

## Proposed Through-Life Management Approaches for Bonded Repair of Primary Structure

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### **ABSTRACT**

*The inability of current non-destructive inspection (NDI) procedures to confirm bond integrity has greatly limited the application of bonded repairs to primary structure, especially in applications where failure of the repair would lead to safety in flight concerns. In view of these concerns applications to primary structure are generally limited to situations where the residual strength of the parent structure in the absence of the repairs can exceed the design limit load by an acceptable factor, most conservatively as high as 1.5, which is the design ultimate strength.*

*This paper proposes technologies and associated strategies that could lead to some relaxation of the current residual strength requirements. This makes possible the wider application of bonded repairs to primary metallic structure suffering fatigue cracks or composite airframe structure suffering visible impact damage.*

*The detection of weak, and even absent (“Kissing”), adhesive bonds generally requires the application of a significant stress to the region of the adhesive bond which cannot be achieved by conventional NDI – ultrasonic techniques for example. Since it is generally not feasible to stress the actual repair patch, a “Proof Test” has been developed. This test requires the application of shear stress to a bonded repair coupon (BRC) made of the patch material. The BRC is bonded to the parent structure concurrently with the patch. This test can confirm both the initial and through-life structural integrity of the repair bond. However, it must be agreed by the appropriate authority that the BRC is fully representative of the patch system.*

*In addition to the proof test more critical repairs also require through-life structural health monitoring (SHM) with a focus on detecting patch disbonding and a secondary focus in the case of repairs to metals of monitoring crack growth. It is concluded that to detect patch disbonding, a simple approach using resistance strain gauges, or more robustly optical fibre sensors, holds the most promise, at least in the short term. Disbonding is detected from measurements of reductions in strain transfer from the parent structure into the patch, an approach previously demonstrated during full-scale fatigue testing of a repaired F111*

wing.

*Based on use of the Proof Test together with SHM, as appropriate, decision charts are presented for the management of bonded repairs to fatigue cracks in metallic structure in which the crack is not removed and to composite structure following removal of visual impact damage. A simple approach to the management of stress-reducing reinforcements for metallic components is also presented.*

## **1.0 INTRODUCTION**

Modifications to airframe components are often required to repair structural damage or reduce stress to increase strength or extend service life. This is generally achieved by attaching patches over the damaged region or applying reinforcements in regions where damage may be anticipated - the patches or reinforcements may be metallic or fibre composite.

For metallic airframe components, typically aluminium alloys, fatigue cracking is the main service damage considered in this paper. Whilst for composites, typically carbon/epoxy, it is visible damage caused by severe impact.

Using structural adhesive bonding to attach repair patches or reinforcements has many advantages over conventional fastening procedures using bolts or rivets [1]. For example, reinforcing efficiency (for example reduction in stress intensity, in the case of metals) is much greater and the damage associated with extra fastener holes is avoided.

An important advantage of using adhesively bonded patch repairs for metallic components is that existing fatigue cracks need not be removed, providing that patch system (patch and adhesive bonding) structural integrity can be assured.

High-performance composites, especially boron/epoxy and carbon/epoxy, have many advantages over metals for use as patches and reinforcements [1]. The advantages include high strength and stiffness, fatigue and corrosion resistance, and formability. In addition, composites can be customized to precisely match stress and stiffness requirements. The low electrical conductivity of boron/epoxy is an important advantage for its use as a reinforcement or patch for metals because it avoids galvanic corrosion problems with the parent structure. It also enables the use of eddy-current non-destructive inspection (NDI) to detect cracking in the parent structure. However, the main disadvantage is the development of significant residual stresses which arise from the difference in the thermal expansion coefficients between the composite patch and parent material.

The advantages of patch attachment by structural adhesive bonding are, however, significantly offset by the inability of conventional NDI to detect inadequate bond integrity, although physical flaws such as significant voids, delaminations and disbonds can readily be detected. This is in contrast with mechanical fastening where a physical connection (even if defective) is immediately obvious.

Currently, conventional NDI techniques cannot reliably detect weak bonds, including those that become degraded during service. Even so called “Kissing Bonds”, in which there is intimate contact between the adhesive and parent surface but no actual bond [3] or where a liquid which has entered a disbanded region by capillary action acting as a coupling agent for sound waves cannot be detected easily.

Significant research is currently focussed on NDI detection of “Kissing Bonds” so a capability may eventuate; however, even so the detection of weak bonds is even more challenging.

It is reasonable to assume that when a bond is weak from the outset it is most likely the result of processing errors such as surface contamination, inadequate surface preparation, or incomplete cure, or the use of severely out-of-life adhesive. A bond that becomes weak while in service is generally the result of degradation of the adherend surface, and in the case of metals, it is usually the result of hydration.

For bonded repairs applied under strictly controlled conditions, it is generally sufficient to base qualification solely on high-level quality control of pre-bond surface treatment and adhesive cure conditions, following

satisfactory NDI confirmation of the absence of unacceptable defects. However, in most repair situations it may be infeasible to ensure the required level of quality control.

The inability to validate the structural integrity of adhesive bonds by NDI currently limits the application of repairs to primary structure to situations where the damaged component retains acceptable residual strength in the absence of the repair. This requirement is generally well above the design limit load and as high as design ultimate load, as described in the next section.

This paper presents an alternative non-destructive test procedure to assess bond structural integrity and proposes approaches for through-life management of bonded repairs and reinforcements. The objective is to facilitate wider use of bonded repair and reinforcement, including primary airframe structure, based in part on this test procedure, and when appropriate Structural Health Monitoring (SHM).

The primary focus of this paper is the through-life management of repairs and reinforcement of metallic components since this application is the most challenging. Approaches for managing composite structures are also presented. While the mechanical behaviour of composites under service conditions is quite different from metals, the need to ensure structural integrity of the patch system is the same.

To provide a background for these proposals a brief overview of the airframe design philosophies for metallic and composite airframe components is provided in the next section.

## 2.0 AIRFRAME DESIGN PHILOSOPHIES AND THEIR IMPLICATIONS FOR BONDED REPAIRS

To ensure safety in flight the basic requirement is for the airframe structure to retain a specified level of residual strength (RS), typically design ultimate strength, under all service conditions. Repair or reinforcement is required when the structure cannot meet this requirement.

The certification basis for aircraft structures is long standing and is in part based on three main design approaches, described shortly. These were originally developed for metallic structure and so are largely driven by the susceptibility of metals to fatigue cracking under predominantly tensile dominated spectrum loading. However, in composites the primary concern is with static compression strength following impact damage; damage growth, or significant further RS reduction is not generally an issue under service spectrum loading within the design range.

The initiation and growth of delaminations and disbonds under cyclic loading is a significant fatigue issue for laminated composites, and for bonded joints, for example, from unanticipated through-thickness stresses. However, this is not considered here except in relation to bonded repairs.

Two design levels of strength are generally specified: Design Ultimate Load (DUL) and Design Limit Load (DLL). DLL is frequently defined as the load which can be expected just once in the life of the airframe and is generally given by  $DUL/1.5$  - see Figure 1. In this paper the equivalent terms Design Ultimate Stress (DUS) and Design Limit Stress (DLS) are also used interchangeably with DUL and DLL.

Regarding fatigue considerations one of the following alternative approaches may be taken:

**Safe- Life Approach:** No significant cracking should occur in the life of the airframe that could lead to failure. This approach was used in the design of most of the metallic structure in older fighter aircraft, and is still used for US Navy fighter aircraft, such as the F-18 and generally in helicopters.

### Fail-Safe Approaches:

**A. Alternate Load Path:** The structure is damage tolerant in that cracking may occur but will not reduce strength below acceptable level before being detected. This requirement is generally achieved by multi-load-path design such that should one load path fail, the remaining load paths can continue to provide the required level of residual strength until the damage is detected. This approach is generally used in the structure of large military transport and civil aircraft.

**B. Slow Crack Growth Approach:** The structure is damage tolerant in that cracking may occur but cracks will grow slowly and will not cause failure for the full life of the structure or prior to detection by planned inspection (safety by inspection). This approach can be applied to single load-path structure, where failure could be catastrophic. Damage tolerant design for a single-load-path structure is based on the assumed presence of flaws at fatigue critical locations.

## **2.1 Implications of Fatigue Crack Repairs in Metallic Structures**

The repair of an alternate load path structure is the least critical due to the existence of alternate load paths in the event of patch failure. If cracking occurs in a safe-life structure and the cracks cannot be removed, then the incident can be managed as prescribed for a damage-tolerant structure, as described below.

The recommended approach for managing a damage-tolerant metallic structure is illustrated in Figure 1, which is based on a diagram in Reference [2]. The upper section of Figure 1 shows the reduction in RS caused by the growth of fatigue cracks under service spectrum stresses (shown schematically) as a function of time. The schematic of spectrum loading depicts stresses up to DLL (DLS). The lower section of Figure 1 shows the inspection capability as a function of damage (crack) size.

The inspection interval  $F_{NDI}$  is based on estimated crack growth behaviour from an assumed crack just below the limit of detectability. The NDI interval is such that cracks can be detected and repaired (generally firstly by removal) well before RS falls to the fail-safe requirement (typically  $1.2 \times DLL$ ). Restoration is generally achieved by crack removal followed by reinforcement, traditionally using a bolted or riveted patch. This patch should restore the residual strength to DUL. It is assumed in Figure 1 that the crack growth rate after repair is similar to the growth rate prior to repair.

The ideal requirement is for the unrepaired structure to have an RS above the fail-safe strength before repair, as shown in Figure 1. Restoration to an RS above the design ultimate stress (DUS) is an ideal result which should also be met for a bonded repair, even when the crack is not removed. Then, if structural integrity of the adhesive bond can be assured, the subsequent inspection interval can be based on a patched crack. However, if structural integrity of the patch cannot be assured, then crack growth estimations are based on the unpatched crack. This is discussed in more detail in Section 6.

However, the RS requirements may not be achievable and engineering decisions are required to determine an acceptable alternative. For example, it may be acceptable to use a bonded repair to recover RS to the fail-safe strength, providing the RS is not below DLS in the absence of the repair.

In a more extreme case, a crack may not be detected before RS is below DLS, as in the case [4] of a repair to the wing of an F111 fighter aircraft. Here, safety in flight depends on the structural integrity of the repair system. These issues, and a proposed approach for their management, are described in Section 6.

Through-life management requirements differ depending on the type of structure. Repairs to a multi-load path structure are the least demanding. The efficacy of bonded composite repairs to a multi-load path structure was demonstrated in an extensive repair program [5] conducted by the USAF on its fleet of Lockheed C141 aircraft.

The wings of the C141 aircraft experienced extensive fatigue cracking towards the end of the life of the aircraft fleet. The cracks had initiated at weep holes in the integral risers. Smaller cracks were removed by reaming the holes, but many cracks were too large for this treatment, and in some cases the cracking had extended into the wing plank so could not be removed. To address this problem an extensive bonded repair program was completed using boron/epoxy patches bonded with a structural film adhesive. More than 2000 individual patches were applied to 770 weep holes. In most cases, Australian patching technology was used.

The C141 fleet was retired after ten years of service following the repairs. Retired aircraft were then selected for teardown [5] to determine the effectiveness of the repairs, and to locate evidence of degradation or disbonding of the patch systems. The USAF and Lockheed experts concluded, conditionally, that bonded composite repairs could be certified for use with a multi-load path structure.

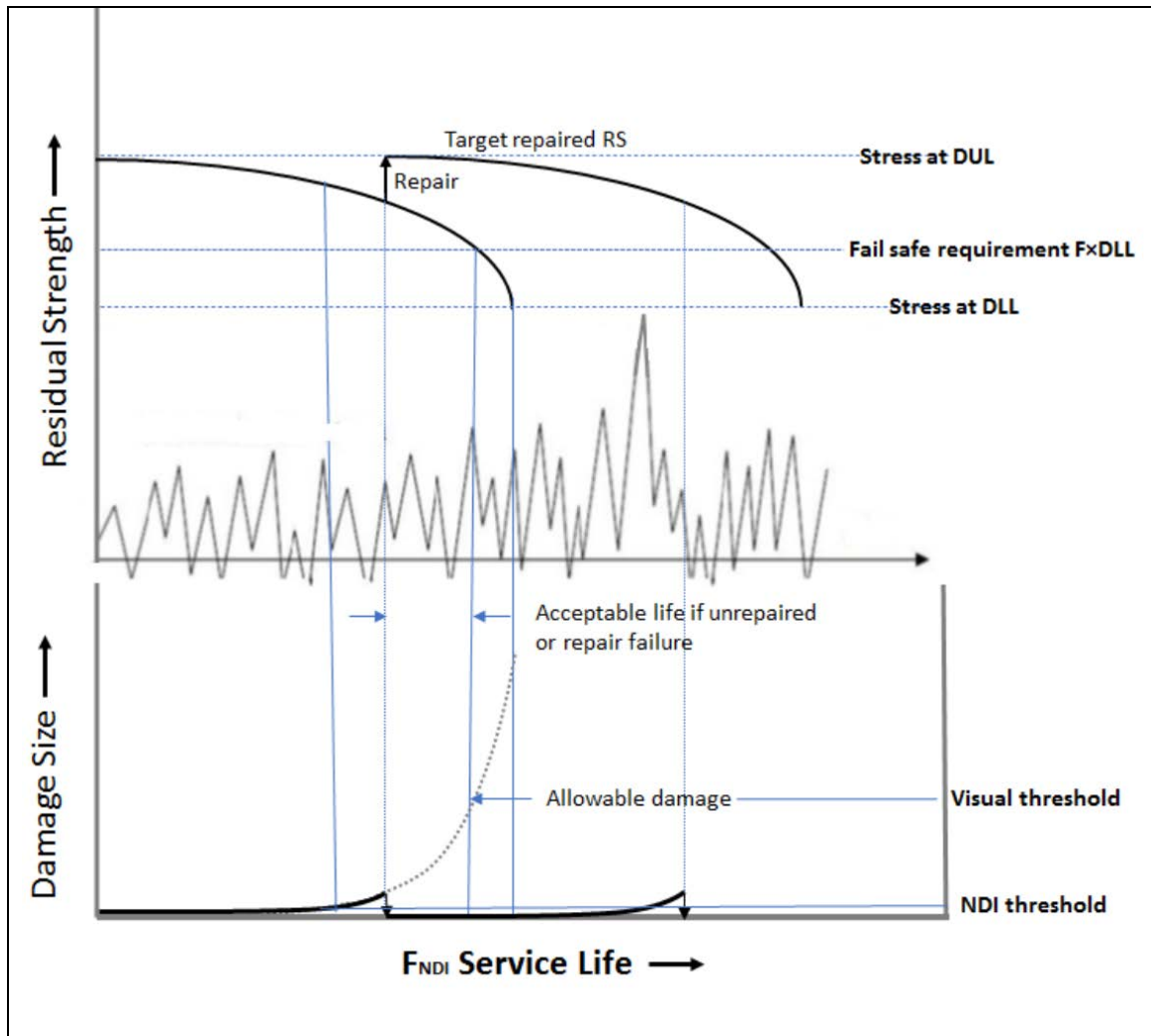


Figure 1: Residual strength and repair requirements for a damage tolerant metallic structure.

The through-life management of bonded repairs to cracks in a damage-tolerant, single-load path structure is a considerably greater challenge than through-life management of cracks in a multi-load path structure. Section 6 describes a proposed approach for through-life management of this more challenging case.

The through-life extension of a safe-life structure may be achieved using a bonded reinforcement to reduce stress resulting in an extension of the crack initiation period. This may be achieved following a “confidence cut”, in which undetected small cracks are removed by surface machining [6]. Through-life management is based on a retrospective safety-by-inspection approach for the reinforcement, as proposed in Section 7. However, if cracking is significant it may be necessary to manage as for a damage tolerant structure, as described in Section 6.

## 2.2 Implications for Repairs to Damage in Composite Components

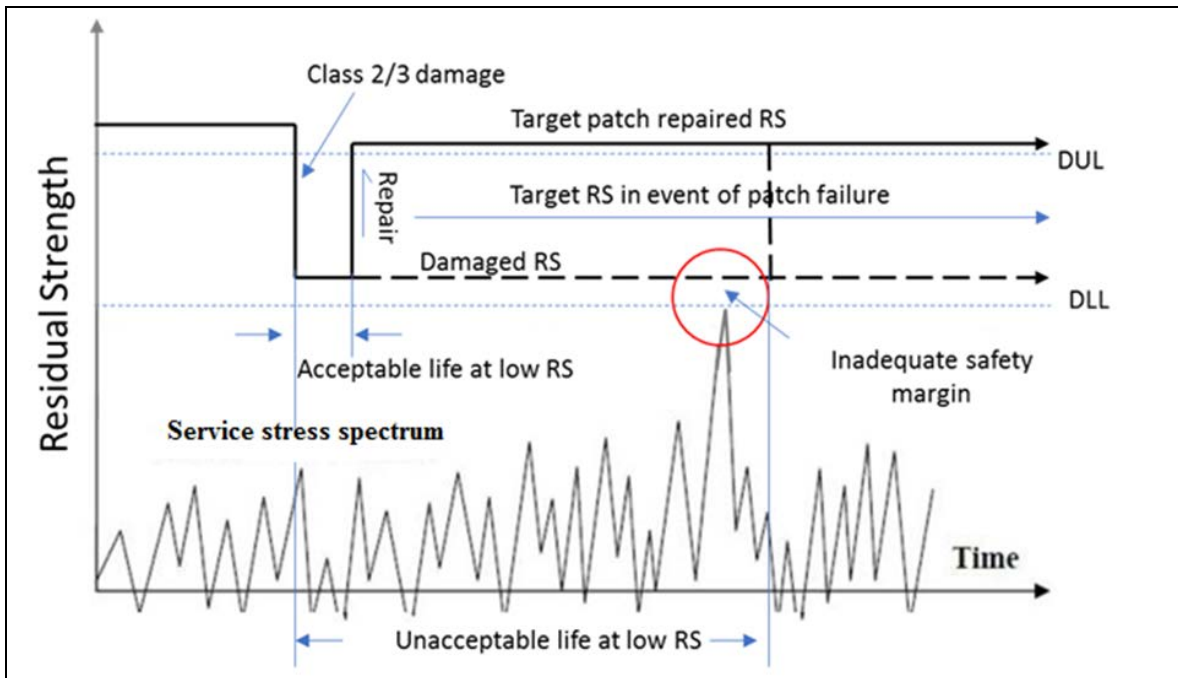
Impact damage to composite structures, and the effect on RS are classified as follows [7]:

- Class 1 damage: Barely Visible Inspection Damage (BVID). This type of damage causes a reduction in RS. However, based on experimental evidence, it is assumed there is no further growth under service loading within the design stress range. Allowance is made for RS reduction from BVID in the development of design allowables.



- Class 2 damage: Visible Inspection Damage (VID). This damage results in a reduction in RS below DUS, but above DLS. Assuming no significant further RS reduction under service loading (as appears to be the case), repair action can be delayed for a short period, pending inspections and other safety considerations.
- Class 3 damage: is more severe VID, and frequently results in full penetration of the composite. Since it potentially reduces RS close to (or possibly below) DLS repairs are required as soon as feasible after visual detection.
- Any more serious damage requires instant repair, as RS will be below DLS.

Based on the no-growth assumption, the repair requirements are illustrated in Figure 2, which is the composites version of Figure 1.



**Figure 2: The influence of Visual Damage on Residual Strength, and the ideal repair requirements for the repair of a composite structure.**

Figure 2, is based on Reference [8], again including a schematic of spectrum loading, and shows a case in which Class 2 and 3 damage results in an instantaneous drop in RS to near DLS. This damage generally does not grow or result in further reductions of RS for a short period (e.g. a flight to depo facilities) under normal service loading. However, if the damage is not repaired for a prolonged period, safety in flight is compromised. This is shown in Figure 2, as “Inadequate safety margin” because, on a statistical basis, stress in the order of DLS can be expected several times in the life of the airframe, especially if usage is more severe than originally planned, which is frequently the case.

The repair of visible damage to composites usually requires removal of the damaged region and the application of a reinforcing patch. The patch (composite or metallic) can be mechanically fastened or adhesively bonded. For thick section repairs [9], a bonded scarf or stepped configuration is used.

As indicated in Figure 2, the ideal repair requirements are recovery to the design ultimate stress (DUS), and the RS above DLS (typically  $1.2 \times DLS$ ) is maintained prior to repair and in the event of failure of the repair. These are the same requirements as for crack repairs in metallic structures.

However, to allow for possible patch failure the damaged region can be reshaped during removal and prior to

repair, so a target strength above DLS is achieved. In a severe case this approach may not be feasible if the damage has reduced RS significantly below DLS. In this case, reliance is placed on the structural integrity of the repair system.

As with cracks in metals, a possibility for composites is to leave the damaged material in place, for example, in the case of a Class 2 damage zone. This is potentially an attractive option for repairs to thick section composites because preparation for the application of a scarf or stepped repair involves extensive removal of parent material. However, unlike the properties of cracks in metals, where fracture mechanics can be used to make reliable predictions, the ability to estimate RS in damaged composites is, at present, very limited and unlikely to be acceptable.

### 3.0 MONITORING BOND INTEGRITY

There is considerable incentive for using a bonded repair as an alternative to a mechanically fastened repair, provided the bonded repair can be certified. This is especially true for repairs to metallic airframe structures when it is not feasible to remove the fatigue crack.

The key requirement for certification of this application is the assurance of the initial and through-life structural integrity of the patch system, and specifically, the adhesive bond. Unfortunately, standard NDI techniques cannot detect weak bonds, primarily because they are unable to adequately stress the bond interface and must rely on second order changes, such as ultrasound reflection, or attenuation.

An alternative approach is to conduct periodic proof tests which can adequately stress the adhesive bond. This can be used to demonstrate that the adhesive bond has achieved an acceptable level of strength initially and ensure that an acceptable level of strength is maintained during the service-life of the repair.

Another and potentially complementary alternative is to attach sensors to the structure and monitor the repaired region throughout its life, to detect early indications of a loss of bond integrity, and possibly damage growth in the parent structure. This is the structural health monitoring approach (SHM). However, SHM and conventional NDI have similar limitations because they can only detect physical damage but not weak bonds including bond degradation. The advantage of SHM is the ability to detect disbonding and crack growth as it occurs, so it can function as an early warning system. However, it cannot ensure initial bond integrity.

These two approaches, proof testing and SHM, are described in the following sections.

### 4.0 PROOF TESTING

Proof testing is generally applied directly to the component being evaluated either as an alternative to NDI where NDI is unable to detect significant defects or in addition to NDI when the probability of detection is low and the consequences of failure are unacceptable. The cold proof test applied periodically on the wing structure of F111 is an example [10]. It is of critical importance that the proof test interrogates the structure to a sufficiently high stress to be meaningful without causing significant damage.

In the proof-loading approach for adhesively bonded repairs the adhesive bond must be subjected to relatively high levels of stress, both immediately after patch application and, ideally, periodically throughout service life. Laser shock [11] is one approach to achieving this and direct mechanical loading of the patch is another. However, laser shock may be too hazardous and potentially damaging to be used in repair applications [12] whilst direct mechanical loading of the repair is infeasible in most cases.

#### 4.1 Proof Testing of a Bonded Repair Coupon

A practical and cost-effective option is the periodic proof test of bonded repair coupons (BRCs). The BRCs are made of the same material as the repair patch, and are bonded concurrently with the patch, and therefore, as far as possible, under identical conditions to the repair patch. This test is based on the assumption that the bond strength of the BRCs represents the bond strength of the patch. An example is shown in Figure 3. On

this basis, the recorded strength of each periodic (destructive) BRC test represents the strength of the repair patch.

The BRCs must be very thin to minimise local change in the parent material surface profile, and to match stress conditions experienced at the ends of the patch. The BRCs are sensitive to both the initial and ongoing strength of the repair bond-line between the repair patch and the substrate material for the in-service environmental and stress conditions predicted for the patch.

In the approach described in Reference [13], the BRCs are bonded to the parent structure over a narrow annulus and subjected to shear loading that is applied manually using a (digitised) torque wrench through an adaptor. The adaptor is bonded to the BRC using a weaker adhesive and is subsequently removed by modest heating.

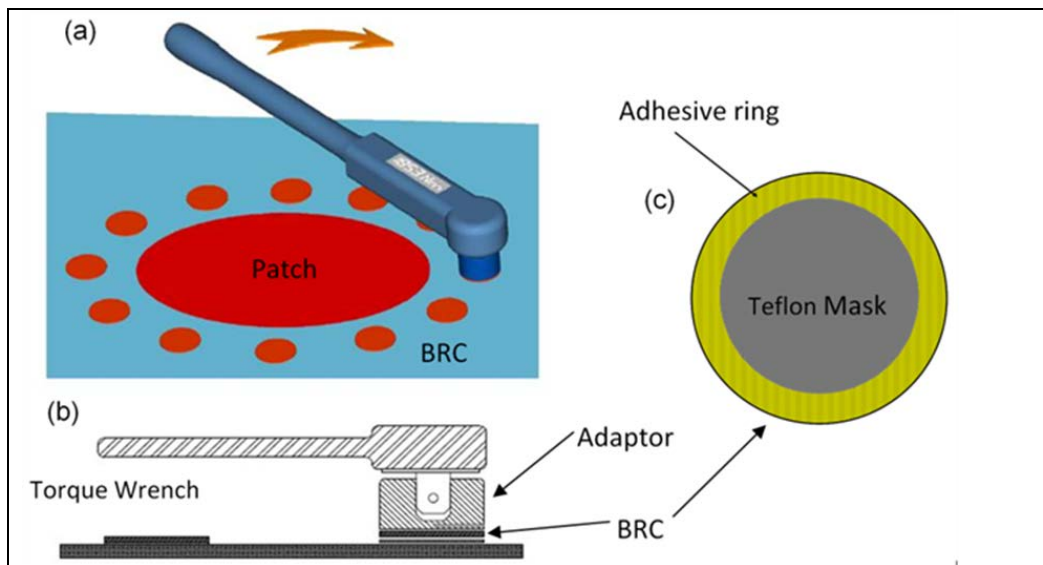
Figure 3 provides a schematic illustration of the test configuration. Mechanically, this approach is similar to a standard (ASTM E229) test, the Napkin ring test, which involves applying torsion to end bonded metallic tubes. The ASTM test is primarily used to measure shear modulus and shear yield stress and the strength of the adhesive film. The BRC proof test may be considered as a complimentary method to the Boeing wedge test (ASTM D3762) and PATTI® testing (Pneumatic Adhesion Tensile Testing Instrument) to interrogate and ensure bond quality. The Boeing wedge test, a very effective method to assess bond durability, has been widely used to qualify a bond repair process as well as to qualify a repair technician's competence. The use of a wedge-test approach [14] was proposed some time ago for assessing the ongoing strength and durability of adhesively bonded aluminium alloy structures. It was proposed that several small thin strips of the aluminium alloy should be bonded to the structure at the same time as the main adhesive bond is formed. After bonding some strips would be pulled off manually at periods through the life of the repair. The mode of failure (adhesive or cohesive) would be used to assess the integrity of the adhesive bond. This rather crude approach whilst probably quite effective was never adopted.

The PATTI test can be used directly to measure the bond strength of a repair patch. However, it is usually used as a destructive test, since it involves cutting into the patch - so is generally used on a test article or on a repaired aircraft component that is no longer in service (see Section 4.2.3 for an example). Whilst the PATTI test could also be used for proof loading of the BRCs it applies a through-thickness tensile stress, whereas shear is generally more relevant in bonded patch repair applications and importantly, is much less damaging to a composite parent structure.

The purpose of the BRC proof test is to establish a confidence limit for the lower bound proof load for standard (non-degraded) patch/adhesive/parent structure combinations. These values are determined by laboratory tests to provide a statistically sound basis to assess the degree of degradation under various service conditions.

The sensitivity of the proof test described in the following sections is illustrated in studies [13] conducted on carbon/epoxy BRCs bonded to a carbon/epoxy laminate using the adhesive FM 300-2K. For full details see the original reference.

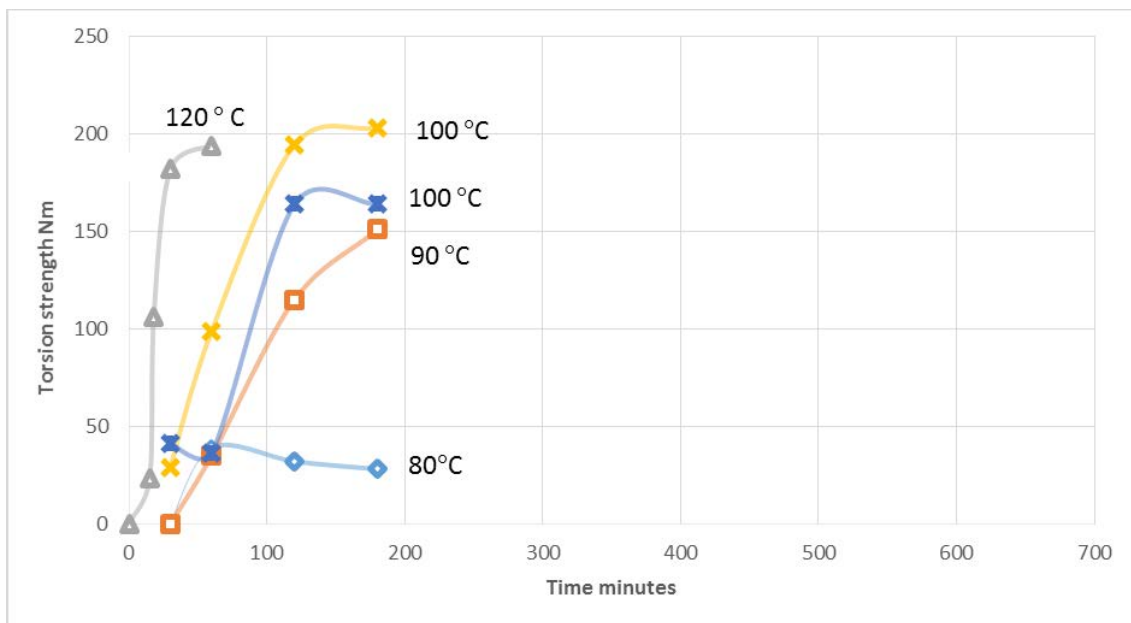




**Figure 3:** a) Repair patch and satellite BR C with one BR C under the torsion loading test, b) BR C with the steel adaptor in place for torsion loading using the torque wrench, and c) Adhesive Ring, created using a Teflon Mask attached to the BR C/adaptor configuration and applied to the surface of the parent structure.

### Cure Condition

The sensitivity of the test for cure conditions is provided in Figure 4, which shows torsion strength as a function of cure times for the adhesive Cytec FM300-2K, at the temperatures shown in the diagram. The results show that the test can readily detect an inadequate cure. As shown in Figure 5, the results for heat of reaction (an indication of epoxide consumption) of this adhesive [15] approximately follows the torque strength curves where the cure temperatures are similar. Conventional NDI is generally unable to detect the degree of cure except in extreme cases of under-cure.



**Figure 4:** Torque strength versus time-to-cure for adhesive FM300-2K at the temperatures shown. (Two batches of specimens were cured at 100 °C, the weaker was an old batch.)

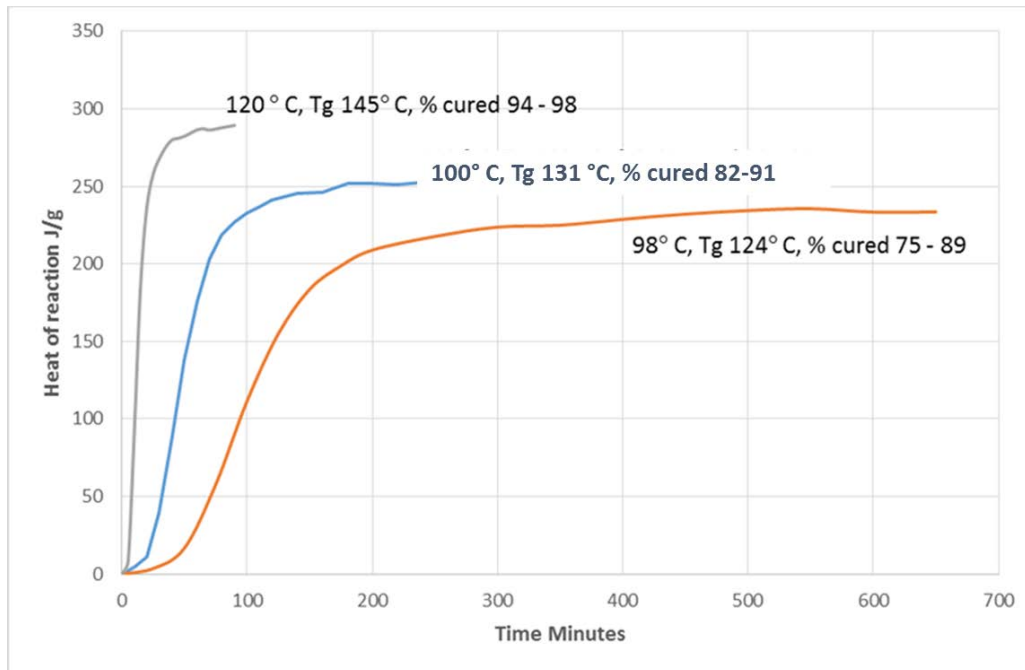


Figure 5: Heat of reaction at various temperatures and times, showing glass transition temperature, Tg, and % cure.

### Surface Contamination

The sensitivity of torque strength to surface condition is shown in Figure 6. This shows that torque strength with no surface treatment, and the torque strength with heavy contamination from hydraulic oil after surface treatment fall below the proof strength for the 95% and 99% confidence limits [13] (CL) for the torque strength of BRCs with standard surface treatment. Extreme contamination, such as contamination by a release agent, results in a kissing bond that is easily detected by the BRC proof test because torque strength is close to zero. However, even this extremely weak condition cannot be reliably detected by conventional NDI.

Mild contamination with hydraulic oil does not result in detectable degradation of bond strength. This is not a limitation of the proof test, but rather evidence that FM 300-2K can absorb the contamination and still provide a strong initial bond when the surface is lightly contaminated with hydraulic oil. If contamination did affect the long-term strength or durability of the repair, it would be detected by subsequent proof tests.

### Repeated Testing

If proof testing is used for the initial and ongoing in-service inspection it is important that BRCs are not weakened by repeated proof testing for the period that accommodates the expected number of proof tests during the lifetime of the repair. The repeated testing also includes the heating cycle used to remove the steel adaptor.

Figure 7 shows the results of repeated testing of several BRCs to the 95% proof torque of approximately 185 Nm, including heating to approximately 80°C following each loading. All BRCs survived 12 cycles of proof testing, and then two BRCs failed. The remaining BRCs were then tested to failure and appeared to be undamaged by the repeated testing, as shown in Figure 7. From this limited test it can be tentatively concluded that well bonded BRCs are undamaged for at least 10 cycles of repeated testing.

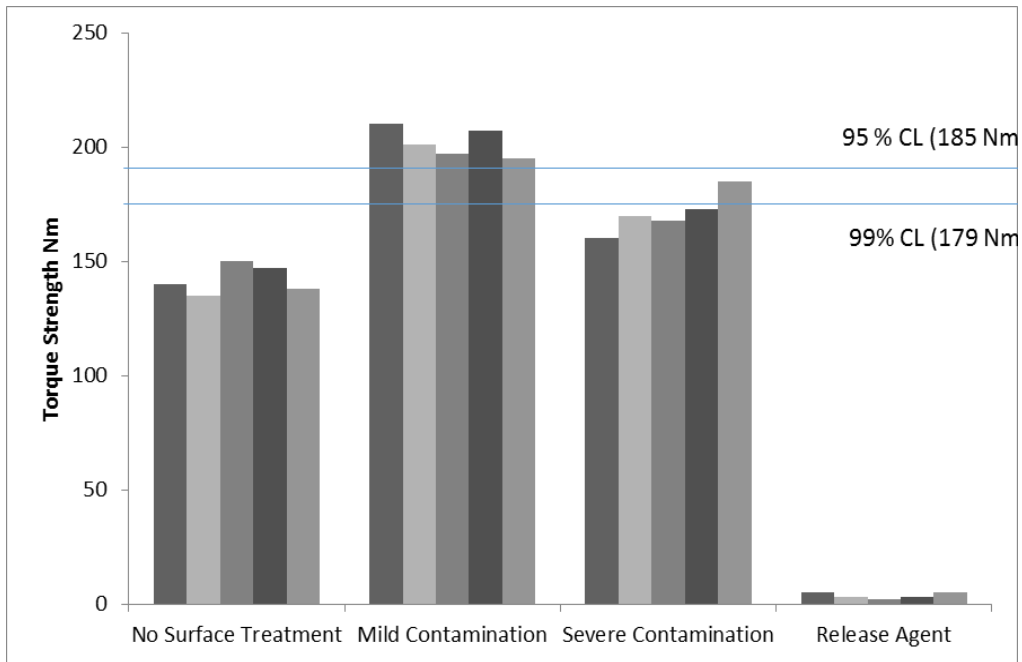


Figure 6: Torque strength for various degrees of contamination from hydraulic oil and release agent; the 95 and 99% confidence torque estimates are also shown. The various columns for each condition show repeat tests

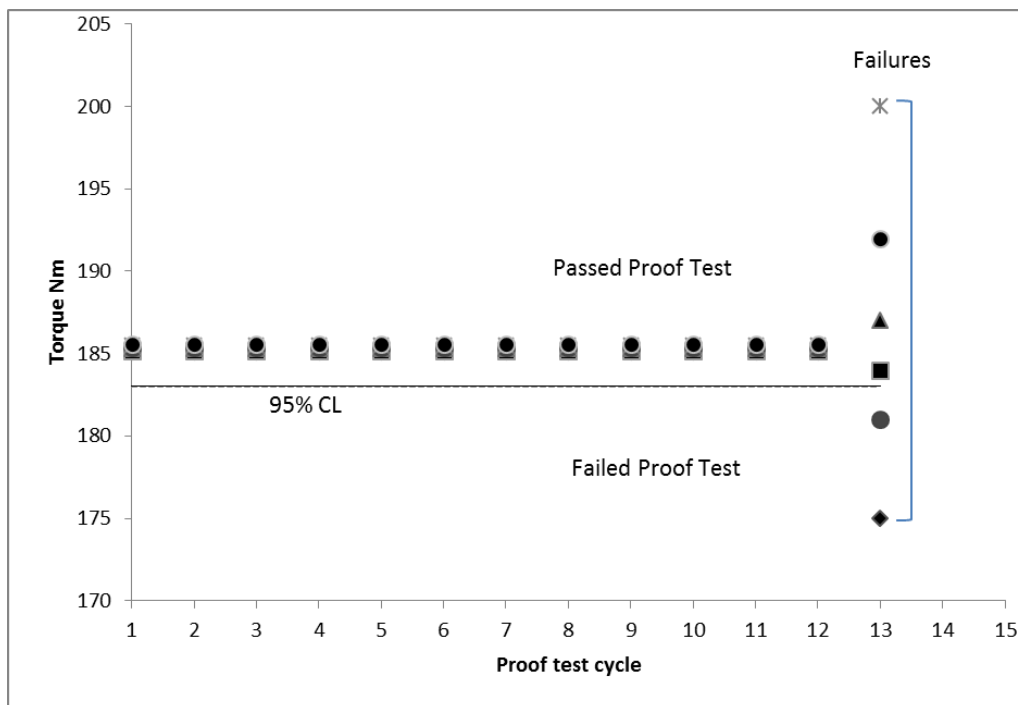


Figure 7: Torque value for carbon/epoxy BRCs versus proof test cycle that included a heating cycle; the only failures are those noted. The 95% CL proof torque for standard carbon/epoxy BRCs is shown as a horizontal line.

## 5 STRUCTURAL HEALTH MONITORING - SHM

### 5.2.1 SHM Requirements

The chosen SHM system must have a high reliability and probability of detection (POD), consistent with airworthiness standards. To be reliable the SHM system must be durable (fatigue resistant), and rugged enough to survive extremes in temperature, vibration and mechanical loading during aircraft operation; it must also have minimum impact on existing aircraft systems and, as far as possible, be stand-alone and autonomous – including self-powering. Ideally, sensors should be embedded in the patch. Surface-mounted sensors can be exposed to antagonists such as sunlight, moisture (and other aircraft fluids), erosion, and mechanical contact. Polymer matrix composite patches and reinforcements are conducive to the embedded approach because sensors (and other elements of the SHM system) can be incorporated during manufacture of the patch. The SHM sensors must withstand high strains and severe environmental conditions for prolonged periods; for example, they must be able to operate under temperature ranges from  $-50^{\circ}\text{C}$  to over  $100^{\circ}\text{C}$  and cyclic strains up to 3000 microstrain (strain  $\times 10^{-6}$ ) for more than  $10^5$  cycles. To allow for failures and uncertainties a considerable degree of sensor redundancy needs to be included in the SHM system to obtain the required level of reliability.

Some options for SHM of patch disbanding [16] are categorised in Figure 8.

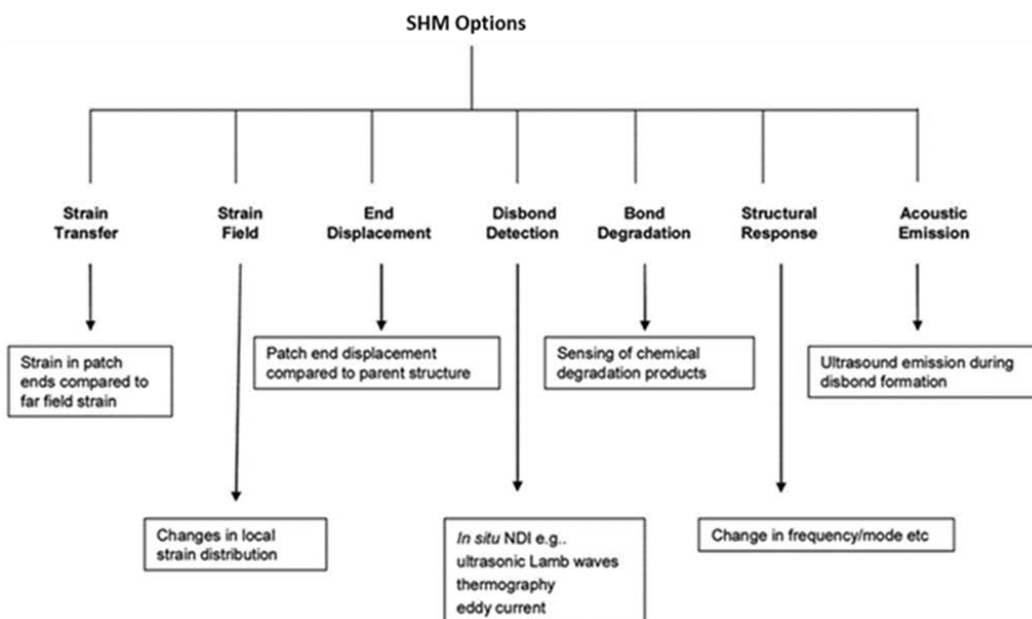


Figure 8: Some SHM options for detecting patch disbonding.

Crack detection can be achieved using commercially available SHM technology and includes: comparative vacuum monitoring; conformable eddy current sensors; and electrical resistance crack propagation gauges. Some of these techniques are potentially suitable for use with patch repairs and sensors can be embedded in the patch system. However, the sensors must be bonded to the parent structure under the patch and in the expected path of a crack. Therefore, this location becomes a potential disbond site.

Alternatively, crack growth can be assessed from strain changes in the patch itself. For example, Reference [17] describes the use of a fibre Bragg grating sensor to detect cracks from changes in the strain field in the patch associated with crack growth.

Of the various SHM options for detecting delaminations, the strain transfer approach is considered by the

authors to be most suitable for application, at least in the shorter term. Because of space limitations only this approach is discussed in the following section.

### 5.2.2 SHM Strain Transfer

The measurement of strain transfer into the tapered ends of an external patch, or reinforcement under service loading, can be used as a direct indicator of patch integrity. Any reduction in strain transfer is an indication of disbonding of the patch from the parent structure [16]. The basic in-service measurement is simply the synchronous ratio between the strain measured at the ends of the patch and a far-field strain in the same load path on the parent structure. It is important to note that this approach does not require knowledge of the service loads.

The strain-transfer technique requires the use of fatigue-durable strain gauges bonded to the patch, or ideally, embedded in the patch and on the parent structure. Strain-measuring options include conventional electrical-resistance strain gauges, piezoelectric-film strain gauge, and optical fibres using either Bragg gratings [16] or distributed sensing with Brillouin or Rayleigh scattering [18]. Fibre Bragg grating sensors are well established and have many advantages for strain measurement, including: good static strength and fatigue resistance; high sensitivity; low power requirements; immunity to electromagnetic interference; and a capacity for multiplexed or distributed measurement, which in the case of Rayleigh scattering can achieve spatial resolutions in the order of 1 mm at a low unit sensor cost. The small diameter of an optical fibre facilitates unobtrusive embedment in composite laminates. However, as with resistance strain gauges, the egress points are highly vulnerable to fatigue and mechanical damage unless well designed and protected.

### 5.2.3 Demonstration of the SHM Strain-Transfer Approach - F111 Wing Repair

Reference [19] describes an evaluation of a strain-based SHM approach on a critical boron/epoxy patch repair to a fatigue-cracked F-111C wing during full-scale fatigue testing.

Figure 9 shows details of the fatigue-cracked region. The cracking is the result of secondary bending in a region over a fuel vent gap in the spar land. When the crack was repaired with a boron/epoxy patch no further crack growth was detected in service for 670 flying hours, followed by approximately 9000 simulated flight hours during the full-scale fatigue test.

The following summary is provided to illustrate the viability of the SHM approach. Figure 10 shows details of the repaired wing and the patch region, and it also shows the patch and strip gauges used for SHM.

Strain measurements were taken during a full-scale laboratory fatigue test on the F111 wing. The test was conducted after the repaired wing had completed more than 3 years of service. Because loading during the test was highly repeatable, direct strain readings were used rather than the strain ratio described in the previous section.

The two graphs in Figure 11 show strain measurements made during the fatigue test. Each block corresponds to 500 simulated flight hours. The circles drawn on the patch show the regions where pull-off tension tests were made at the end of the fatigue test to measure the residual bond strength. The numbers in the circles show the nominal measured bond strength in MPa.

The SHM results correlated reasonably well with the ultrasonic NDI indications.

It is important to note that the plots for strip gauges on the left side of the patch show no strain reduction, indicating there was no disbonding. However, the plots for the right side of the patch clearly show significant strain reduction and correspond with the indications of degradation and disbonding noted on the right side of the patch.



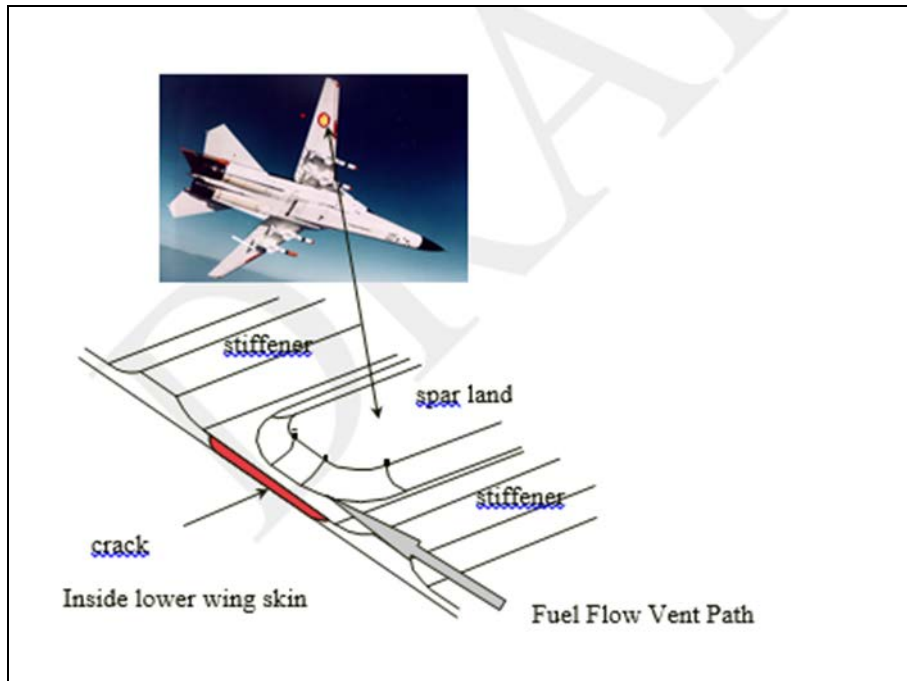


Figure 9: Typical fatigue crack in the F-111C lower wing skin caused by secondary bending at a discontinuity in the spar land.

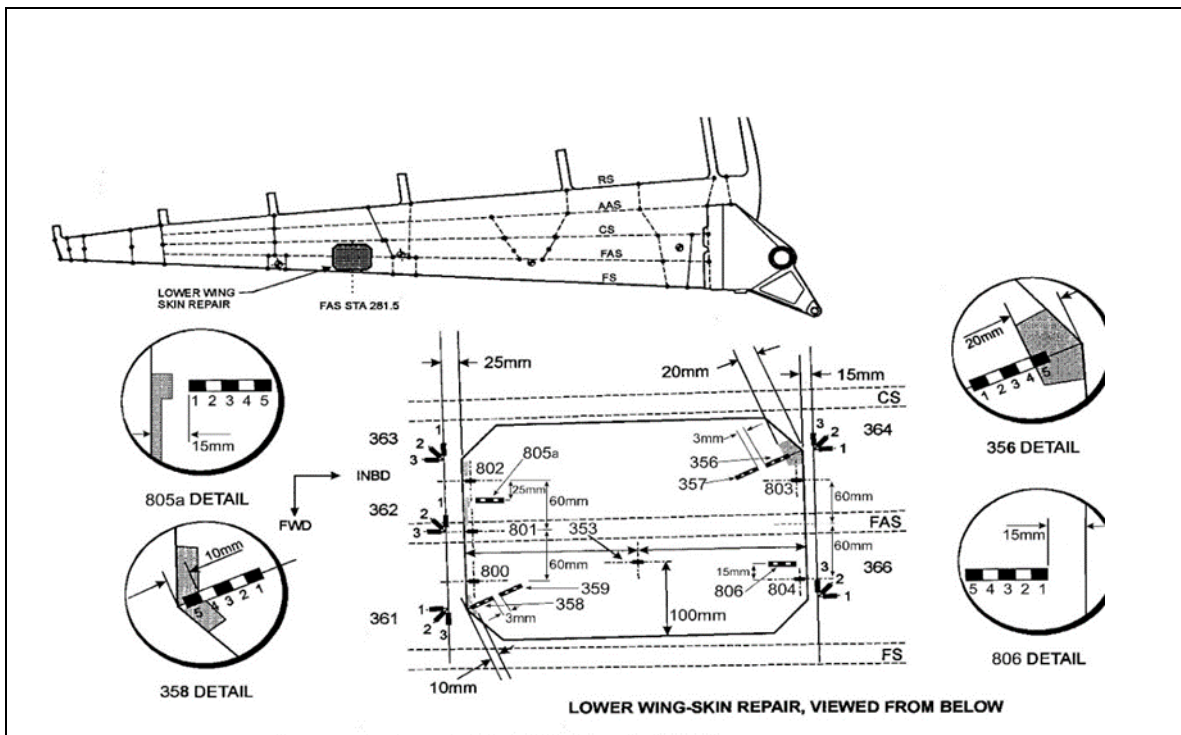


Figure 10: Location of the lower skin patch and SHM strain gauges on a fatigue-cracked F111 wing subjected to a full-scale fatigue test.

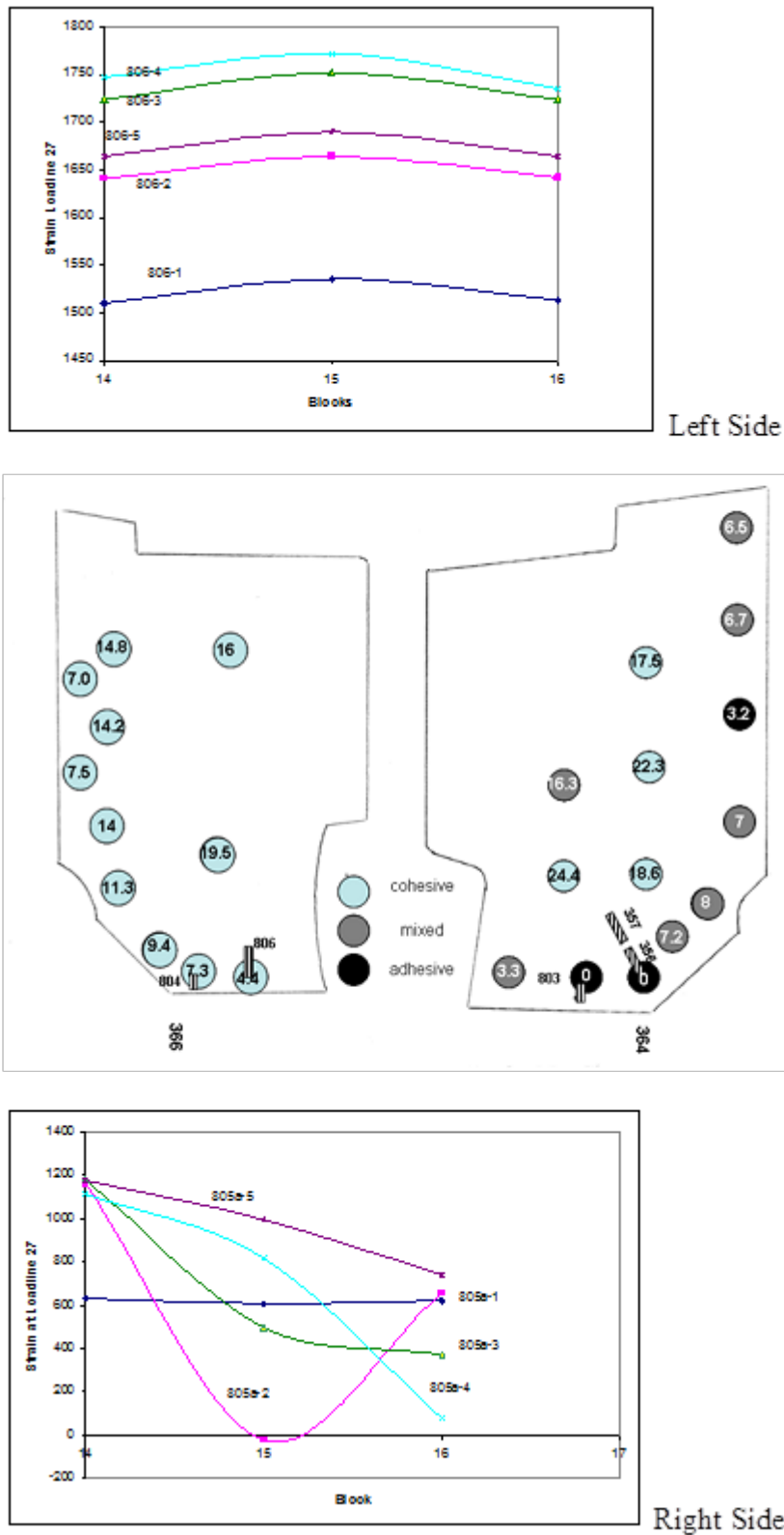


Figure 11: Details of strain changes in the boron/epoxy patch during full-scale fatigue testing. The patch is depicted in the centre diagram; the circles show regions of the patch where post-fatigue pull-off tests were made to evaluate patch bond strength. The graphs show strain readings at a fixed load level as a function of blocks of fatigue loading during the test.

#### **5.2.4 SHM Conclusions**

Based on experience with application of the strain-transfer approach on the F111 full-scale fatigue test and a successful flight test (including power harvesting) on a demonstration repair to an F/A-18 aileron hinge [20], the authors conclude that, at least in the short to medium term, strain-transfer is the most viable approach based on the following characteristics:

- Ability to detect patch disbonding directly and unambiguously when the strain sensors are bonded in the correct locations with sufficient sensor redundancy, and the sensors and associated wiring are strongly bonded, and have the required fatigue and environmental durability.
- Not significantly affected by extraneous factors such as load path, temperature variations, and structural complexity.
- Implementation should be relatively simple and cost-effective.
- Possibility of making the SHM system stand-alone (self-powering) and embedding the sensors, and possibly some of the electronics, in the composite patch.

In support of the above conclusions, it should be noted that conventional resistance strain gauges are used extensively for fatigue-life management in modern military aircraft. They are also used to measure flight spectrum loads. Stresses in (possibly originally unknown) “hot spots” are inferred by stress analysis and use of appropriate transfer functions. However, in this case gauges are usually applied in benign locations where there is low strain and strain gradients. This differs from the proposed strain-transfer SHM approach, which generally exposes strain gauges and connections to more severe loading.

While SHM, based on strain transfer, is feasible for monitoring patch repairs, SHM in general is a costly and complex solution, and depending on the approach adopted, can impact the aircraft’s electrical systems. Thus, even if fully self-contained, SHM is best avoided unless required for particularly critical applications, such as discussed in the next section, or the inspection of hidden structure where conventional NDI is not possible without extensive disassembly of the airframe.

## **6.0 REPAIR MANAGEMENT IN SINGLE-LOAD PATH DAMAGE-TOLERANT METALLIC STRUCTURE**

This proposed management approach is based on the foregoing analysis and addresses the management of repairs to the primary airframe structure when the decision is made to leave the crack in place, as in the case of the F111 wing repair<sup>5</sup>. However, a similar approach can also be appropriate when the crack has been removed and a bonded repair/reinforcement applied. Additionally, the approach applies, even in a modified form, as a very conservative approach to the reinforcement of a fatigue “hot-spot”, although a simpler, and more cost-effective approach for this application is described in the following section.

### **Assumptions**

- A BRC type proof test is accepted as providing assurance of the initial and ongoing structural integrity of the repair bond when the assessment is based on the initial and periodic inspections.
- The patch system is adequately designed to provide the desired reduction in stress intensity in the patched crack, and the patch system can survive the service loading in the absence of environmental degradation.
- Crack growth, including any increase resulting from disbond growth in the patch, can be predicted using a fracture mechanics approach for patched cracks.
- Inspection intervals can be estimated when credit is given to the patch system.
- The SHM system meets the POD and reliability requirements mentioned in the previous section and is primarily used for detecting patch disbonding and possibly also crack growth.

The decision chart shown in Figure 12 is an updated version of the chart shown in Reference 21.

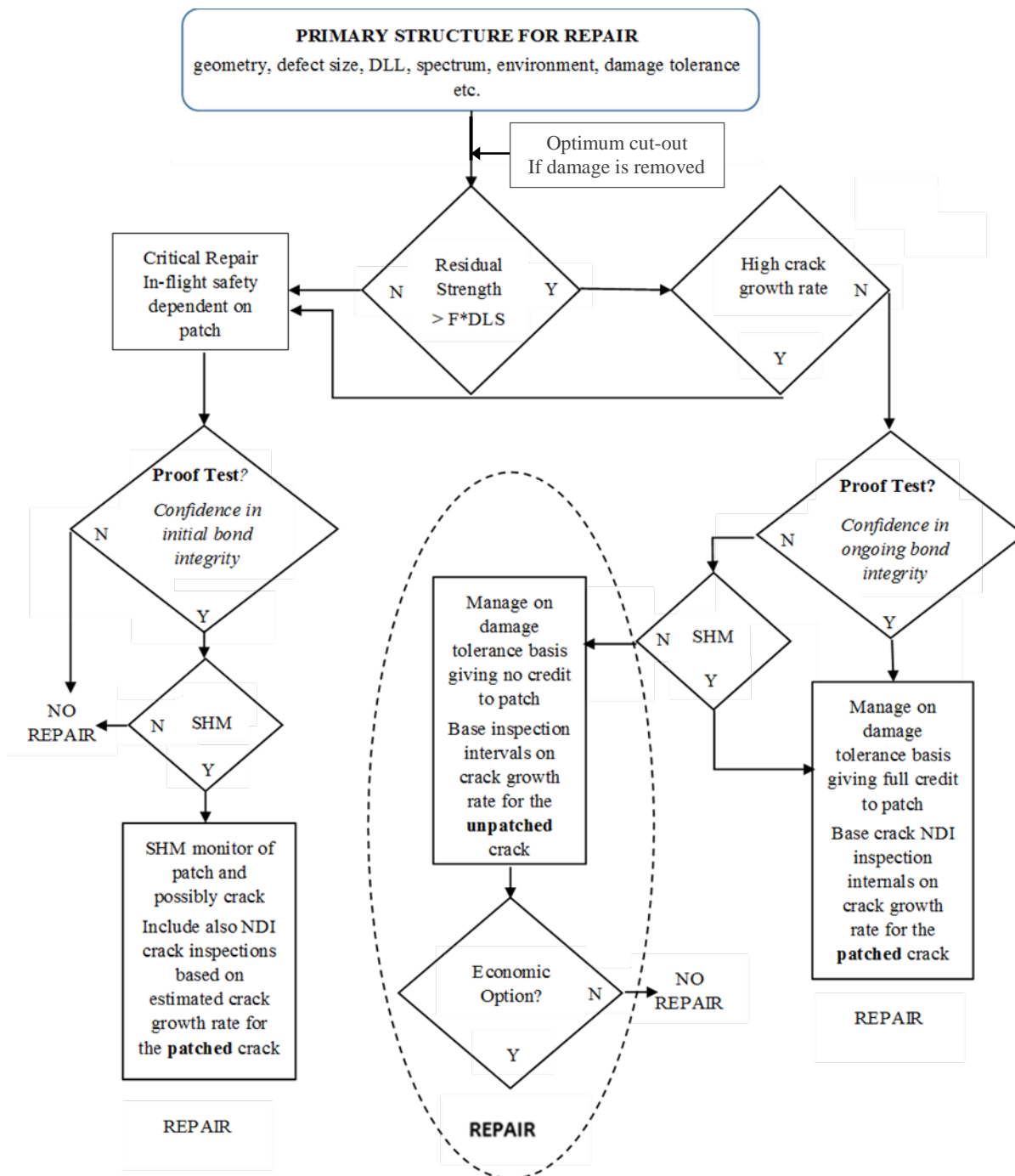


Figure 12: Decision chart for application of bonded repairs to a single-load path structure with fatigue cracking. Dotted area is the conservative damage tolerance approach with no credit given for the patch for reducing crack growth.

- a) Following crack detection, the damage must be assessed. The assessment determines if the crack has reduced the unpatched residual strength below an acceptable level which is defined as an agreed factor,  $F \times DLS$ . If  $F$  is 1, then the residual strength requirement is the design limit stress,  $DLS$ . However, an  $F$  of 1.2 or as high as 1.5 would be selected to provide a safety margin. If the crack has not reduced residual strength below the required level, and the anticipated crack growth rate is slow, then the approach shown on the right side of the chart is followed. If the crack has reduced residual strength below this level, or the anticipated rate of crack growth is high, then the left side of the chart applies.

- b) The right side of the chart shows that if the requirements of the proof test, including periodic inspections, are met, crack inspection periods are based on the rate of growth in the patched crack, with allowance for any reduction in patching efficiency caused by disbond growth over the patch, if this is predicted.
- c) If SHM is used for the patch, periodic proof tests are not required. The proof test is then used only to validate the initial bond strength.
- d) If the proof test or SHM has not been used, then no credit is given to the patch for slowing crack growth, and the inspection period must be the same as the unpatched crack, as indicated by the dotted region in Figure 12.
- e) If the residual strength with the unpatched crack is below the requirement, for example 1.2x DLS [the F111 wing repair mentioned in Section 5.2.3 is an example of this] or if the residual strength is sufficient but crack growth rate in the absence of the patch is anticipated to be very rapid, the approach on the left-hand side of the chart must be taken. This calls for both satisfactory passing of the proof test to ensure initial strength of the patch system is adequate and SHM to continuously monitor the patch. The SHM may also include crack detection. If not, NDI inspections are conducted, based on estimated crack growth rate for the patched crack.

If reliance is made only on periodic proof tests as an option for the right-hand side of the chart, that is when residual strength exceeds  $F \cdot DLS$ , the inspection interval (re-test period) must be determined. Clearly there would be little benefit if the interval is as short as that required for the unpatched crack. However, since the proof test detects degradation and not disbonding it is conservative so the period between proof tests could be quite substantial – a key issue which requires more investigation.

## **7.0 MANAGEMENT OF METALLIC STRUCTURE REINFORCEMENT**

Generally, the objective of reinforcement in a safe-life structure is to reduce strain at so called “hot spots” to extend fatigue life to crack-initiation. Since reinforcement is not a repair the very stringent requirement that residual strength should not fall below DUS in the absence of the reinforcement can generally be met at this stage.

Specifically, the requirement may be either to: a) to extend fatigue life or b) to restore the required life because of either i) underestimation of service stresses or ii) more severe usage.

For safe-life structure NDI may be not conducted during the nominal safe-life period since crack size at initiation will be below the level of detection by NDI. However, if safe-life is to be extended by reinforcement or usage is more severe, interim NDIs will be conducted both on the structure and reinforcement at potential hot-spots.

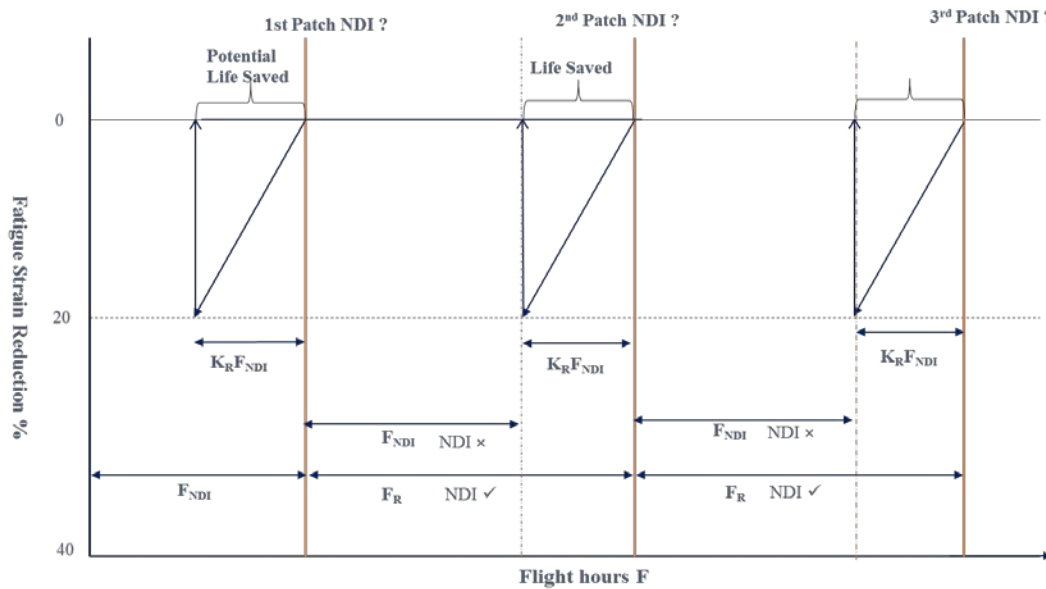
To ensure the absence of undetected cracks, the surface layer may be removed in the region of the hot spot prior to application of the reinforcement. Typically, this would involve surface removal by a “confidence cut”, for example by reaming fastener holes, to remove any cracks that may be present below the limits of NDI. However, if undetected cracks are present, or develop under the reinforcement (for example, due to an insufficient confidence cut) the reinforcement will slow crack growth, making the structure, in effect, more damage tolerant. For small cracks, this is due to the reduced strain, and for larger cracks it also includes crack bridging by the reinforcement.

The SHM approach and the proof test approach can also be used for monitoring reinforcements in a safe-life structure. However, there is a simple alternative, the “retrospective” approach, which does not require SHM or proof testing.

This approach is based on the following assumptions:



- a. Strain reduction by the reinforcement produces a predictable increase in life to crack initiation based on S/N curves developed, for example, from tests on representative structural details – making allowance for residual stresses.
- b. NDI can reliably detect disbonding of the reinforcement following flight exposure, as already demonstrated on the F111 full-scale fatigue test, mentioned in Section 5.2.3.
- c. If the patch is sound, additional flight hours can be added to the next inspection interval based on the predicted crack growth at the lower strain level.
- d. However, if the patch is found to be significantly disbonded then reversion must be made to the original inspection interval – even if the reinforcement is replaced.



**Figure 13 Retrospective approach for extending inspection interval in a safe life structure.**

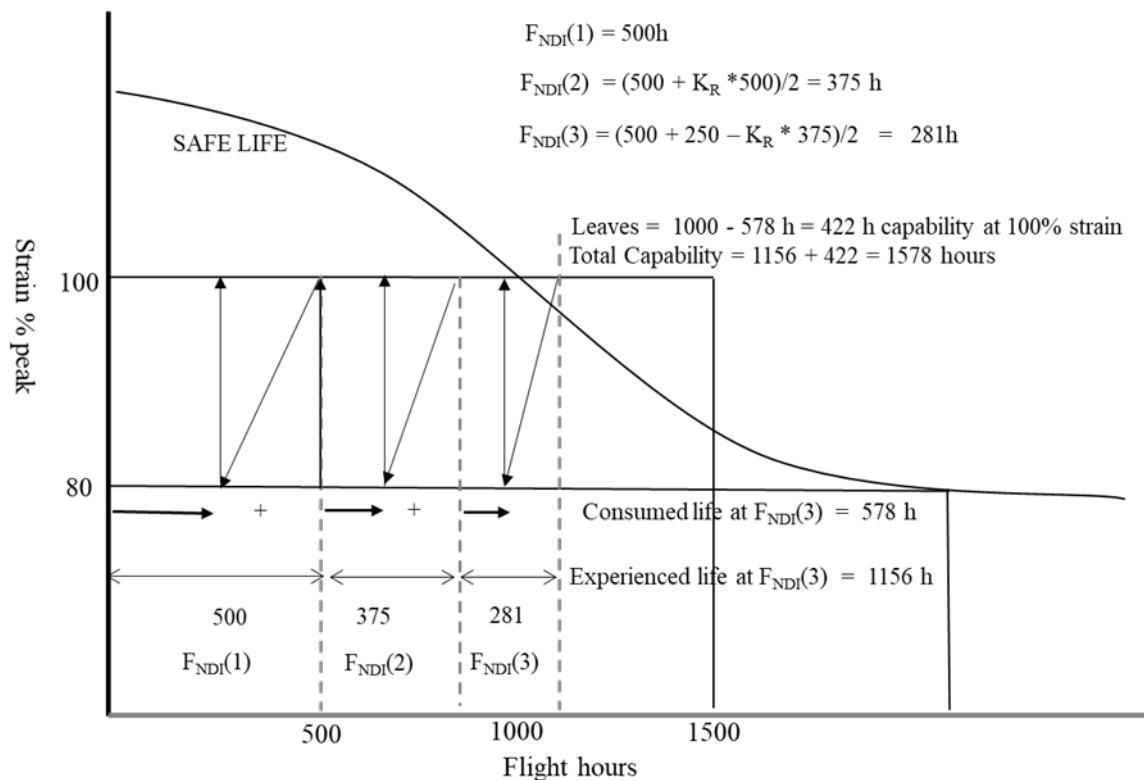
Based on the above assumptions, the procedure is schematically illustrated in Figure 13, which shows the percentage strain (or stress) reduction produced by the reinforcement and the flight hours to each inspection period. The first inspection interval gives no credit to the reinforcement, so the patch and parent structure are inspected at  $F_{NDI}$  flying hours, as required previously for this region of the structure. If the patch is unaffected the next inspection period is increased by:  $K_R * F_{NDI}$  hours, where  $K_R = (1 - F_{NDI}/F_R)$ , and  $F_R$  is the predicted inspection flight hours at the reduced strain level, ideally, based on test data.  $F_{NDI}$  is the current inspection interval.

The credit for the increase in life to initiation by the reinforcement in reducing strain is low risk because it is based on proven effectiveness of the reinforcement in the **previous** period. This retrospective approach is used for each subsequent period, unless the patch is disbonded.

In Figure 13,  $K_R$  is assumed to be 0.5 for a 20% strain reduction and each inspection interval is assumed to be the same as the second interval, if the reinforcement is sound.

More realistically, the inspection interval may be based on 50% of the remaining safe-life at 100% peak spectrum strain. As an example, assume the requirement is to extend the safe life from 1000 to 1500 hours by reinforcement. Figure 14, shows this procedure schematically, based on an idealised safe-life S/N curve

for peak spectrum strain versus flight hours. The unreinforced peak spectrum strain is taken as 100% and this is reduced 20% by reinforcement, again assuming  $K_R = 0.5$ .



**Figure 14 Schematic of the retrospective approach with inspection intervals based 50% of the remaining safe life for an assumed safe life S/N curve for  $K_R = 0.5$ .**

As previously stated, when the retrospective NDI approach is used there is no requirement for SHM or proof testing. However, a proof test validates the integrity of adhesive bond in preparation for the first inspection interval and avoids further accumulation of fatigue damage during this period should the bond integrity prove to be inadequate.

## 8.0 MANAGEMENT OF REPAIRS TO COMPOSITE STRUCTURES

As stated in section 2.2, repairs are required when there is visible damage to a composite structure, for example, a very deep dent, or a surface penetration. This is because residual strength may be reduced to a level close to, or below, DLL. Figure 14 shows a proposed damage management decision chart, which is an updated version of the original chart proposed in Reference [21].

In most cases of damage, the region is removed prior to implementing a repair. Therefore, the damage analysis addresses the repair of a composite component having a circular or elliptical cut-out, or a cut-out designed to provide optimum open hole strength [22, 23]. This facilitates the analysis of residual strength since methods for assessing RS with visual damage are currently unavailable or unproven, especially if further RS degradation under cyclic loading is considered possible.

As shown in Figure 14, the first step is to determine, by analysis, if the damaged structure with a cut-out has an acceptable RS. A conservative assessment may require DUL, while a less conservative requirement can be some other factor  $F \times DLL$ , where  $F$  is lower than 1.5, for example  $F = 1.2$  is called the proof strength.

However, even with a simple cut-out there is the possibility that RS with a cut-out may decrease under service spectrum loading, so a knockdown factor,  $K_{fco}$ , is applied to allow for this eventuality. The effect of service loading is often to reduce stress concentrations resulting from the cut-out by the growth of delaminations and through thickness splits especially in the highly loaded zero-degree plies with the result that  $K_{fco}$  may be greater than one; however, use of this “knock-up” value would not be considered.

For thick material in the primary structure, i.e. greater than 3 mm, the edges of the cut-out require a very shallow taper ( $\sim 3^\circ$ ) to enable application of a scarf repair [9]. This requires the removal of a significant amount of parent material, which together with the tapered edges required, typically results in the RS falling below DUL, so  $K_{fco}$  may indicate a significant reduction.

In the case of part-through damage, removal of the damaged material may be possible using a small cut-out in the form of a flat bottom cavity, with or without tapered sides. This technique can be an option for thick skin material such as a wing, where the thickness may exceed 10 mm.

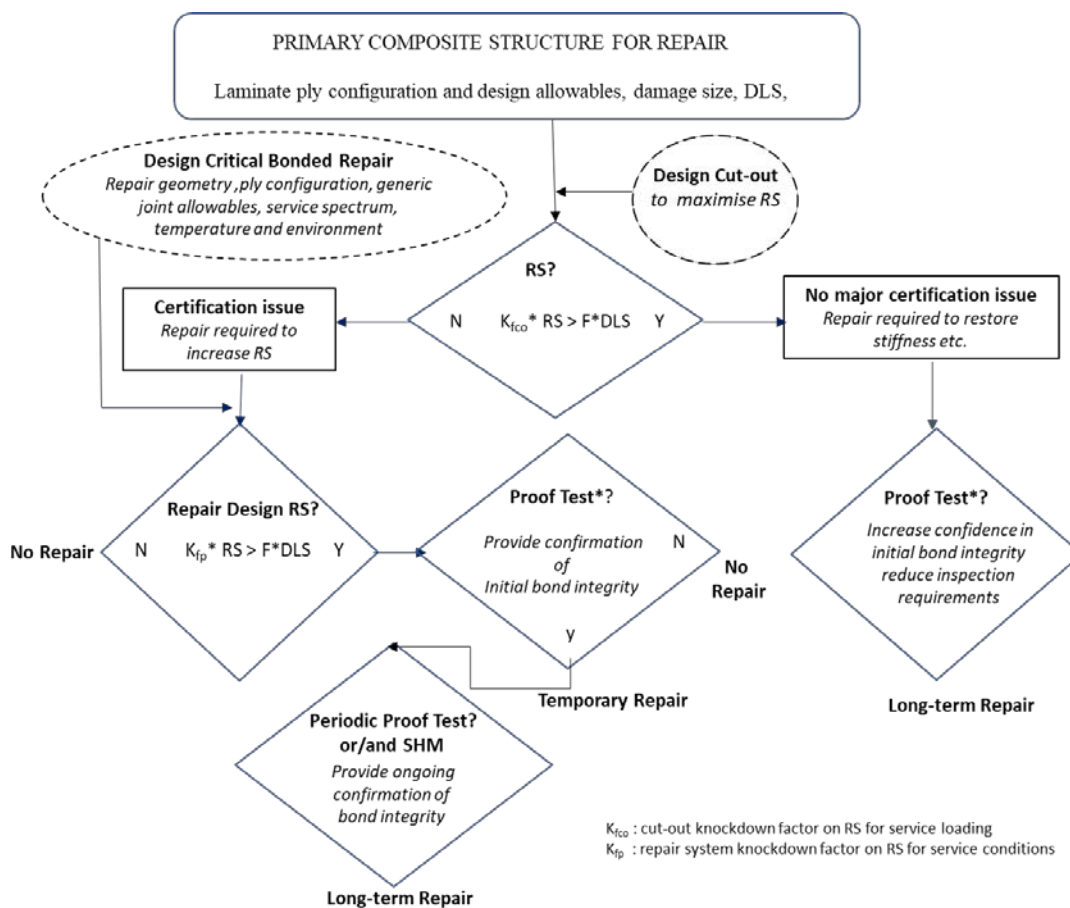


Figure 14: Decision chart for repair of composite primary structure suffering visible damage.

If the RS is unacceptable following a cut-out for damage removal, allowing for  $K_{fco}$ , a patch repair is required to restore RS to an acceptable level. In this case, the left side of the chart in Figure 14 applies because there is a definite certification issue due to reliance on the structural integrity of the patch.

As mentioned previously, RS following repair for the cut-out again must be either DUL, conservatively, or some acceptable factor  $F \times DLL$ , including a knockdown factor,  $K_{fp}$ . In this case the  $K_{fp}$  is a compound factor for estimation of RS degradation resulting from fatigue damage and environmental degradation in the repair system under service conditions. Ideally,  $K_{fp}$  should be obtained from representative data for the

repair joint, as proposed in Reference 21.

Thus, the requirements for the left side of Figure 14 can be summarised as:

- Demonstrate that the patch is capable of restoring RS to the required level. As it is generally not feasible or cost-effective to conduct testing on the specific repair unless it affects several aircraft the design and analysis must be based on generic data for repair joints [21].
- Show that the structural integrity of the patch system can be ensured through the life of the repair. This can be achieved, as indicated in Figure 14, by using the proof test (or an equivalent procedure) to ensure initial strength. This is required to provide initial confidence in the repair, and for a temporary repair, may be all that is required. For long-term application using SHM of the repair, ideally it is combined with periodic proof testing. However, in some intermediate cases, periodic proof testing alone should be adequate.

## **9.0 CONCLUSIONS**

- It has been shown, based on experimental studies, that a torsion proof test conducted on bonded repair coupons can detect weak bonds. Therefore, this test has the potential to be accepted as a method for assessing the initial and ongoing structural integrity of an adhesive bond in a repair patch. However, this claim requires acceptance by the appropriate certification airworthiness authority.
- Several structural health monitoring approaches are available to detect patch disbonding in service, and crack growth under the patch in the case of fatigue cracks in metals if these cracks are not removed prior to repair. Based on practical experience gained in a full-scale fatigue test on an F-111 wing, the most promising approach in the short to medium term is the strain transfer approach, based on resistance strain gauges or optical fibres.
- Management strategies for repairs to composite or metallic airframe structures are proposed based on the decision charts presented in this paper using: a) the proof-test approach to establish, by inference, initial and ongoing bond integrity, and b) SHM to directly check for patch system integrity and damage growth in the parent structure.
- Using these approaches, it may be possible to relax significantly the requirement for residual strength of the component in the absence of a repair. It is important to note that with growing confidence in these approaches, it may be possible to approve repairs in situations where, even with a single-load path structure, the residual strength, in the absence of the repair, may be close to or below the design limit load.
- In the case of reinforcement (rather than repair) of metallic structure a simple retrospective NDI management approach to detect disbonding is described. Although not required, this approach would further benefit from use of the Proof Test to check for initial bond integrity.

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