constituent particles. These nucleated pits, in turn, serve as sites for potential initiation of fatigue cracks. Because alloy 2524 has a significantly lower constituent particle density as a result of its compositional and processing improvements, the fatigue strength of 2524 exceeds that of 2024 when both are tested in a precorroded state. Figure 6, derived from the experimental work of Bray et al.,[15] shows a 2524-T3 (bare) 13% fatigue strength improvement over 2024-T3 (bare) when both sets of specimens were tested in a precorroded condition simulating 1 year at seacoast. When tested in the uncorroded state both alloys yielded identical fatigue performances. The effect of fewer and smaller constituent particles on fatigue initiating corrosion pits is illustrated in Fig. 7.

For older aircraft the combined toughness and crack growth advantage of 2524 over 2024 provides an added safety margin to accommodate stress increases arising from change of mission, metal loss from corrosion or corrosion repair, or added concentrations of stress from patch repairs. The 2524 capability improvement also allows for an increase in inspection interval that translates to lower operating costs. Inspections are easier because larger crack sizes can be tolerated and longer critical crack lengths translate to an increase in safety. The calculated effect of skin alloy (2524 vs 2024) and operating stress on inspectable crack growth life is illustrated in Fig. 8 for a longitudinal fuselage skin crack under an intact frame. The substantial residual strength and cyclic life improvement of 2524 over 2024 for the MSD scenario has been experimentally verified with results from uncorroded MSD coupon tests (Fig. 9) and also from results of more limited testing on precorroded MSD coupons.[14] New airplane build programs are interested in the improved damage tolerance capabilities of 2524 because they provide the opportunity to save weight and/or reduce manufacturing costs, for example, to



Comparison of constituent particle distribution in 2024 and 2524 sheet (0.125 in. thk., L-ST plane)

Cumulative failure-initiating pit-depth* distributions in 2024 and 2524 sheet
*** measured in plane of specimen fracture**

Fig. 7 Effect of constituent particle distribution on fatigue-initiating pits.

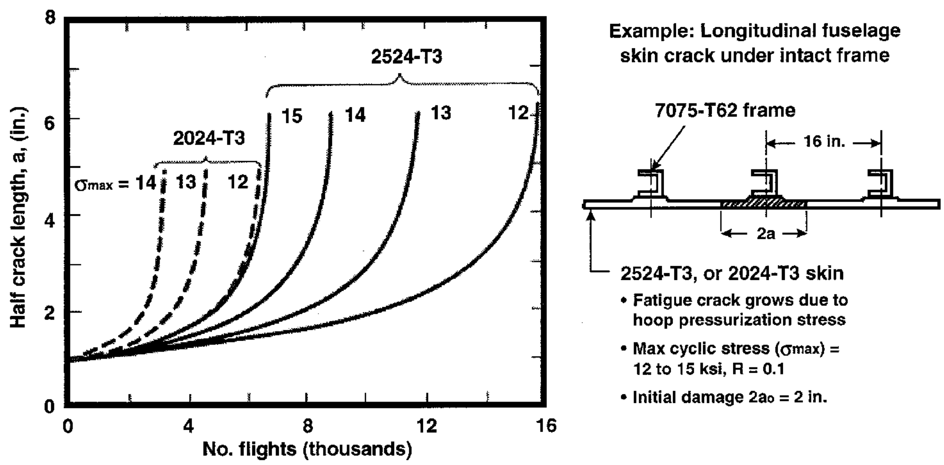


Fig. 8 Effect of skin alloy and operating stress on crack growth life.

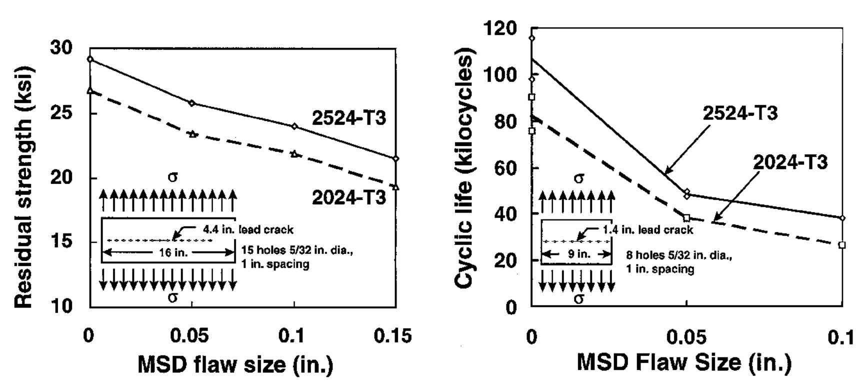


Fig. 9 Residual strength and cyclic life capabilities of 2524-T3 and 2024-T3 clad skin sheet (0.05 in. thick) from tested wide, multiholed panels with a central lead crack and varying size MSD cracks (two per hole).

eliminate some tear straps as was done for the Boeing 777. More recently, Airbus has specified 2524-T3 for the fuselage skin of their A340-500/600 aircraft, and Bombardier has likewise done the same for their new Global Express business jet.

Improved Stress Corrosion Cracking (SCC) Management in Integral Machined or Forged Parts

In the post-World War II era increasing numbers of SCC problems appeared with the introduction of high-strength 7xxx series aluminum alloys and growing use of integral components finish machined from thicker starting stock. Although historically SCC failures have almost never resulted in crashes or other catastrophic vehicle failures, SCC damaged parts often do entail significant economic loss due to cost of replacement and down time. Comprehensive surveys of failure incidents during the 1960s and 1970s,[29,30] revealed that the majority of aluminum structures documented as failing by SCC were manufactured from the high strength 7xxx and 2xxx series aluminum alloys that contain Al, Cu, Zn, and Mg. Of these, alloys 7075, 7079, and 7178 in the peak strength T6 condition and alloy 2024 in the naturally aged T3 condition contributed to more than 90% of the reported aluminum SCC failures.

Almost all SCC failures have been observed to be characteristically intergranular, making early visual detection difficult. The intergranular attacks were related to the compositional difference of grain boundaries that makes them electrochemically sacrificial (anodic) to the rest of the microstructure so that SCC propagates selectively along them. Thus, in addition to alloy and temper, susceptibility to SCC within a finished part is also influenced by intrinsic grain flow attributes of the host from which it is machined. For example, plate rolling operations elongate and flatten grains in the rolling plane, whereas die-forging operations produce elongated grains that tend to follow the billet contour, except at parting plane locations where grains may be flattened. In contrast to thin sheet where short transverse stress is seldom a problem, pocketed structure can experience significant short transverse stresses to drive SCC along the weakest grain boundary planes. Reliable SCC assessment in older systems is further confounded by unknown possibilities for short transverse stress addition from usage change or retrofit actions. The use of newer and more SCC resistant materials (e.g., 7050, 7150, and 7055, in an overaged T7X temper condition) is, therefore, recommended as the ideal fix for SCC problem prone parts. Form-fit functionality also needs to be considered, as, for example, in weighing the expediency advantage of converting a die-forged part to one machined from plate. In this example, the repair turn around advantage of machined plate should be weighed against the greater long-term SCC risk as contrasted to that of the original forging.[19]

In the aforementioned surveys of SCC, residual stresses from manufacturing operations were identified as primary contributors to many SCC failures. Moreover, little or no accounting for residual stress in the original part design process was usually found to be the norm. Residual stresses from heat treatment and fabrication were the most often cited stress drivers for SCC failure. In large part this was because most of the SCC failures experienced were large forgings where residual stresses have been a long historical problem. Recent developments in commercial quench and temper practices (e.g., T7452 or T7454) to improve forging machining performance (Fig. 10)[31,32] have the ability to yield substantial residual stress reductions in originally nonstress relieved parts. This technology is available to help mitigate historical SCC problems associated with nonstress relieved forgings. The process is applicable to retreatment of sick forgings, where die tools exist. Moreover, the process is alloy-independent allowing the same alloy to be specified, which helps to reduce the engineering burden associated with change.

New Generation Al–Li Alloys for Reduced Density and Improved Fatigue Crack Growth Resistance

The second generation of Al–Li alloys (the first being the Alcoa alloy 2020, and Russian alloys VAD 23 and 1420) were developed in the late 1970s and early 1980s (alloys 2090, 2091, and 8090). The Al–Mg–Li alloy 1420 and the Al–Li–Cu–X alloys 2090 and 8090 are now in service in the MIG 29 and the EH 101 helicopter and were used in the C17 transport. Alloy 1420 has only moderate strength and the Al–Li–Cu alloys (which contain approximately 2% lithium) exhibit a number of technical issues that include anisotropy of mechanical properties, crack deviations, a low stress–corrosion threshold stress in the short transverse direction, and less than desirable short transverse ductility and fracture toughness. Newer Al–Li alloys have been developed with lower lithium concentrations than 8090, 2090, and 2091. These alloys do not appear to suffer from the same technical issues. The first of the newer generation was Weldalite 049® (2094) that can attain a yield strength as high as 700 MPa (101 ksi) and an associated longitudinal elongation of 10%. A refinement of the original alloy, 2195, which has a lower copper content, is now being used for the U.S. Space Shuttle superlightweight tank. Alloy 2195 replaced 2219 and, along with a new structural design, saved 7,500 lb on the 60,000-lb tank. This allows an increased payload for the Shuttle and reduces the number of flights necessary for the construction of the International Space Station, thus saving millions of dollars.

Three other recent derivatives of the third generation of Al–Li alloys are 2096, 2097, and 2197. They contain lower copper than 2024 and slightly higher lithium content compared to 2195. Alloys 2097 and 2197 contain a very low Mg content to improve SCC resistance and Mn to prevent strain localization normally associated with the shearable Al_3Li present in the higher Li-containing alloys. Alloy 2097/2197 was recently selected[2] for replacing 2124, which had fatigue problems, for bulkheads on the F16 fighter. Alloy 2097 has a 5% density advantage over 2124 and at least three times better spectrum fatigue behavior or approximately 15% higher spectrum fatigue stress allowable. Although Al–Li alloys are more expensive than conventional aluminum alloys, the replacement of 2124 × 2097 for the BL 19 Longeron of the F16 doubles the service life of the part, saving over \$21 × 10^6 for the fleet of 850 U.S. Air Force aircraft.[2] Engine access cover stiffeners, currently made from 2124, are also being replaced by Al–Li alloys due to their better fatigue life. This is an excellent example of retrofitting with a new, higher performing material to reduce life-cycle costs as described by Austin et al.[2]

Aluminum–Beryllium Alloys

A significant proportion of fighter airframe design is stiffness driven. Thus, there are significant opportunities for a new metallic material providing high specific stiffness relative to currently available materials. For example, aluminum beryllium alloys may provide an affordable sheet metal alternative to resin matrix composites

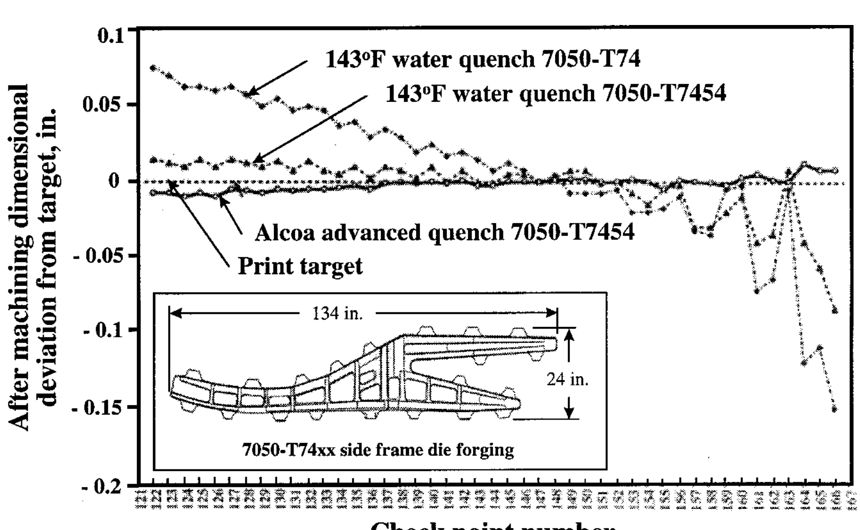


Fig. 10 Al 7050-T74xx die-forging machining distortion study showing significant residual stress reduction attainable via coupling new Alcoa quench practice and compression stress relief.

Table 2 Possible scenarios for exploiting new materials benefit potential

Repair option	Primary requirement	Potential benefits	Potential disadvantages	Time/risk resources
Identical component/ material replacement	Maintain safety and get it flying	Straightforward	Prolongs the agony with high repeat repair costs	Lowest
Form-fit function (material upgrade)	Reduce maintenance cost, improve readiness	Some capture of new materials benefits	Requires materials and process, design, and analysis expertise	Moderate
Reoptimize with material upgrade	All of the preceding plus performance	Some capture of new materials benefits	Requires extensive materials and process, design, fabrication, and analysis expertise	Moderate to high
Total redesign with new concept	Maximize life-cycle economics and performance	Full capture of best available technology	Requires full original equipment manufacturer capabilities	Highest

in stiffness critical airframe structure.[33] Aluminum beryllium alloys are currently used for secondary structures such as equipment shelves and support structure due to low density, high stiffness, and good vibrational damping characteristics. Application of these materials in primary structure requires the establishment of a database for design and manufacturing, safe manufacturing practices that avoid exposure to beryllium dust and particulates, and corrosion protection methods for components in the field. Also, appropriate safety protocols to control exposure to beryllium in field maintenance must be applied. Active topics for research include alloy development for improved strength, corrosion and corrosion protection, and processing of both wrought and cast products. A few scenarios for exploiting the potential benefits of new material replacements are given in Table 2.

Conclusions

Older aircraft can be retrofitted with new durability and damage-tolerance improved materials to achieve life extension and sustainment cost reduction goals. Continuous improved and derivative variants of existing alloys have the broadest utilization potential. Many of these materials are already flying on new aircraft (e.g., the Boeing 777, Airbus 340-500/600, and C-17) and/or have been used or are being contemplated for retrofitting older aircraft, for example, the KC-135, P-3, C-130, and F-16. Some alloys may be considered as preferred equivalents to their predecessors regardless of application (e.g., 2524 for 2024), and others may be considered preferred replacements within limits, for example, 7xxx-T7x for 7075-T6.

To facilitate the retrofitting of aging aircraft with new materials, a generic material substitution system is needed for rapid/broad implementation of the best material solutions. This system should include ways to improve the efficiency of the substitution process by substantiating new materials as preferred replacements, by approving the alloy substitution matrix, and by defining opportunities and cost/benefit trades for replacement scenarios. In addition, the repair and maintenance centers should stock qualified substitutes to reduce down time for retrofitting.

Acknowledgment

E. A. Starke Jr. would like to acknowledge support of U.S. Air Force Office of Scientific Research under AFOSR Grant F49620-97-1-0234, Spenser Wu, Grant Monitor.

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