COMPOSITE REPAIR FOR METALLIC AIRCRAFT STRUCTURES DEVELOPMENT AND QUALIFICATION ASPECTS

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Abstract

Military fighter aircraft are primarily designed for high agility manoeuvre performance throughout their flight envelope for a defined usage life and spectrum. Structural weight of the aircraft is minimized using advanced design principles, highly integrated structures and high strength materials. Compared to civil aircraft structures designed usage life is comparatively short while the stress spectrum is more severe, therefore damage propagation of cracked aluminum structures is significantly higher, requiring reliable detection methods and short inspection intervals. This environment hat originally driven the "Safe Life Design " or "No cracks within specified usage time and spectrum" of many fighter aircraft, later converted to a more "Damage Tolerance Design" approach by some military users to improve fleet utilization.

With increasing demand for extended operational life of many existing aircraft beyond their original design life and the variation in usage / load spectrum caused by operational role changes over time the issue of structural integrity and flight safety must often been ensured through local structural reinforcements and repairs of the primary structure, regardless of design philosophy. Long downtime periods in depot environments for repair or replacements are major cost drivers for ageing fleets together with reduced fleet readiness.

Bonded repairs using composite patches on metallic structures have already proved as cost effective method to increase the life of damaged structures such as external skins and panels of moderate thickness, curvature and global stress levels.

This paper describes the development aspects of a repair-programme for a specific high load transfer area of a metallic engine inlet duct panel featuring a complex geometric contour in an integral fuel tank area of a fighter aircraft.

Design aspects including material selection criteria, local aircraft component loading and the stress analysis of the reinforcement will be discussed.

Qualification requirements for primary structure bonded repairs are presented, the required data substantiation through coupon tests are discussed as well as manufacturing trials for process optimization and the necessary NDT programme to fulfill the quality assurance requirements.

Finally, results of a successful full scale installation trial under "on Aircraft" conditions without disassembly of the structure using the major airframe fatigue test article are presented.

1 Introduction

Military fighter aircraft are primarily designed for high agility manoeuvre performance throughout their flight envelope, the platform as part of the system contributes to this overall performance through lightweight structures with adequate stiffness and strength.

Multiple, sometimes contradictive requirements lead to an optimum compromise through tasks from many disciplines to achieve this structural performance for a defined life and usage spectrum. Modern design principles often use highly integrated structures, tailored material distribution for strength and stiffness and highly loaded joints between components. Compared to military transport or civil aircraft structures usage life is comparatively short for fighter and trainer aircraft, while the maneuvre spectrum and therefore the stress spectra are more severe, Fig. 1.1 shows typ. exceedances of vertical acceleration levels per 1000 flight hours for different classes of aircraft.

This environment had originally driven the "Safe Life Design " or "No cracks within specified usage time and spectrum" concept of many fighter aircraft, later converted to a more "Damage Tolerance Design" or "Slow crack growth" approach by some military users, Fig. 1.2

With increasing demand for extended operational life of many existing aircraft beyond their original design life the issue of Aging Aircrafts have occurred. Other than commercial transport planes, fighter A/C stay in the mil. inventory until they become operational obsolete, but often see changes in their operational usage, when they become non-competitive for their original design mission, but can still be economically operated in a new role, also because of the high cost of replacing a complete AC fleet for the same mission. Originally designed for a typ. lifespan of 20 years, many A/C- fleets exist that have been operational 40 years and longer, sometimes extending the original required design flight hours by a factor of two and more.

This demand for extended usage and variation in load spectrum requires adaptation of maintenance programs, sometimes structural modifications and increasing repairs of the primary structure to ensure the required level of structural integrity.

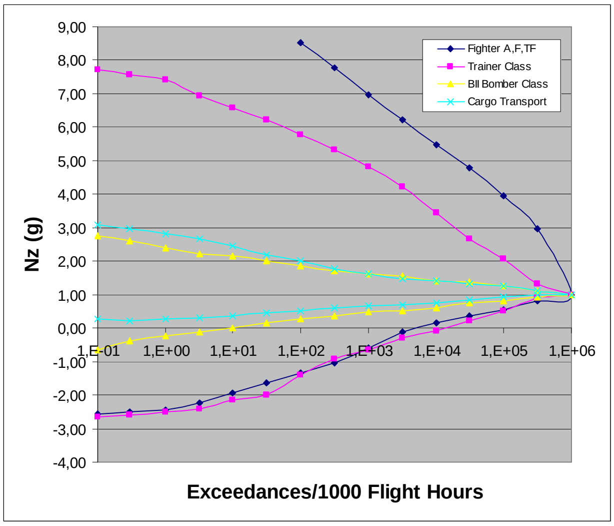

Fig. 1.1 Exceedances /1000 Flight Hours

Aircraft Type	Country	Design Philosophie	Service Introduction
Alpha Jet	France / Germany	Initial Safe Life -> Damage Tolerance	1979
F-14	USA -Navy	Safe Life	1974
F-15	USA –Air Force	Damage Tolerance	1976
F-16	USA –Air Force	Damage Tolerance	1974
Eurofighter	GE / IT / SP / UK	Safe Life	2006
F/A-18	USA -Navy	Safe Life	1983
Hawk MK 1	UK	Safe Life	1976
Mirage 2000	France	Damage Tolerance	1984
JAS-39 Gripen	Sweden	Damage Tolerance	1996
TORADO	GE / IT / UK	Safe Life	1981

Fig. 1.2 Design philosophies used for fatigue management by various operators

2 Design criteria & principles for structures

Although composite materials have replaced high strength aluminium alloys in many applications, complex geometries, assembly reasons, material cost and repairability keep metallic design in aircraft components like machined bulkheads, floor structures and shear walls.

The typ. selection criteria for structural materials are shown in Fig.2.1 for structural mechanical properties only (no cost, production processes and technology aspects included)

Design goal / requirement	Contributing material properties
Minimum structural weight	Specific weight
Stiffnes of structural components and functional elelements (Control surfaces, Launcher, Actuator-Brackets etc.)	Static strength
Adequate static strength for max. expected loads in service	Youngs modulus
Sufficient fatigue life for defined usage spectrum	Fracture toughness
Acceptable damage tolerant und – growth potential for in service damages	Crack growth performance Corrosion resistance

Fig. 2.1 Design goals and material selection properties

3 "In-service"- Damages on metallic A/C structures

Structural fatigue cracking is a result of the usage and will finally occur in all aging aircraft, and service findings in AL-alloys like 7075 and 7079 are common, especially for T-6 heat treatments, tailored for high strength but compromising crack initiation / growth performance and corrosion resistance.

A typical machined aluminium bulkhead between engine ducts and the upper fuselage from a fighter center fuselage is shown in Fig.3.1. Although riveted assembly of parts and substructures generally allow disassembly for inspection, repair and local reinforcements, these activities must often been performed at the aircraft with limited access due to either space or usage of compartments for fuel tanks and equipment bays.

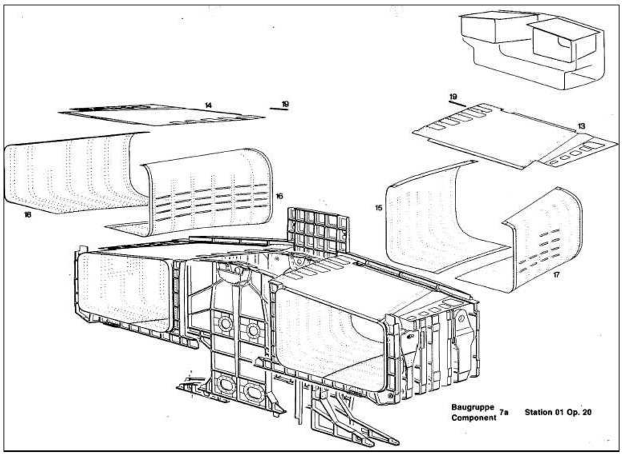

Fig. 3.1 TORNADO Center Fuselage structure

Component	Typ. Materials	Typ.DLL Stress level (N/mm2)	typ. max. Stress concentration factors	typ. (initial) crack growth rate R=0 (mm / loadcycle)
Frames / Bulkheads /	7075 – T7365	200 – 250	2,5 – 3,8	4 * 10-5
Skins, Shearwalls, floors,ducts	2024 – T62	180 - 210	2,0 – 3,6	7,3 *10-5

Fig. 3.2 Typical materials used and global stress levels at design limit load level

Once damage in form of cracks occurs, the damage propagation of cracks per load cycle becomes significantly higher, Fig. 3.2

Monitoring of crack growth in metallic structures requires reliable detection methods and spectrum related inspection intervals to ensure timely repairs / replacement of components before cracks become instable, Fig 3.3

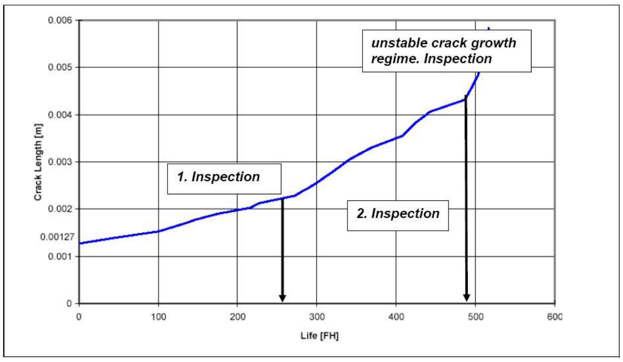

Fig. 3.3 Typ. Crack growth in AL 7075 components at high stress level and crack intensity factors

4 Bonded Repairs of metallic Structures

While component replacements and extensive large scale repairs often cause long downtime periods in depot environments, even higher efforts are required for disassembly of larger parts of the fuselage or wing. Usually this is limited to major overhaul and life extension programs of the fleet. If technically feasable, a local repair or reinforcement of the structure is more economical.

The primary and traditional repair technique for damaged metal structure is to bolt-on a metal sheet or machined doubler made of aluminium, titanium or steel.

The load is transferred into the patch through the fasteners by shear forces.

The external bolted patch is the easiest repair to fabricate. The patch overlaps the parent skin with sufficient area to install the required amount of fasteners to transfer the load. The patch may be stepped and different size fasteners may be used in different rows to ease the load transfer.

For complex repairs, multi-row fastener patterns will be required to gradually introduce the load from the part being repaired into the repair patch. It is virtually impossible to distribute the load evenly between all the fasteners in a multiple row pattern, but careful design of patch geometry, fastener diameter and spacing can alleviate the high loads at the first fasteners.

The external patch thickness may be limited by aerodynamic considerations and by the induced load eccentricity due to neutral axis offset. [1]

Pioneered by the Aeronautical Research Laboratory in Australia [2] for many years composite repair doublers bonded to metal substructures have been used in development programs and partially in fleet repair programs, as an alternative to bolted repairs. [3]

Fig. 4.1 shows the principle of a typ. bonded composite patch repair

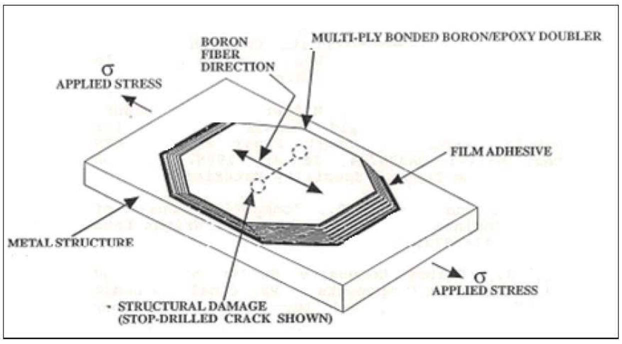

Fig.4-1 Bonded composite patch (typ.)

Bonded composite doublers have the ability to slow down or stop fatigue crack growth, replace lost structural area due to corrosion grindouts, and structurally enhance areas with small and negative margins. This technology has often been referred to as a combination of metal bonding and conventional on-aircraft composite bonded repair. The U.S. Air Force and the Royal Australian Air Force have been using this technology for over 25 years on aircraft ranging from F-5 to Boeing 747 to C-130 to C-141 to B-1B. Commercial aircraft manufacturers and airlines are starting to adopt this technology to their needs.

Boron epoxy, GLARE® and graphite epoxy materials have been used as composite patches to restore damaged metallic wing skins, fuselage sections, floor beams, and bulkheads. As a crack growth inhibitor, the stiff bonded composite materials constrain the cracked area, reduce the gross stress in the metal, and provide an alternate load path around the crack. As a structural enhancement or blendout filler, the highmodulus fiber composites offer negligible aerodynamic performance impacts and tailorable structural properties. [1]

Repair design criteria, part configuration, and the logistic requirements will dictate whether the repair should be bolted or bonded.

4.1 Advantages and Disadvantages of Bonded Repairs on metallic Structures

The overall task for bolted and bonded joints in aircraft structures are identical:
Restoring the damaged structure to a required capability in terms of strength, stiffness, functional performance, safety, service life. Ideally, the repair will return the structure to original capability and appearance.

The continuity in load transfer is re-established in a damaged part by attaching new material by bolting or bonding thus bridging the gap or reinforcing the weakened portion. Thus the repair is in reality a joint where a load is transferred from the parent material into and out of the patch.

The main advantage of bonded repairs is that no new holes are created in the damaged structure which additionally weaken the parent structure and increase stress concentration and local load transfer.

Carefully designed bonded repair doublers provide a more homogeneous stress distribution in the repair zone than bolted repairs.

The out of plane loading from thick metallic doublers is reduced due to the fact, that boron doublers in general require only 33 to 50% of the thickness of a comparable AL-doubler, although bondlines are more susceptible to out of plane loads than mechanical fasteners.

The big advantage of bolted joints is that they can be applied in a comparatively short period of time, the aircraft mechanics are used to work with it and no special tools are required. In comparison the bonding technique is far more complex. Critical areas of the installation process include a good thermal cure control, providing and maintaining water break-free bond surfaces, chemically and physically prepared bond surfaces, technician training and certification.

Hot temperature application for curing the repair materials induces thermal stress in the metal structure.

Also the inspection of bonded repairs is difficult, bonded repairs require a higher commitment to quality control, better trained personnel, and cleanliness.

Still today no satisfactory technique is available for the detection of poor adhesion, this possible defect must be eliminated by checking the adherents prior to bonding and careful process control. Time elaborating ultrasonic-, sonic vibration and X-Ray techniques are the methods most commonly used for the detection of physical disbonds and porosity.

5 The Challenge of Bonded Joints "On Aircraft"

While bonded joints are commonly used in the production phase of aircrafts, the challenge of producing reliable bonds "on A/C" becomes evident when the conditions and manufacturing parameters of this task are considered:

- Qualified original processes are not applicable
- Manufacturing environment is different (depot manufacturing, single side access only, tooling and space limited, etc.)
- Simplified techniques and references required (repair kit, mobile NDI)

Other challenges with repairing aircraft structures with adhesives are often linked to the fact, that thermoset composites repair patches and adhesives are originally cured by a chemical reaction that takes place at temperatures of 175°C (350°F) and a pressure of at least 3 bar (40 psi) in an autoclave.

In a repair situation, particularly if removal of the component from the structure is difficult or impossible, only "hot-bonders" including heat blanket and vacuum pressure can be used for repairs on the aircraft. The alternative use of a room temperature curing adhesive system is usually not possible since these systems are unable to provide the required mechanical strength in the "hot/wet"-environment for fighter aircraft [2].

Therefore not only the production environment is affected, but:

- New materials are required for repair processes (reduced pressure and temperature)
- Manufacturing risk is higher (component integration, aircraft system installation, repairs are often "unique" cases)
- Additional nonproduction processes, i.e. non tank surface treatment for metallics
- Tolerances for curing parameters are higher than during production (i.e.heat blanket temperature distribution and heat-up/cool down rates)

One of the major risks is the application of excessive temperature through the heat blanket, causing serious problems to the surrounding structure and fully assembled parts since they are not designed for such high thermal loads.

While typical cure temperature tolerances for autoclaves are ±5°C (±9°F) and heat-up rate tolerances are as small as ±0,5°C (±°F) the respective values of heat-blankets are up to 4-6 times larger.

Summarizing, the requirements for "On A/C bonding" differ significantly from production processes and experience gained during the original manufacturing of components and aircrafts cannot be automatically transferred to the repair task.

6 Qualification of Bonded Repairs

Whilst bolted joints are basically redundant loadpathes through individual, "independent joints" of multiple fasteners, bonded joints act as "Single Fastener Systems", a bondline failure may lead to a “catastrophic" failure of the complete joint.

In order to make bonded doublers work over an extended period of usage, careful qualification and quality assurance of the manufacturing process is mandatory. Especially the longterm behaviour under environmental conditions (humidity and temperature) proved critical to bonded joints.

Fig. 6.1 shows the route to qualification of bonded repairs and the involved disciplines.

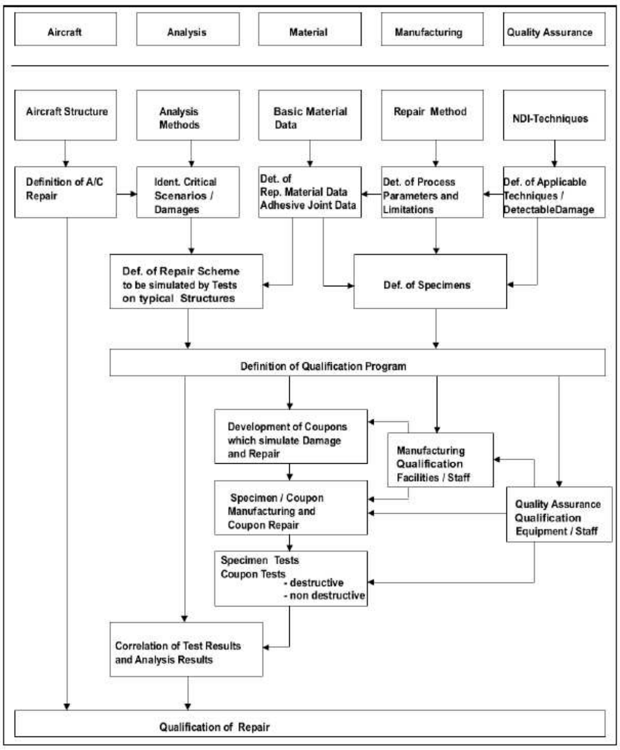

Fig 6-1 Route to Qualification of A/C- repairs

Fig. 6-1 shows the required inputs and disciplines involved in the qualification of a bonded repair.

Special attention during qualification has to be paid on the control of the bonding process and especially the critical manufacturing steps.

Fig. 6-2 shows the principle repair steps

Step	Description	Remark
1	removal of surface protection	
2	sealing of crack and bolds	in order to get vacuum tightness
3	manufacturing of Composite-Doubler	cobonding, secondary bonding
4.	surface treatment of structure	critical step
4.1	chemical treatment of metallic surface	for corrosion prevention non tank technique
4.2	application of Primer	
5	hot bonding of the Composite Doubler	critical step metallic structure and sub structure creates heat sinks
6	NDI of Repair	US, x-ray
7	application of surface protection	

Fig. 6-2 Principle repair steps for bonded repair of metallic structures

For structural bonds, reliable surface treatments are needed to ensure adequate adhesion and durability of the metal-polymer interface. Especially the long-term stability of the bonded joint under severe service environments (corrosion prevention), which is mostly influenced by the metal surface preparation, is crucial. Therefore, complex tank processes, like phosphoric or chromic acid anodising in combination with chromic sulphuric acid pickling are used in standard manufacturing processes.

Here the part is immersed in different bathes according to a defined procedure. All the process parameters are well defined and controlled.

However this processes cannot be applied in maintenance and on A/C-repair scenarios.

Only derived and adopted "local" surface treatment procedures with special techniques can be applied like local anodising processes Phosphoric Acid Anodizing Containment System (PACS), chemical etching and processes based on silane coupling agents to perform durable and reliable bonded doubler installation on aluminium structures.

Another critical step in the bonding process is the curing of the adhesive and the reinforcement material. Similar to the surface preparation standard manufacturing equipment (autoclave) can't be used. For on A/C-repair local heating equipment is used. Because of thermal conductivity of the metal and the high thermal capacity of the substructure application of an uniform curing temperature as required by the repair materials is difficult to perform. Heat sinks, high temperature gradients and non uniform temperature distribution are the consequences.

6.1 Basic Qualification Programme

The qualification programme should establish all data needed for a save application of the repair method to an aircraft structure and satisfy the certification requirements for the platform.

The basic qualification programme includes qualification of both the repair material and adhesive including curing parameters, establishing material allowables, establish confidence in the metallic surface preparation suitable for "on Aircraft application" and a full understanding of the structural requirements and behaviour.

Analysis and closed form analytic solutions are used to design and analyse bonded composite and metallic doubler repairs.

6.2 Qualification Programme

Primary attention must be paid to the structural deficiency, eliminated by the repair i.e. fatigue cracks or corrosion. Therefore "representative Repair Specimen" are a major item of any qualification programme and should not be mixed-up with standard coupons or even components tested for process development and -variation purposes.

In order to establish confidence in a repair method for a given damage and type of structure the different parameters influencing the quality of a repair need to be evaluated and quantified where possible for the manufacturing facilities, quality assurance equipment and staff; i.e. effects of porosity in bondlines or repair laminates and their influence on US-Scan/-interpretation as well as strength properties can only be assessed if considered in this part of the programme.

Striving for the perfect bonded joints, inspected by highly trained and perfectly equipped laboratory quality assurance personal is of questionable value, if later "some porosity" is found with less sophisticate NDI-equipment on the real aircraft and the effects of these findings for the real repair case cannot be assessed. The qualification programme must therefore establish "process-windows" and NDI-margins, where mechanical properties can be based on. These "windows" and margins again need to be tailored to "On-AC-repair" conditions. i.e. heat-up rates and max/min. temperature distribution as well as bondline thicknesses.

The correlation of mechanical and QA-inspections, both for process and repair results is the major step to overall qualification and certification, before a method can be applied to load carrying structures on the aircraft. The same attention is paid for personal qualification and any primary or auxiliary material used during production of the "repair components". Only those materials and equipment that can be used later on-aircraft should be used to manufacture the specimen.

Only personal, experienced with the limitations of performing repair work for composites on the aircraft should manufacture the specimen.

6.3 Example: Coupon Qualification Test Programme

The following is an overview of a typ. repair coupon test programme focusing on: [4]

- Repair concept suitability for applied loads spectra
- On aircraft surface preparation for bonding
- Curing parameters and quality control of the adhesive / bondline
- NDI methods for crack propagation monitoring

6.4 Coupon Repair Concept

The principle repair concept is shown in Fig. 6.4-1 where a typ. fatigue crack in the base material (high strength Al-2024/T3, 3,2 mm thickness) has been repaired using a single side, tapered multi-ply composite Boron Epoxy (Bor/Ep) prepreg patch, bonded to the substrate using film adhesive

The ply orientation of the B/Ep-doubler with its high stiffness (elastic modulus is 210000 Mpa) is generally perpendicular to the crack propagation direction to ensure high stiffness ratios and therefore reduced substrate stresses with min. patch offset.

The patch can be either cured in situ with the adhesive in a single cure application, secondary bonded after cured at the crack location (Film adhesive replaced by release film for patch curing) or precured in an autoclave and secondary bonded.

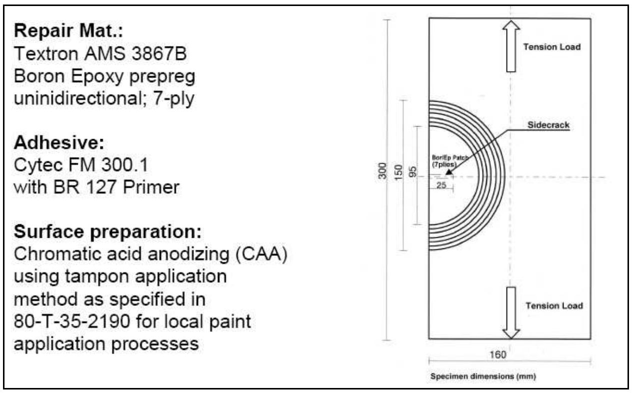

Fig. 6.4-1 "Single side" Crack Tension Specimen

The standard autoclave curing cycle for AMS 3867 Boron Epoxy prepreg, Fig. 6.4-2 has been used together with the so called "hard patch repair process", a method developed in earlier programs for high quality composite patches for "On-Aircraft" applications, where an autoclave cured patch is bonded on the aircraft structure in a second process, thus avoiding patch quality variations in this programme.

Cure Parameter	Textron 5521	Unit
Cure Temperature	120	°C
Heat up rate	2-3	°C/min
Cure pressure	5×10^5	Pa
Dwell time	60	min

Fig. 6.4-2 Autoclave cure of repair patch

The bonding cure cycle has been modified according to the on aircraft requirements in terms of temperature and pressure, Fig. 6.4-3. the dwell time has been extended to compensate for reduced temperatures.

Cure Parameter	FM 300.1	Unit
Cure Temperature	125	°C
Heat up rate	4 - 6	°C/min
Cure pressure	vacuum	
Dwell time	24	h

Fig. 6.4-3 Repair cure cycle

The controlled application of temperature in the repair cure cycle regime (125°C) required tests on typical aircraft structures with metal substructure to identify the effect of heat sinks and heat flow control via thermocouples. Test on wing skins indicated heat loss gradients within a conventional portable heat blanket of 10-20°C, therefore a sufficient large heat blanket with 0,76 W/cm^2 heating capability has been chosen for this programme.

A repair material characterization programme including parameters like moisture pick-up and non-destructive testing parameters for the boron epoxy patch was performed using a vacuum cured laminate of typical repair patch thickness to assess quality assurance performance for later repair tasks. Especially parameters to monitor existing cracks underneath the repair doubler were of interest.

Results of the NDI-Test using various techniques under simulated "on- aircraft" conditions are summarized in Fig. 6.4-4 .

Test Method	Bondline Inspection	Crack Monitoring
US Impulse Echo	Debonds and porositiy in bondline ok.	Not feasible
HF Eddy current (dedicated probe)	not feasible, patch thickness ok.	length and width of simulated crack
X-Ray	ok.	ok.

Fig. 6.4-4 NDI Results

Additional tests were performed for special "On-Aircraft" repair scenarios like bonding patches over flush rivets in skins to verify sealing methods for the rivets, avoiding penetration of fluids during the CAA process into the joints as well as applying the surface preparation process to vertical / overhead skins, a situation often encountered on wings, inside fuselages and engine ducts.

6.5 Coupon Test Programme

The component test programme covered 9 specimen as shown in Fig. 6.5-1, tested to identical load levels of 138 MPa but different spectra under RT/dry and RT/saturated conditions.

Spec.No	Test Condition	Loading	Environment
1.1	Reference, no Rpair	Const. Amplitude R=0,1	RT
1.2	Reference, no Rpair	FALSTAFF R=0,05	RT
2.1	Repaired	Const. Amplitude R=0,1	RT
2.2	Repaired	Const. Amplitude R=0,1	RT
3.1	Repaired	Const. Amplitude R=0,1	RT saturated
3.2	Repaired	FALSTAFF R=0,05	RT saturated
4.1	Repaired, patch sealed	Const. Amplitude R=0,1	RT saturated
4.2	Repaired	FALSTAFF R=0,05	R saturated

Fig. 6.5-1 Coupon Test Programme

6.6 Coupon Test Results

Typ. results for the reference and the repaired specimen are shown in Fig. 6.6-1 for the const. amplitude spectra and in Fig. 6.6-2 for the flight by flight spectra.

As expected, the repair patch proved to be highly efficient, increasing total life by a factor of seven and above for this load level. Specimen indicated stable slow crack growth up to the moment where the crack met the boundary of the patch. Saturated specimens indicated a detrimental effect of the humidity on bondline performance leading to higher crack growth rates (Spec. Nr. 3.1 versus Spec Nr. 2.2) while sealing the patch after curing seemed to improve the patch performance (Spec. 4.1 versus Spec 3.1), however these results could not be fully verified due to the lack of statistical basis.

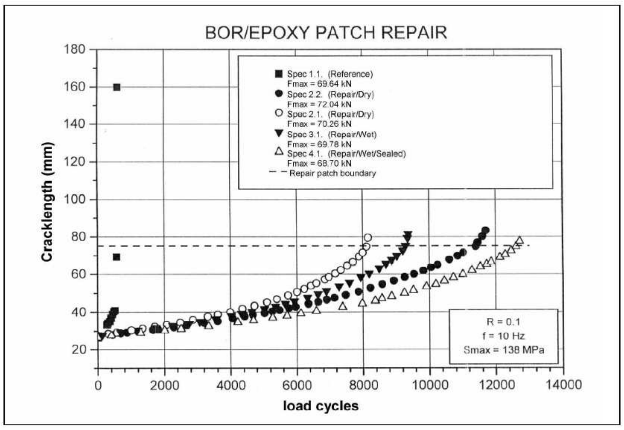

Fig. 6.6-1 Component Tests (Const. Amplitude)

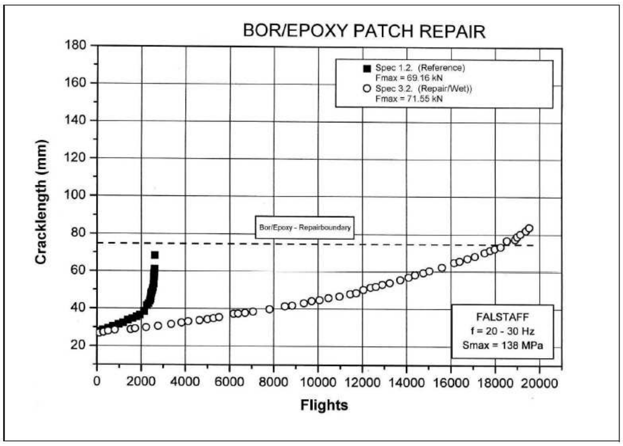

Fig. 6.6-2 Component Test (FALSTAFF)

7 Full Scale Trial Installation under "on Aircraft" Conditions

Qualification of the required structural life according to the weapon systems design specification is generally achieved through a combination of analysis and test, with the full scale fatigue test article representing a real aircraft structure, loaded with real time load - histories and -spectra.

The test article is selected as the formal test bed for a repair development programme at a high stress transfer zone, were conventional repairs and reinforcements have already been introduced and tested before.

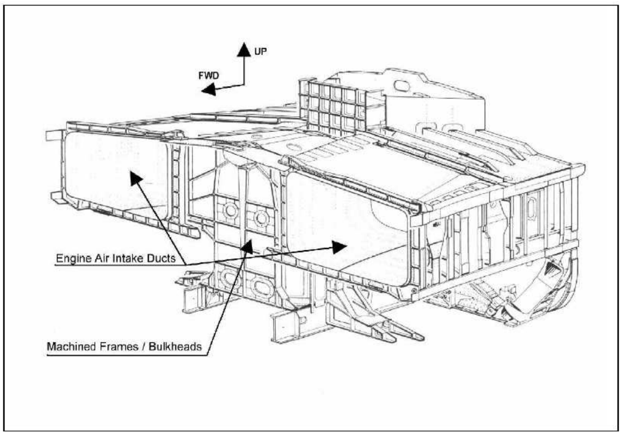

Fig. 7-1 Typ. center fuselage structure (outer skin panels not shown)

Access to the location is limited from inside the engine duct only, the location is close to the lower end of the upper radius of the duct structure. Target for the repair was test continuation for additional 1000 test hours with no further crack growth during this period. Using boron epoxy bonded doublers, due to the curvature of the duct panel in circumferential direction and the change of radius in the longitudinal direction the patch geometry was limited to 9 plies with a 4 mm taper / ply.

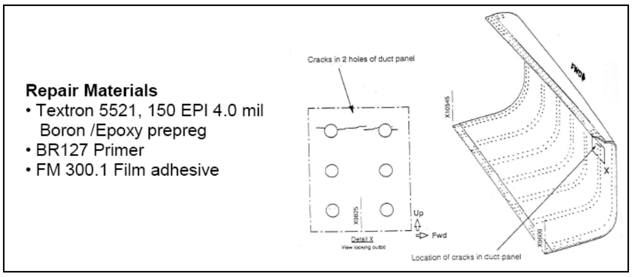

Fig 7-2 Repair location in engine duct

7.1 Repair design

Analytical investigations for the max. loading conditions indicated a circumferential stress level of 200 MPa at Design limit load (DLL) in the repair zone. In order to achieve no crack growth for the desired additional testlife, crack growth analysis indicated a reduction of approx. 50% in the substructure would be required.

Initial closed form calculations for a simplified model were performed to identify

- min. adhesive length (identical to non tapered doubler length)
- reduction of plate stresses under doubler
- max. shear / strain for adhesive
- max- tension stress/strain for doubler
- max. tension stress in plate at edge of doubler

A more detailed analysis was then performed to include thermal stresses from curing, bending stresses due to single side doubler and adhesive elastic- plastic behaviour for max. ultimate load level of 300 MPa (equal to 200 Mpa at DLL) gross stress.

Results of the detailed analysis :

- Min. bondline length of 91 mm + taper of 13 mm
- Plate stress under doubler reduced to 175 Mpa (- 40 %)
- Max. tension stress in plate at edge of doubler increased to 337MPa (+12%)
- Max tension boron epoxy strain of 5300 µm/m (Design allowable 5000 µ)
- Max. shear strain for adhesive of 0.2771 rad (ideal elastic/plastic) for nominal thickness, reduced to 0.1973 rad for actual bondline thickness (mat. allowable 0.1600 rad)

Although the analytical results were slightly unconservative, it was decided to stay with the initial design for test purposes since the local stress levels in the aircraft structure were considered conservatively high.

7.2 Repair doubler application (including Repair steps)

The repair doubler was cured off the aircraft using standard heater blankets with the following parameters

Cure Parameter	Textron 5521	Unit
Cure Temperature	125	℃
Heat up rate	2 - 3	℃/min
Cure Pressure	0.25	atm abs.
Dwell time	60	min

Fig.: 7.2-1 Repair curing cycle

After completing the cure cycle for the doubler, the patch was tested via X-Ray and considered free of delamination and unacceptable porosity (<1 %), the patch bondline interface was then surface treated using abrasive blasting.

Surface preparation process of the engine duct skins, Fig. 7.2-2

- Remove paint in repair area using grid 120 - 280 + scotch-brite + cleaning
- Seal fastener heads and any structural gaps using adhesive
- CAA repair area according to EADS 80-T -35-2190 process spec.
- Apply primer BR127 and cure using heat lamps

Fig. 7.2-2 Surface preparation in repair zone

After surface preparation a bondline check with the cured patch and heat blankets using EADS-"Hard patch process" specification was performed to verify temperature distribution in the bondline and the geometric match of the cured patch with the substructure, vital for the thickness distribution and local cure pressure of the adhesive. The results indicated sufficient temperature conformity between 115°C (min) and 142°C (max.). Fig. 7.2-4 indicates the location of temperature sensors during the bondline check process.

Cure Parameter	FM 300.1	Unit
Cure Temperature	120	°C
Heat up rate	2 - 3	°C/min
Cure Pressure	0,25	atm abs.
Dwell time	24	h

Fig. 7.2-3 Bonding cure cycle

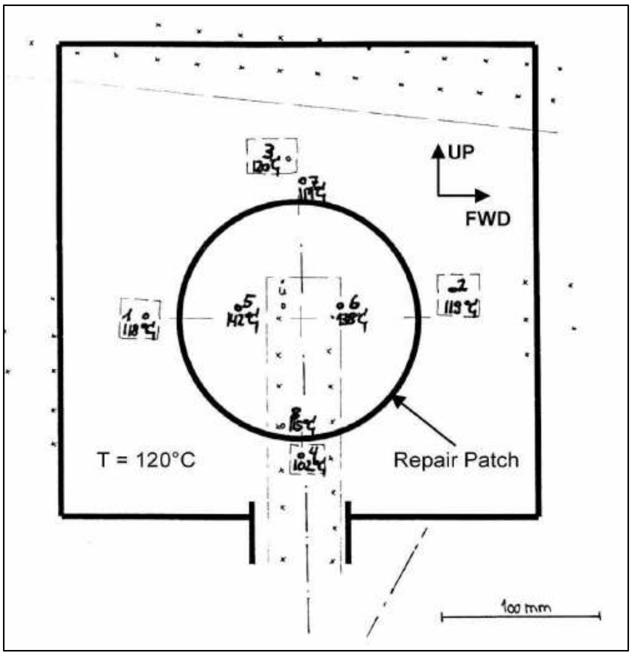

Fig. 7.2-4 Heat sensor elements during simulated bondline curing

While the min. temperature will determine cure cycle dwell time, max. temperatures need to be closely monitored in real aircraft applications in order to stay within limit of sealants for fuel tanks, cable looms or other equipment installed in the vicinity of the repair, especially when access is limited to single side

The final application process of the repair used a total cure time of 24 h and 120 °C, as shown in the a.m. Fig 7.2-4, the repair set up is shown in Fig. 7.2-5, the completed repair after debagging in Fig. 7.2-6

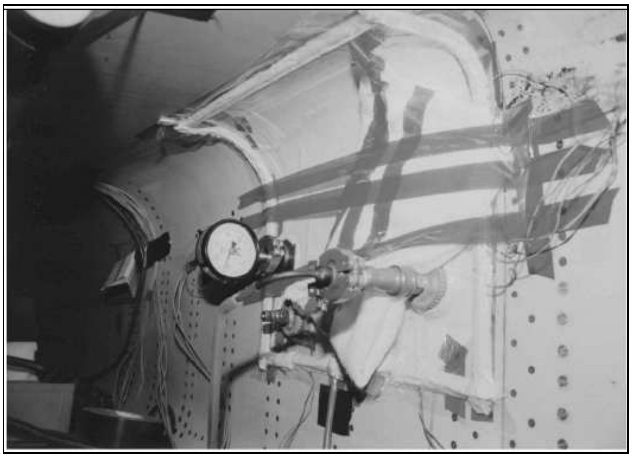

Fig. 7.2-5 Repair bonding process

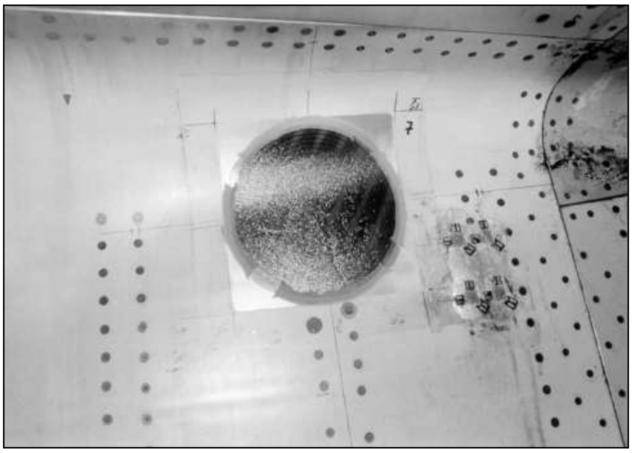
Fig. 7.2-6 Repair bonded to duct skin

7.3 Test Results

The bondline was inspected after curing using standard ultrasonic NDI and was considered free of delaminations and unacceptable porosity. The crack between two fasteners was detected through the repair patch using eddy current NDI, however, the two smaller cracks developing from the fasteners to either side opposite of the link-up crack could not been detected at this point in time. After restart of the test, NDI of the repair was performed after 500 test hours using identical techniques to indicate any changes [5]. No changes were detected for the bondline and eddy current (ED) showed the two smaller cracks LH / RH of the two fasteners missed before in the first inspection, original NDI plot see Fig. 7.3-1.

After 1000 testhours these two cracks increased in length and ultrasonic testing of the patch revealed delaminations in the bondline around the two fasteners and the crack location, Fig. 7.3-2.

Although the full "crack arrest target" was not achieved, the repair proved successful in slowing down the crack growth and enabling the full scale test continuation for add. 1000 test hours without unstable crack growth.

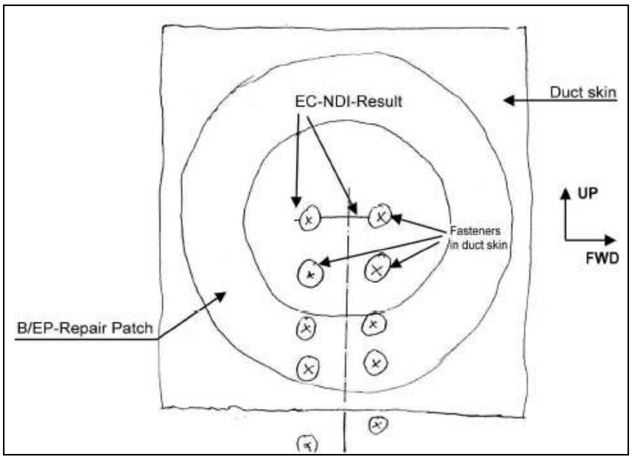

Fig. 7.3-1 EC-NDI Results at 500 Testhours

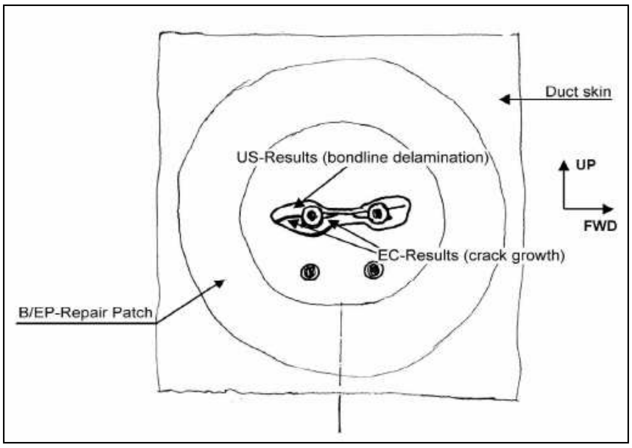

Fig. 7.3-2 EC-NDI Results at 1000 testhours

7.4 Economical Consideration

Literature shows cost saving in workmanship up to 50% - 75%. [3]

An cost analyses of the repair of the Tornado MAFT Duct panel repair shows overall cost saving of 80 % compared to component replacement (Fig. 7.4-1). This is mainly due to the fact that a high effort is required for aircraft component disassembly in order to be able to remove the affected panel. The damaged panel is bolted to the structure with more than 200 bolts which also have to be removed.

The bonded repair was applied according to the repair steps as outlined in Fig. 6-2.

It is assumed that with this method for a fleetwide repair additional cost savings could be achieved by standardising the applied repair procedure.

Action	Costs	Remarks
Replacement of Panel	100 %	high disassembly effort necessary
Bolted Metal-Doubler	14 %	
Bonded Composite Doubler	21 %	+ additional advantage in case of fleet wide application => fleet availability

Fig.7.4-1 Comparison Repairing Costs

Figure 7.4-2 shows the test article and the location of the repair.

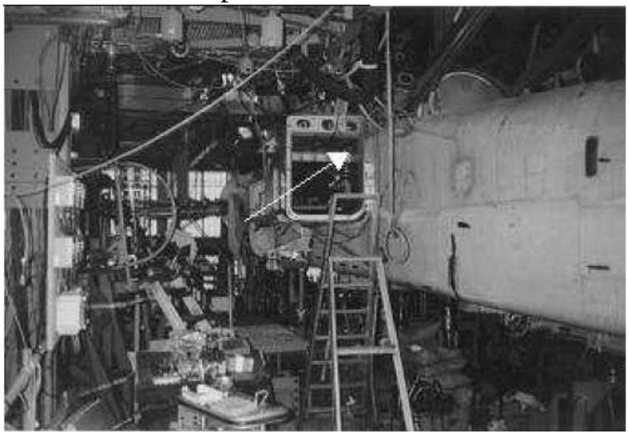

Fig. 7.4-2 Repair Location

8 Summary

Structural fatigue cracking in airframes is a result of the usage and will finally occur in all aging aircraft, extensive repairs are often linked to major disassembly of structures and long downtime periods

Bonded repairs using composite patches over metallic structures have been evaluated as a cost effective method to increase the life of damaged structures

A typical damage within a highly loaded riveted skin to substructure fuselage component has been repaired under "on aircraft conditions".

The repair data substantiation through coupon tests as well as manufacturing trials for process optimisation and the necessary NDT – techniques proved essential to fulfil the quality assurance requirements.

The component test programme carried out to validate repair principles and especially the surface preparation process indicated the feasibility of a bonded boron/epoxy repair within the physical limitations encountered later. Analysis indicated also the need for careful design including aspects like thermal loads from curing introduced into substructure and adhesive, however secondary bending from both single side doubler and the structures curvature in the repair area did influence the behaviour and would have required detailed investigation if applied to fleet aircrafts. The thermal checks using the "hard patch method" proved again vital for bonding processes using heat blankets on skins with complex substructures due to the effect of heat sinks from massive aluminum frames and longerons attached.

Non destructive testing using eddy current was able to monitor the repair patch and the crack behaviour.

Crack growth and the start of delaminations in the bondline around the crack after 1000 test hours due to the very high load transfer in this area indicated the need for addition design, analysis and test effort for bonded repairs at these stress levels.

References:

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[2] A.A. Baker / R Jones "Bonded Repair of Aircraft Structures" 1988

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[4] A. Maier, A. Rottschäfer "Longterm behavior of composite repair patches for metallic structures", EADS - Internal Report, TN-CFK-0111-A; 1998; unpublished

[5] A. Maier, W. Riedl "Bor /Epoxy repairs on a Fullscale Fatigue Test Article" EADS Internal Report; unpublished

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