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FIELD REPAIR OF COMPOSITE PLATES USING FIBRE METAL LAMINATES

By
Sheldon Clarke, B.Eng.

A thesis submitted to
The Faculty of Graduate Studies and Research
In partial fulfilment of
The requirements for the degree of

Master of Applied Science

**Department of Mechanical and Aerospace Engineering
Ottawa-Carleton Institute
For Mechanical and Aerospace Engineering**

Carleton University
Ottawa, Ontario
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FIELD REPAIR OF COMPOSITE PLATES USING FIBRE METAL LAMINATES

Submitted by Sheldon Clarke, B.Eng.
In partial fulfilment of
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Thesis Supervisor

Co-supervisor

Chair, Department of Mechanical and Aerospace Engineering

Carleton University
January 2002

ABSTRACT

This investigation evaluated the fibre metal laminate GLARE as a possible patch material for field repair of composite laminates. A two-dimensional finite element model was used to simulate the mechanically fastened patch repair of typical penetration damage in a wing skin laminate. The effects of varying patch parameters such as thickness, fastener diameter, and geometry were considered. The analysis results were compared with experimental data gathered from biaxial testing of repaired biaxial test specimens. Patch effectiveness was judged by the decrease in strain near the damage after patching, and the strain introduced by the fastener holes in the composite and the patch. Thicker patches resulted in a reduction of the stress concentration near the damage but caused higher strain at the fasteners. Changes in fastener diameter did not significantly affect the results. By using a moderate modulus material, GLARE, in place of a stiffer material, titanium, the repaired area experienced lower strain. A GLARE patch does not attract as much load to the damaged area and is sufficiently strong to carry the required load without failure. This technique of soft patching has potential benefits in repairing composite laminates.

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UNITS

The U.S. system of units is used in place of the International System (SI) throughout the thesis in recognition of its use in the aerospace industry of North America.

U.S. Units	SI Units
1 in.	25.4 mm
1 ft.	0.305 m
1 lbf.	4.45 N
1 kip (1000 lbf)	4450 N
1 psi	6895 Pa
1 ksi (1000 psi)	6.895 MPa

TABLE OF CONTENTS

LIST OF TABLES	XI
LIST OF FIGURES	XII
1 INTRODUCTION	1
2 COMPOSITE REPAIR.....	4
2.1 LEVELS OF REPAIR.....	4
2.1.1 Depot Repair.....	4
2.1.2 Field Repair	6
2.2 COMPOSITE DAMAGE TYPES.....	6
2.2.1 Matrix Cracks.....	7
2.2.2 Delaminations	8
2.2.3 Debonds.....	8
2.2.4 Fibre Fracture	9
2.3 REPAIR CRITERIA.....	9
2.3.1 Static Strength and Stability.....	10
2.3.2 Durability	10
2.3.3 Stiffness	11
2.3.4 Aerodynamics	11
2.3.5 Weight and Balance.....	12
2.3.6 Temperature and Environmental Effects	12
2.3.7 Cost, Personnel and Schedule	13
2.4 REPAIR OPTIONS.....	14
2.4.1 Non-Patch Repairs	14
2.4.2 Patch Repair	14
2.4.2.1 Bonded Patch Repair	15
2.4.2.2 Mechanically Fastened Patch Repair.....	17

3	COMPOSITE REPAIR STRENGTH PREDICTIONS.....	20
3.1	LOAD DISTRIBUTION.....	20
3.2	STRESS FIELD DETERMINATION.....	20
3.2.1	Open Hole Stress Field.....	21
3.2.2	Bolt Loaded Hole Stress Field	22
3.3	FAILURE PREDICTION.....	23
4	FIBRE METAL LAMINATES	24
4.1	GENERAL HISTORY	24
4.2	MATERIAL PROPERTIES.....	25
4.2.1	Fatigue	26
4.2.2	Environmental Durability.....	27
4.2.3	Impact Resistance	27
4.2.4	Flame Penetration	28
4.3	APPLICATIONS.....	28
4.4	REPAIR.....	29
4.4.1	Soft Patching	29
5	PROJECT DEFINITION.....	32
5.1	OBJECTIVES.....	32
5.2	APPROACH	32
6	SPECIMEN DESCRIPTION.....	34
6.1	OFF-AXIS SPECIMEN	34
6.2	THIN-WALLED TUBULAR SPECIMEN	35
6.3	FLAT PLATE CRUCIFORM SPECIMEN	36

6.3.1	Previous Bearing Test Program	37
6.3.2	Cruciform Specimen Dimensions & Properties	38
7	FINITE ELEMENT ANALYSIS	42
7.1	FINITE ELEMENT MODEL DESCRIPTION	42
7.1.1	Structural Problem	43
7.1.2	Methodology	44
7.1.2.1	Elements.....	45
7.1.2.2	Two-dimesional Elements	46
7.1.2.3	One-dimensional Elements	47
7.1.2.4	Mesh.....	49
7.1.2.5	Materials	50
7.1.3	Loads and Boundary Conditions	51
7.1.3.1	Load Introduction.....	51
7.1.4	Boundary Conditions	52
7.1.4.1	Multi-point Constraints (MPC)	53
7.2	TEST PROGRAM.....	54
7.3	FINITE ELEMENT ANALYSIS AND RESULTS.....	58
7.3.1	Test Case A – 8 in. x 8 in. Patch, 1 in. Fastener Pitch - Results	58
7.3.2	Test Case B – 6 in. x 6 in. Patch, 1 in. Fastener Pitch	64
7.3.3	Test Case C – 8 in. diameter Patch.....	70
8	TEST SETUP.....	77
8.1	TEST FRAME.....	77
8.2	TEST CONTROLLER	79
8.3	DATA ACQUISITION.....	79
8.4	PHOTOELASTIC OBSERVATIONS	80
8.5	FASTENERS.....	82

9	EXPERIMENTAL VERIFICATION PROGRAM	85
9.1	TEST CASE U – UNREPAIRED SPECIMEN	85
9.1.1	Experimental and Analytical Comparisons	89
9.2	TEST CASE A6 – LARGE RECTANGULAR REPAIR PATCH	91
9.2.1	Experimental and Analytical Comparison	96
9.3	TEST CASE C10 – CIRCULAR REPAIR PATCH	101
9.3.1	Experimental and Analytical Comparison	106
10	FAILURE INVESTIGATIONS	110
10.1	TEST CASE B MODIFICATION	110
10.1.1	Finite Element Investigation	111
10.1.1.1	Failure Criteria	111
10.1.1.2	Iterative Failure Investigation	113
10.1.2	Test Method	116
10.1.3	Experimental Results	120
10.2	TEST CASE F – LARGE DAMAGE, SQUARE REPAIR PATCH	122
10.2.1	Finite Element Investigation	123
10.2.2	Test Method	125
10.2.3	Experimental Results	128
11	DISCUSSION OF RESULTS	132
11.1	GLARE AS A COMPOSITE REPAIR PATCH MATERIAL	132
11.2	EFFECTIVENESS OF GLARE PATCH	135
11.3	FINITE ELEMENT MODEL ACCURACY	135
12	CONCLUSIONS	138
13	FUTURE WORK	139

LIST OF REFERENCES.....	140
BIBLIOGRAPHY.....	145

LIST OF TABLES

TABLE 4-1 COMMERCIALLY AVAILABLE GLARE (ASUNDI & CHOI 1997)	25
TABLE 4-2 GLARE MATERIAL PROPERTIES – ALL LAYUPS 3/2 (ANON 1997)	26
TABLE 6-1 – LAMINA MATERIAL PROPERTIES (HARRISON 1998)	39
TABLE 6-2 CRUCIFORM SECTION PROPERTIES	41
TABLE 7-1 FEA MATERIAL PROPERTIES (MSI, IN) (HARRISON 1998)	50
TABLE 7-2 GLARE MATERIAL PROPERTIES (MSI, IN.)	51
TABLE 7-3 FEA TEST MATRIX	55
TABLE 7-4 TEST CASE A FEA RESULTS	59
TABLE 7-5 TEST CASE B – MAX. PRINCIPAL STRAIN UNDER 30 KIP LOADING	65
TABLE 7-6 TEST CASE C - FEA STRAIN RESULTS UNDER 30 KIP LOADING	71
TABLE 8-1 PHOTOELASTIC COATING PROPERTIES	81
TABLE 9-1 TEST CASE U - PARAMETERS	86
TABLE 9-2 TEST CASE A6 - PARAMETERS	92
TABLE 9-3 TEST CASE C - PARAMETERS	102
TABLE 10-5 GLARE BEARING STRENGTH (SLAGTER 1992)	112
TABLE 10-1 TEST CASE B – FAILURE INVESTIGATION – GLARE3-2/1 PATCH	113
TABLE 10-2 TEST CASE B – FAILURE INVESTIGATION – GLARE3-2/1 PATCH – LIMITED FASTENERS	114
TABLE 10-3 TEST CASE B – FAILURE INVESTIGATION – GLARE3-3/2 PATCH – LIMITED FASTENERS	115
TABLE 10-4 TEST CASE B - PARAMETERS	118
TABLE 10-6 TEST CASE F - PARAMETERS	126

LIST OF FIGURES

FIGURE 2-1 MATERIALS USE IN F/A-18 HORNET	5
FIGURE 2-2 COMPOSITE LAMINATE DAMAGE TYPES (HESLEHURST 1996)	7
FIGURE 2-3 TYPES OF PATCH REPAIR (NIU 1992)	15
FIGURE 2-4 TYPICAL BONDED PATCH REPAIR (ARMSTRONG & BARRETT 1998)	16
FIGURE 2-5 MECHANICALLY FASTENED PATCH REPAIR (CHICKEN ET AL 1997)	17
FIGURE 4-1 LOAD TRANSFER IN REPAIR	30
FIGURE 6-1 OFF-AXIS BIAXIAL SPECIMEN (WHITNEY ET AL 1982)	34
FIGURE 6-2 TYPES OF BIAXIAL CRUCIFORM SPECIMENS (OHTAKE ET AL 1998)	36
FIGURE 6-3 CRUCIFORM SPECIMEN (LUCKING 1990)	37
FIGURE 6-4 CRUCIFORM SPECIMEN DIMENSIONS	39
FIGURE 6-5 MATERIAL PROPERTY SECTIONS	41
FIGURE 7-1 STRUCTURAL PROBLEM	44
FIGURE 7-2 SPECIMEN MESH	46
FIGURE 7-3 FASTENER MODEL	48
FIGURE 7-4 LOADING CONDITIONS	52
FIGURE 7-5 BOUNDARY CONDITIONS	53
FIGURE 7-6 MULTI-POINT CONSTRAINT	54
FIGURE 7-7 TEST CASE A	56
FIGURE 7-8 TEST CASE B	56
FIGURE 7-9 TEST CASE C	57
FIGURE 7-10 TEST CASE A FEA MESH	59
FIGURE 7-11 TEST CASE A6 MAXIMUM PRINCIPAL STRAINS – COMPOSITE PLATE TEST SECTION AT 30 KIP	60
FIGURE 7-12 TEST CASE A6 – MAXIMUM PRINCIPAL STRAINS – GLARE3-3/2 PATCH AT 30 KIP	61
FIGURE 7-13 TEST CASE A – 1/8 IN. FASTENERS	62
FIGURE 7-14 TEST CASE A - 3/16 IN. FASTENERS	63
FIGURE 7-15 TEST CASE A - 1/4 IN. FASTENERS	63
FIGURE 7-16 TEST CASE A – STRAIN VARIATION WITH PATCH THICKNESS FOR DIFFERENT FASTENER DIAMETERS	64
FIGURE 7-17 TEST CASE B FEA MESH	65
FIGURE 7-18 MAXIMUM PRINCIPAL STRAINS – COMPOSITE PLATE AT 30 KIP	66
FIGURE 7-19 MAXIMUM PRINCIPAL STRAINS – GLARE3-3/2 PATCH AT 30 KIP	67
FIGURE 7-20 TEST CASE B - 5/32 IN. FASTENERS	68

FIGURE 7-21 TEST CASE B – 3/16 IN. FASTENERS	69
FIGURE 7-22 TEST CASE B – 1/4 IN. FASTENERS	69
FIGURE 7-23 TEST CASE B – STRAIN VARIATION WITH PATCH THICKNESS FOR DIFFERENT FASTENER DIAMETERS	70
FIGURE 7-24 TEST CASE C – FEA MESH	71
FIGURE 7-25 TEST CASE C7 – MAXIMUM PRINCIPAL STRAINS – COMPOSITE PLATE TEST SECTION AT 30 KIP	72
FIGURE 7-26 TEST CASE C7 – MAXIMUM PRINCIPAL STRAINS – GLARE3-3/2 PATCH AT 30 KIP	73
FIGURE 7-27 TEST CASE C – 5/32 IN. FASTENERS	74
FIGURE 7-28 TEST CASE C - 3/16 IN. FASTENERS	74
FIGURE 7-29 TEST CASE C – 1/4 IN. FASTENERS	75
FIGURE 7-30 TEST CASE C – STRAIN VARIATION WITH PATCH THICKNESS FOR DIFFERENT FASTENER DIAMETERS	76
FIGURE 8-1 MTS PLANAR BIAXIAL TEST FRAME WITH CRUCIFORM SPECIMEN	78
FIGURE 8-2 HORIZONTAL FLOATING GRIP	78
FIGURE 8-3 INSTRUNET 100 DATA ACQUISITION MODULE ON TEST FRAME	80
FIGURE 8-4 PHOTOELASTIC EQUIPMENT AND SETUP	81
FIGURE 8-5 TI-MATIC BLIND FASTENER (ANON 1999)	82
FIGURE 8-6 TI-MATIC INSTALLATION SEQUENCE (ANON 1999)	83
FIGURE 8-7 BLIND FASTENER INSTALLATION TOOL MODEL 225	83
FIGURE 9-1 TEST CASE U – GEOMETRY	86
FIGURE 9-2 TEST CASE U – LOADING PROFILE	87
FIGURE 9-3 STRAIN GAUGE LOCATIONS – TEST CASE U	88
FIGURE 9-4 STRAIN GAUGE WIRING – TEST CASE U	88
FIGURE 9-5 TEST CASE U – STRAIN GAUGE RESULTS	89
FIGURE 9-6 TEST CASE U – PHOTOELASTIC FRINGES	90
FIGURE 9-7 TEST CASE U – POST FAILURE	91
FIGURE 9-8 TEST CASE A6 – GEOMETRY	92
FIGURE 9-9 TEST CASE A6 – LOADING PROFILE	93
FIGURE 9-10 TEST CASE A6 – STRAIN GAUGE LOCATIONS ON SPECIMEN	94
FIGURE 9-11 TEST CASE A6 – STRAIN GAUGE LOCATIONS ON PATCH	94
FIGURE 9-12 TEST CASE A6 - STRAIN GAUGE WIRING ON SPECIMEN	95
FIGURE 9-13 TEST CASE A6 – STRAIN GAUGE WIRING ON PATCH (INSTALLED)	95
FIGURE 9-14 TEST CASE A6 – PHOTOELASTIC COATING ON PATCH (INSTALLED)	96
FIGURE 9-15 TEST CASE A6 – STRAIN GAUGE R1	97
FIGURE 9-16 TEST CASE A6 – STRAIN GAUGE R2	97

FIGURE 9-17 TEST CASE A6 – STRAIN GAUGE R3	98
FIGURE 9-18 TEST CASE A6 – STRAIN GAUGE R4	99
FIGURE 9-19 TEST CASE A6 – STRAIN GAUGE R5	99
FIGURE 9-20 TEST CASE A6 – PHOTOELASTIC FRINGES	100
FIGURE 9-21 FEA (LEFT) AND PHOTOELASTIC FRINGES UNDER 20 KIP LOADING	101
FIGURE 9-22 TEST CASE C10 – GEOMETRY	102
FIGURE 9-23 TEST CASE C – LOADING PROFILE	103
FIGURE 9-24 TEST CASE C10 – STRAIN GAUGE LOCATIONS ON SPECIMEN	104
FIGURE 9-25 TEST CASE C10 – STRAIN GAUGE LOCATIONS ON PATCH	104
FIGURE 9-26 TEST CASE C10 – STRAIN GAUGE WIRING ON SPECIMEN (INSTALLED)	105
FIGURE 9-27 TEST CASE C10 – STRAIN GAUGE WIRING ON PATCH (INSTALLED)	105
FIGURE 9-28 TEST CASE C10 – INSTALLATION STRAINS	106
FIGURE 9-29 TEST CASE C – STRAIN GAUGE R1	107
FIGURE 9-30 TEST CASE C – STRAIN GAUGE R2	107
FIGURE 9-31 TEST CASE C – STRAIN GAUGE R3	108
FIGURE 9-32 TEST CASE C – STRAIN GAUGE R4	109
FIGURE 9-33 TEST CASE C – STRAIN GAUGE R5	109
FIGURE 10-1 TEST CASE B – GEOMETRY	117
FIGURE 10-2 TEST CASE B – STRAIN GAUGE LOCATIONS ON PATCH	118
FIGURE 10-3 TEST CASE B – LOAD PROFILE	119
FIGURE 10-4 TEST CASE B – STRAIN GAUGE WIRING (INSTALLED)	119
FIGURE 10-5 TEST CASE B – INSTALLATION STRAINS	120
FIGURE 10-6 TEST CASE B – STRAIN AND LOAD VS. TIME	121
FIGURE 10-7 TEST CASE B – POST-FAILURE	122
FIGURE 10-8 TEST CASE F – GEOMETRY	123
FIGURE 10-9 TEST CASE F FEA MESH	124
FIGURE 10-10 TEST CASE F – MINIMUM PRINCIPAL STRESS (KSI) NEAR FASTENER HOLE AT 20 KIP	125
FIGURE 10-11 TEST CASE F – LOADING PROFILE	126
FIGURE 10-12 TEST CASE F – STRAIN GAUGE LOCATIONS ON SPECIMEN	127
FIGURE 10-13 TEST CASE F – STRAIN GAUGE LOCATIONS ON PATCH	127
FIGURE 10-14 TEST CASE F – INSTALLATION STRAINS	128
FIGURE 10-15 TEST CASE F - LOAD AND STRAIN VS. TIME	129
FIGURE 10-16 TEST CASE F – STRAIN GAUGE R1	130
FIGURE 10-17 TEST CASE F – STRAIN GAUGE R2	131
FIGURE 10-18 TEST CASE F – STRAIN GAUGE R3	131
FIGURE 11-1 TITANIUM ALLOY – GLARE3-3/2 COMPARISON	133

1 INTRODUCTION

Modern military aircraft make extensive use of advanced composite materials such as carbon and aramid fibre-based laminates. The composite structure provides a high specific strength and stiffness alternative to traditional metals, such as aluminium and titanium. With the introduction of advanced composites, new procedures have been developed for repair of composite structures.

For both metallic and composite repairs, the repair methodology chosen is situational. If the damaged aircraft returns to a forward repair station, it must be repaired adequately either to return to operation or to be flown to a depot for permanent repairs. This field repair situation requires quick, simple repairs, which can be accomplished using limited tooling and expertise. Two methods are used to enable the aircraft to remain in service after relatively large-scale damage: bonded or mechanically fastened patch repairs. Both provide an alternate load path by replacing the damaged material.

Bonded repairs require temperature and time-sensitive adhesives and curing agents. These substances require specialised environmental controls and their limited shelf-life causes inventory problems at forward field repair stations. Bonded repair is more suited to depot level repair where all the materials and highly qualified personnel are available. Bonding does produce a more effective repair than mechanical fastening, but it is difficult to implement in field situations.

The riveting or bolting procedure normally involves cleaning up the damage by removing any exposed fibres, then drilling a fastener pattern in the composite and in an appropriately sized patch. The patch is attached using rivets or bolts depending on accessibility. The patch material is chosen based on strength, stiffness, weight, cost and availability.

Traditional patch materials include titanium and aluminium alloys, and various fibre-based composites. Ideally, the same material as the existing structure is desirable to avoid compatibility problems. This is not always possible or practical, so typically titanium is used. Titanium is expensive and requires special machining precautions. An alternative is needed which is easily machined, handled, stored and which provides adequate performance.

Fibre metal laminates (FML) have been developed to combine the qualities of fibre-based composites and metals for use in aircraft applications. The most popular and well characterised FMLs are several versions of GLARE (GLASS REinforced) which consist of alternating plies of pre-impregnated (prepreg) glass fibre and 2024 aluminium alloy. GLARE combines the well known machine shop properties and strength of aluminium alloys with the excellent fatigue properties of composites.

GLARE was originally developed for aircraft fuselage applications but its properties have proved to be suited to a number of other applications. It has superb impact resistance and burn-through characteristics, which are put to use in cargo floors and baggage containers. It also has been evaluated as a patch repair material for existing aluminium aircraft structures (Fredell 1994).

Recent developments in repair technologies have focused on a technique called "soft patching". By properly selecting the modulus of elasticity of the patch material, a repair can be designed which is sufficient to carry the applied load but avoids attracting additional load by not locally stiffening the repaired area. With a lower modulus than titanium, GLARE is a candidate for composite repair. The performance of different GLARE variants in a typical mechanically fastened composite repair operation was investigated in this research study.

Through a parametric finite element analysis, the behaviour of a GLARE repair patch was evaluated. The effects of patch thickness, fastener diameter and patch

geometry were included in the evaluation. The effectiveness of the repair was defined as the combination of the reduction of strain near the damage after the repair, and the magnitude of the strain near the fastener holes. The main goal of the repair is to reduce the strain near the damage, but in attaching mechanically fastened patch, additional strain is introduced near the fastener holes. An effective repair minimises both of these strains. The finite element model represented a laminated composite plate typical of a CF-18 fighter wing skin damaged by a through-thickness penetration of 1 in. diameter.

The results of the modelling were compared to experimental results of biaxial tension-tension loaded repaired laminates. The complex stress state imposed by the biaxial loading is indicative of the loading of the composite structure of a wing skin.

The following chapter reviews the relevant literature regarding composite repair techniques, specifically focusing on military field repair. Various methods of stress analysis of such a repair are covered in Chapter 3. A general overview of the development of fibre metal laminates is included in Chapter 4. Chapter 6 details the selection and description of the biaxial specimen used in the test program. The specimen is the basis of the finite element model described, along with analytical results in Chapter 7. The experimental test setup and equipment are discussed in Chapter 8. Results of the experimental model verification tests are given in Chapter 9. An investigation of the failure behaviour of a repair design is presented in Chapter 10. The results of all the tests are discussed in Chapter 11, followed by conclusions in Chapter 12 and opportunities for future work in Chapter 13.

2 COMPOSITE REPAIR

The continuing need to attain light weight structures in the aerospace industry has led to the introduction and application of fibre-based composite materials in the industry. Particularly in military applications, the high specific strength of composites is attractive, and can translate into significant performance benefits. The use of the material in the field has pushed the need to develop methods for repair of composite structures (Sennet 1989), which experience vastly different failure modes and damage behaviour when compared with traditional metal alloys.

2.1 LEVELS OF REPAIR

Depending upon the situation, material, and expertise available, two basic levels of military aircraft composite repair design and implementation are outlined: depot and field repair. A depot repair corresponds with a maintenance base operation on a commercial aircraft. Field level is similar to a line station repair where limited resources are available. The following discussion will focus on the military idea of composite repair. Commercial aircraft composite repair problems experienced by operators, such as, British Airways (Armstrong 1986) and manufacturers, such as Aerospatiale (Tropis 1994), allow more time for completion; therefore, the majority are depot repairs.

2.1.1 Depot Repair

A depot is generally characterised as the location of a benign environment where materials and expertise are available to develop and implement a permanent repair to a structure. This can be a military repair station removed from the front lines or a commercial maintenance, repair and overhaul facility (Niu 1992). The controlled environmental conditions permit high quality assurance (QA) controls

during the repair operation. A description of depot capabilities of a United States Naval Aviation repair station is outlined by Radford (1997).

Figure 2-1 Materials Use in F/A-18 Hornet

The structure of the F/A-18 Hornet, Figure 2-1, contains approximately 9% carbon fibre/epoxy composite material by weight. The wing skins, vertical tail, rudders and the all-moveable horizontal tail are constructed of graphite epoxy material. The F/A-18 follows previous aircraft, such as the AV-8B Harrier II, which introduced significant amounts of composite material in fighters. Many of the repair techniques used today were evaluated on the AV-8B in the United States and by the Royal Air Force in the United Kingdom (Chicken et al 1997).

Because of the equipment available at depots, QA can be maintained at a high level. Many composite prepregs must be kept refrigerated to maintain their properties until they can be cured in place. If the component must be removed from the aircraft to be repaired, all the tooling and infrastructure are available including large autoclaves for vacuum curing. At forward field repair stations, these advanced resources are not available.

2.1.2 Field Repair

The amenities of depot operations are not available in field repair stations. Typically, this situation requires a minimum amount of operator skill level, equipment and time (Bohlmann et al. 1981). An aircraft repaired in the field must be able to fly again quickly to either return to service or to travel to a depot. The carrier-based aircraft of the U.S. Navy must be repaired quickly and under limitations of ship-board logistics (Manno 1981). Both the Navy's F/A-18 and AV-8B have been the focus of field repair investigations.

Aircraft structural repair manuals (SRM) are directed towards this "quick-fix" level of repair. Field repairs are generally in-situ with no significant material being removed from the aircraft. This is accomplished by plugging a through thickness hole using epoxy resin or applying a patch over the damaged area. This concept is the basis of the development of aircraft battle damage repair (ABDR) procedures. ABDR has two aspects: (1) Restore the operational capability of the aircraft, but not necessarily the fatigue life, and (2) create the best possible repair in the time available (Chicken et al 1997). Battle damage can occur in many different forms, especially when composite structures are involved. The identification of the type of damage is key.

2.2 COMPOSITE DAMAGE TYPES

Laminated composite materials experience unique types of damage and failure, when compared to metallic materials. The damage evaluation must be accurate in order to apply the appropriate repair scheme. Three of the four main types of composite damage are shown in Figure 2-2: matrix cracks, delaminations, and fibre fracture (Heslehurst 1997). Debonding, which is delamination between two structures, will also be discussed.

Figure 2-2 Composite Laminate Damage Types (Heslehurst 1996)

2.2.1 Matrix Cracks

Failure within a lamina (ply) is referred to as matrix cracking. Fatigue cracks in metal alloys initiate and grow from a stress raiser or flaw in the material. Matrix cracking is more complex and does not follow this mechanism (Mall 1997). Intralaminar cracks do not have an adverse effect on the strength of the laminate; however, if they reach significant density, a stiffness reduction will occur. Cracks in an interior lamina are difficult to detect, but surface cracks are visible and should be repaired as necessary.

If dissimilar materials are placed in contact, thermal expansion coefficient differences may result in cracking (i.e. composite to aluminium) as one material expands or contracts at a rate different than the other. Impacts, such as foreign object damage or tool drops, may also initiate cracking.

2.2.2 Delaminations

If component lamina of the laminate separate from each other, delamination has taken place. This may occur simultaneously between different laminae or be isolated to a single interface. Delamination can affect the integrity of the structure by reducing stiffness. The structure will lose strength in compression due to lamina buckling.

Delamination can occur as a result of manufacturing defects or in-service damage (Armstrong & Barrett 1998). Various examples include incorrect cure cycles, time-expired material, impact, overtightening of fasteners, moisture ingress, or improper drilling. All of these events may occur within the life of the laminate.

2.2.3 Debonds

Debonding is a defect unique to adhesively bonded joints. In contrast to delaminations, debonding occurs between two distinct structures, not two different laminae. It can occur in repair designs between the adhesively bonded patches and the damaged laminate. Generally, debonding only occurs due to manufacturing defects, impact damage, or long-term fluid penetration (Armstrong & Barrett 1998).

During in-plane loading, a debonded region is only a concern if it is larger than the area of the bonded joint where the adhesive remains elastic. This would load the elastic area to a higher level, as load originally carried by the debonded area is shed. The overload may cause the bondline to undergo excessive plastic deformation. Stress concentration may occur in the adhesive as load is forced to bypass the debonded area (Helsehurst 1997).

2.2.4 Fibre Fracture

If loading reaches the ultimate strength of the laminate or the laminate is subjected to high energy loading, i.e. impact, then fibre fracture may occur. The fracturing may be accompanied by localised delamination and debonding between the fibre and the matrix. The debonding can result in a loss of stiffness along with a reduction in strength due to loss of fibre continuity.

The reduction of strength due to the damage, referred to as the notched strength, is an important parameter in repair. If the notched strength is still acceptable in terms of the structure, limited work is needed. If the strength is reduced below design levels, then a more substantial repair must be done.

2.3 REPAIR CRITERIA

Once the damage has been identified and classified, a repair method must be selected. Small cracks, penetrations, or cosmetic damage can be repaired with adhesive filler through resin injection; however, patch repairs must be employed for major damage. Patches attempt to restore the ultimate strain allowables in the damaged laminate (Hoskin & Baker 1986). The selection of patch attachment method, be it bolted or bonded, and patch material is dependent on the following considerations as presented in the Northrop Advanced Composite Repair Guide (Anon 1982) and also by Niu (1992) and Heslehurst (1997).

- Static strength and stability
- Durability for the life of the part
- Stiffness requirements
- Aerodynamic smoothness
- Weight and balance
- Operational temperature

- Environment
- Related airframe systems
- Cost and schedule
- Multi-site Damage
- Stealth

The most important of these considerations are discussed in the following sections.

2.3.1 Static Strength and Stability

In order to allow a damaged aircraft to take to the air again safely, the design strength of the damaged structure must be restored. The repair design must replace the removed material with a structure that will allow the airframe to undergo its allowable or ultimate design strain. At design ultimate load, the repaired structure should be able to support gross strains in excess of 4000 microstrain (Bohlmann et al 1981). This criterion has been applied to military repair programs for the AV-8B Harrier II and F/A-18 Hornet which have composite laminate wing skins up to 0.50 in. thick.

The stability of the structure is critical in compressive loading environments. The stiffness should be restored to its original value so that buckling instability is avoided.

2.3.2 Durability

By definition, a field repair will have a relatively short life. This does not preclude the application of sound damage tolerance methods in repair design. High stress concentrations should be avoided, if possible, by rounded patch edges or filleting corners. Also, a firm lifetime limit should be established on the repair.

This time limit will depend on the susceptibility of the repair to such effects as cyclical loading, corrosion and environmental degradation (Heslehurst 1997). If the structural repair is in a highly dynamic area, i.e. the wing, then the design life will be lowered. If the repair material has a high galvanic potential difference with the parent laminate, then corrosion can reduce the life-span of the repair unless it is properly protected. If the repair does not provide a proper seal, other environmental factors such as temperature, and especially moisture, can cause degradation of adhesive and cause damage to the original structure. The patch should not allow moisture into a delaminated or debonded area.

2.3.3 Stiffness

Stiffness restoration is often more important than strength in the repair of lightweight structures such as aircraft (Heslehurst 1997). Problems can arise if the patch material is too stiff as it will attract additional load into the repaired area; however, the stiffness must be sufficient to carry the load of the original structure (Fredell et al 1995). Generally, a repair should maintain the existing structural stiffness of the aircraft (Anon 1982).

In flight-critical locations, such as wings and control surfaces, deflection limits must be maintained. A loss of stiffness may affect the flying characteristics of the aircraft. Also, by changing stiffness, aeroelastic parameters are altered. This could result in premature flutter and handling difficulties.

2.3.4 Aerodynamics

Aerodynamically critical locations, such as wing leading edges and engine nacelles, must remain smooth. This entails utilising flush repair techniques to ensure the aircraft performance is not significantly degraded. In less critical areas, external patches may be applied.

During initial manufacturing, aerodynamic smoothness is specified and held to tight tolerance of 0.005 in. in laminar flow areas. This includes leading edges of the wings and control surfaces; however, in field repair conditions, an upper limit of 0.15 in. above the skin is recommended (Baker 1997). Flush scarf joint repairs are also appropriate in aerodynamically critical areas, but the operation is time consuming and labour-intensive. External patches can be tapered to reduce the disruption of the airflow; however, the effect on the drag penalty is minimal (Kind et al 1998).

Following the field repair, the aircraft may only need to fly one more flight to return to a depot, so the aerodynamic penalties incurred by a mechanically fastened external patch may be worth the trade-off of repair complexity and time.

2.3.5 Weight and Balance

Typically, a structural repair does not significantly affect the overall weight of the aircraft and consequently its performance. However, if the repair is in a mass-balance critical location, such as a flight control surface, tight design limits must ensure the repair does not affect the control characteristics of the aircraft.

This may also be applicable to rotating parts, such as rotor blades and flaps. Weight changes will affect their dynamic characteristics, which may adversely affect overall performance.

2.3.6 Temperature and Environmental Effects

In designing the repair, the operating temperature of the aircraft must also be considered. Typically, aircraft operate between -54°C and 71°C ; however, higher temperatures can be reached on leading edges of supersonic aircraft.

The repair must also be able to provide the required strength and stiffness under these conditions.

When selecting a patch material, it is prudent to select compatible thermal expansion coefficients. If the dissimilar materials expand or contract at different rates, thermal stresses are induced in the repair. Bonded repairs are much more susceptible to environmental conditions such as humidity and moisture ingress than bolted designs. The hot/wet condition should be satisfied by the material selection. If the repair is bonded to a wet surface, the bondline can be porous or the parent laminate may blister when attempting a cure (Anon 1982).

2.3.7 Cost, Personnel and Schedule

Skilled personnel, material, equipment cost, and schedule can all determine the extent of the repair. Under field conditions where resources are limited, the repair must be completed in as short a time as possible so the aircraft can return to operational status or be ferried to a rearward repair depot. Many sophisticated bonded repair techniques are therefore excluded from application.

Co-cured patches require autoclave conditions to achieve high quality bonding. Field operations usually use vacuum-bag curing which produces much lower mechanical properties. Portable autoclaves have been developed to allow higher quality curing in the field but still require highly skilled personnel to operate (Cramer & Purcell 1996). Prepreg material has a limited shelf-life and must be stored at low temperatures until ready for use. In contrast, the procedures for bolted repair are very similar to repairing metallic structures and therefore can be accomplished using readily available and easily maintainable materials.

2.4 REPAIR OPTIONS

Depending on the extent of damage to the composite structure, two general approaches can be taken. Less significant damage can be repaired through the addition of adhesive or resin to plug holes. Major damage, such as penetration or extensive delaminations, must be patched to provide an alternate load path through the structure.

2.4.1 Non-Patch Repairs

If there is no extensive internal delamination or cracking (verified through non-destructive investigation), then minor indentations can be filled with resin and allowed to cure (Baker 1997). Essentially, this is a cosmetic operation; it restores the original surface profile and can improve aerodynamic performance. If a fastener hole is elongated, then a machinable resin can be used to fill the hole, then it is redrilled and the fastener is reinserted.

To repair internal delaminations, resin can be injected into the laminate to restore the interlaminar strength. This procedure is only valid if the delamination is due to in-service damage, not contamination. Two holes are drilled at the edge of the mapped delaminated region and resin is injected into one until it fills the other (Armstrong & Barrett 1998). Wetting and flow problems of the resin limit the application of this method.

2.4.2 Patch Repair

Significant damage must be repaired by patching. A doubler, scarf, or flush patch placed over cleaned damage allows an alternate load path in place of the structure that was damaged. The patch can be bonded or mechanically fastened, Figure 2-3.

Figure 2-3 Types of Patch Repair (Niu 1992)

2.4.2.1 Bonded Patch Repair

The obvious advantage to a bonded repair is no additional damage to the structure through fastener holes; however, it also requires a higher degree of skill for application, a longer time to complete, and shelf-life limited materials. External bonded patch repairs are generally restricted to laminate thicknesses less than 0.080" (Niu 1992).

The most efficient bonded repair option is the scarf-repair, Figure 2-3, which involves removing material from the laminate to create a scarf (1:30) surface. The hole is then filled incrementally by a series of prepreg layers corresponding to the lay-up of the original part. The repair must then be cured-in-place using vacuum bags and heat application methods such as heating blankets or hot air guns. It results in a flush surface which does not affect the aerodynamic

smoothness of the surface or in the case of military aircraft, the radar cross-section. The complexity of the operation restricts its application to a depot level repair. Externally bonded repairs are more common in the field. A simple design is shown in Figure 2-4.

Figure 2-4 Typical Bonded Patch Repair (Armstrong & Barrett 1998)

The patch material for a bonded repair can range from precured composite material, such as boron or carbon/fibre epoxy, to alloys of titanium or aluminium. Storage of prepreg material in the field usually requires low temperatures. This can be limited in advanced locations. Bonded repairs restore higher strength levels to a damaged laminate than a similar mechanically fastened repair; however, their application under field level conditions is hampered by a need for highly skilled personnel and specialised equipment.

2.4.2.2 Mechanically Fastened Patch Repair

Using familiar and available materials and tooling, a mechanically fastened repair to a damaged laminate can be achieved quickly and it restores the design strength of the structure. Its lack of complexity and requirements for highly skilled operators suits the field repair scenario.

A mechanically fastened repair is simple. The damage is drilled out to create a circular or elliptical hole in the laminate. A fastener hole pattern, including pitch and diameter, is selected based on design guidelines or a computer-based expert system (Venkayya et al 1987). The patch is machined and used as a template to drill the corresponding fastener pattern in the damaged structure. Depending on the location on the aircraft, backside access may be limited. In that case, blind fasteners must be employed to attach the patch to the laminate, Figure 2-5.

Figure 2-5 Mechanically Fastened Patch Repair (Chicken et al 1997)

The patch material and thickness is also a variable in the design. The extensional stiffness (Young's Modulus x thickness) of the patch is normally selected to match the existing laminate. This approach allows thin, stiff material to be used. Titanium and aluminium alloys are popular, as are precured composite patches. Patch material is selected based on stiffness requirements, machinability and availability.

Generally considered to be the simpler and quicker option for composite structural repair, mechanically fastened patches can be used in most situations. This option does not involve the use of any adhesives or curing agents and therefore is not affected by temperature or humidity, a substantial problem in bonded repairs. When selecting a patch material, the coefficient of thermal expansion (CTE) is key to obtaining efficient load transfer through the patch. Unequal expansion or contraction of the patch and repaired structure may cause residual strain in the design.

Due to the lack of adhesives and other chemicals in the procedure, surface preparation is not as critical which can be key in short time repairs. The techniques used are relatively simple and can be implemented by a lower level of labour or tooling. Backside access is not necessary when the use of blind fasteners is incorporated. Mechanically fastened repairs are generally reserved for thick laminates rather than thin or honeycomb structures.

Obviously, mechanically fastened patches introduce new holes in a highly loaded structure. This may limit the life of the part or necessitate a more substantial repair down the road. The addition of numerous fasteners and a relatively thick patch can add a significant weight to the aircraft, which may affect performance or stability. Unless the patch is countersunk into the laminate, it will protrude and disrupt the aerodynamic smoothness of the skin. Care must be taken in the

drilling of the holes in the composite to prevent delamination and fibre damage.
The same care must be applied to the machining of the patch material.

3 COMPOSITE REPAIR STRENGTH PREDICTIONS

A relatively simple method to predict the behaviour of a mechanically fastened repair would be extremely useful in the field, where time and computational capabilities are limited. Since the introduction of composite materials in military aircraft, extensive work has gone into analysis of field repair. A number of different approaches have been taken; however, they basically follow the same three steps:

1. Determination of load distribution.
2. Determination of the stress field surrounding the damaged region and fastener holes.
3. Prediction of the failure of the mechanically fastened joint along with failure modes.

3.1 LOAD DISTRIBUTION

Calculating the load distribution among fasteners in the repair design can become very complex as the number of fasteners increase. Significant research has been done to incorporate numerical models into computer-based analytical packages that compute fastener load in bolted joints. Venkayya (1987) and Gali (1993) offer a description of the more prominent packages and their capabilities. It is possible to estimate loading for small numbers of fasteners as described by Xiong (1998) and Her & Shie (1998). Heslehurst (1990) outlines a procedure to accomplish a simple repair using a handheld calculator. Once the load on each fastener is known, the stress field determination can proceed.

3.2 STRESS FIELD DETERMINATION

Their inherent anisotropy makes the stress analysis of composite material components more complex than for metal. On the other hand, it is this anisotropy

that allows structural designers to tailor the laminate to provide strength in the desired directions. The effect of discontinuities such as open and pin-loaded holes on the stress field in the laminate are discussed in the following sections.

3.2.1 Open Hole Stress Field

The stress field surrounding an open hole in a composite laminate can be estimated fairly easily using a number of different approaches. The complex variational theory of Lekhnitskii (1968) has formed a basis for many different methods. Essentially, complex parameters are determined for the material based on elements of the laminate compliance matrix. These parameters define the degree of anisotropy in the plate.

The elastic properties of the plate can be found through Classical Laminated Plate Theory (CLPT). The buildup of individual laminae oriented at various angles produces the laminate (Tsai & Hahn 1980). If the elastic properties of each lamina are known then laminate stiffness and compliance matrices can be assembled. From these matrices, the overall laminate properties are obtained.

In a uniaxial problem, the exact solution is provided by Whitney & Nuismer (1974). The stress concentration factor at the edge of the hole is given by the relationship of the laminate compliance matrix elements. The factor is then used with the radius of the hole to provide a normal stress distribution about the hole.

The biaxial problem requires the complex variational approach. Grezczuk (1971) extended the anisotropic plate theory of Lekhnitskii to composite laminates. The governing differential equation is solved in terms of the Airy stress function. The solution gives the stress around the circumference of the hole due to applied uniaxial, biaxial or shear stresses. Further improvements were made by Tan (1987a) who offered a solution of in-plane stresses around an elliptical hole that

is angled to the principal loading directions. The notched and the unnotched stresses are combined to provide the overall stress state. The solution is valid at a characteristic distance from the hole edge where free edge effects, such as intralaminar shear stresses, are negligible. However, for a complete picture of the stress in a repair, the effect of adding a fastener to the open hole must be considered.

3.2.2 Bolt Loaded Hole Stress Field

The problem of a mechanically fastened repair can be narrowed down into a single hole loaded by a bolt. If the load distribution in the laminate is known, then an analysis can be performed on each fastener in the design. De Jong (1977) utilised the superposition principle to estimate the effect of a loaded hole on the stress field in the laminate. By applying a load distribution over half the circumference of the hole, de Jong simulated a hole in bearing. This was combined with the effect of the hole cut-out in the laminate discussed in Section 3.2.1, to provide a solution. This solution applies only for the uniaxial case. Xiong (1998) proposed a stress analysis method that accommodates the biaxial loading condition using the complex variational approach. The bearing load transferred to the laminate by the fastener is represented by a cosine distribution over the half-circumference of the hole. This force is always in equilibrium with the applied loads.

The application of the finite element method (FEM) has allowed the determination of stress distribution around the loaded fastener holes under different loading conditions. Different approaches have been used in the modelling of the fastener-hole interaction. Agarwal (1980) used a fixed radial displacement boundary condition to simulate a rigid close-fit pin. Chang et al (1982) and Her & Shie (1998) used the normal cosine distribution to represent

the contact pressure. Both approaches showed adequate results, but could not accurately describe the contact area in a flexible fastener under complex loading.

Bellinger & Poon (1990) demonstrated the use of one-dimensional spring elements to model the clearance between the fastener and the hole, in the uniaxial loading case. Harrison (1998) utilized similar methods in biaxial bypass/bearing investigations. Shi et al (2001) used spring elements (referred to as gap elements within the NASTRAN finite element software) in a bolted patch repair investigation. The use of gap elements allowed a realistic contact angle to be established for multiple fastener patterns under complex loading.

3.3 FAILURE PREDICTION

Once the stress field around the fastener hole is determined, the point and mode of failure can be determined. Most of the composite laminate failure criteria are ply based. If the stress in a ply satisfies the failure condition, then failure of the laminate is predicted. Quadratic failure criteria, from Tsai-Hill and Hoffman, allow for stress interaction and in the latter, differing compressive and tensile strengths. These first-ply-failures (FPF) are difficult to detect experimentally (Whitney et al 1982), as they only manifest as localised stiffness changes.

The Yamada-Sun (1978) criterion predicts last-ply-failure (LPF). This failure is much easier to detect during experimental testing. Essentially, the entire laminate has failed. Yamada-Sun has been used by a number of authors, namely Chang et al (1984) and Her & Shie (1998), to detect failure in bolted joints of composites; however, these were uniaxial investigations and the Yamada-Sun criterion is limited to this case.

4 FIBRE METAL LAMINATES

Fibre Metal Laminates (FMLs) are a set of hybrid materials consisting of alternating thin layers of metal alloys and prepreg fibre/epoxy layers (Ford 1997). Originally developed as highly damage tolerant materials for aircraft thin walled structures, FMLs have been found to be superior in other aerospace applications.

4.1 GENERAL HISTORY

Due to the mandating of the damage tolerance philosophy of aircraft structural design, a material that showed very slow fatigue crack growth and maintained the strength of traditional aluminium alloys was needed. Development of a highly fatigue resistant material was begun in the late 1970s at the Faculty of Aerospace Engineering at the Delft University of Technology, The Netherlands (Asundi & Choi 1997). The first type of FMLs was a hybrid of aluminium and aramid fibre preregs, ARALL. The result was a material that had superior fatigue crack performance and lower density, along with comparable strength in relation to monolithic aluminium alloys.

In 1990, a new class of FMLs were introduced. GLARE (GLAss REinforced) replaced the aramid/epoxy layers with high strength fibreglass (R & S2). The glass layers can be oriented in different directions creating a material tailored to certain loading conditions (Krishnakumar 1993). The glass fibre also adheres better to the epoxy to create a more reliable and applicable material.

GLARE is available in a number of commercial varieties, Table 4-1. Thicker GLARE is manufactured by adding aluminium and prepreg plies to the laminate. For example, GLARE3-5/4 is a cross-ply variant with five layers of aluminium and four of glass fibre prepreg. In the layup, the aluminium layers are always on the exterior.

Table 4-1 Commercially Available GLARE (Asundi & Choi 1997)

	GLARE1	GLARE2	GLARE3	GLARE4
Aluminium Alloy	7075-T6	2024-T3	2024-T3	2024-T3
Thickness (aluminium)	0.0085" – 0.016"	0.0085" – 0.016"	0.0085" – 0.016"	0.0085" – 0.016"
Thickness (fibre)	0.010"	0.010"	0.010"	0.015"
Fibre Orientation	Unidirectional	Unidirectional	Cross-ply 50/50	Cross-ply 67/33

The properties of GLARE have proved to be attractive for numerous applications such as cargo containers, floor panels, and other secondary aircraft structures.

4.2 MATERIAL PROPERTIES

The material properties that make GLARE such an attractive alternative material for aircraft construction range from a lighter weight than aluminium to excellent flame penetration characteristics. Table 4-2 compares them with a similar thickness monolithic aluminium alloy 2024-T3.

Table 4-2 GLARE Material Properties – All layups 3/2 (Anon 1997)

	Direction	Units	GLARE1	GLARE2	GLARE3	GLARE4	2024-T3
σ_{uts}	L	Ksi	185.9	176.1	104.0	148.9	66.0
	LT	Ksi	51.1	46.0	103.8	88.0	65.0
$(\sigma_y)_{ten}$	L	Ksi	79.0	52.2	44.2	51.1	52.1
	LT	Ksi	49.0	33.1	41.0	37.0	47.0
E_{ten}	L	Ksi	9.4	9.6	8.4	8.3	10.4
	LT	Ksi	7.3	7.3	8.4	7.3	10.4
$(\sigma_y)_{comp}$	L	Ksi	61.5	60.0	44.8	52.9	43.9
	LT	Ksi	58.4	34.2	44.4	41.3	50.0
E_{comp}	L	Ksi	9.7	9.7	8.7	8.7	10.7
	LT	Ksi	7.4	7.5	8.7	7.8	10.7
ρ		lb/in ³	0.091	0.091	0.091	0.089	0.095

 σ_{uts} – ultimate tensile strength $(\sigma_y)_{ten}$ – tensile yield strength E_{ten} – tensile Young's Modulus $(\sigma_y)_{comp}$ – compressive yield strength E_{comp} – compressive Young's Modulus ρ - density

L – longitudinal

LT – longitudinal transverse

4.2.1 Fatigue

The static strength and stiffness of GLARE are very encouraging; however, GLARE also has excellent fatigue properties. By “bridging” a fatigue crack, the fibres severely limit the crack growth rate in the material (Takamatsu et al 1999). A through-thickness crack is impeded by the interlaminar adhesive (Lawcock et al 1995). This dissipates much of the crack's energy, thereby impeding the speed of its progression. Under realistic loading conditions, GLARE exhibits crack growth rates ten to 100 times less than aluminium (Vlot et al 1999). GLARE has shown to have a much slower crack growth rate than monolithic aluminium. This

translates into a much longer life span for the material and a reduction in traditional “crack-stopping” elements such as doublers, in aircraft structure.

4.2.2 Environmental Durability

Under less than ideal environmental conditions, the structural performance of advanced composite materials, such as carbon fibre laminates, degrades fairly quickly. In particular, moisture ingress affects the bond strength between the plies which can be a cause of delamination. The fibre layers of FMLs are only exposed to moisture at the laminate edges so the influence of moisture is limited (Beumler & Gunnink 1995).

To prevent corrosion, the aluminium layers of GLARE are anodized and coated with a corrosion-inhibiting primer prior to bonding. The outer layers can also be clad for additional protection. The fibre layer provides a barrier to any through-the-thickness corrosion (Vlot et al 1999).

4.2.3 Impact Resistance

Impact damage in carbon fibre composites is difficult to detect. Impacts from such debris as hail, stones from taxiing, bird strikes, or tool drops may not be visible but may result in internal delaminations, detectable only using picked up using non-destructive inspection (NDI). In contrast, impact damage in FMLs should be visible through dents in the external aluminium layers making inspection much more efficient (Vlot et al 1999).

GLARE has shown higher resistance to cracking after impact than 2024-T3. The fibres exhibit a high strain rate strengthening and, with high failure strains the fibres can provide very high impact resistance. A crack in impacted GLARE will only grow in the inspectable outer layers of aluminium.

4.2.4 Flame Penetration

Another benefit of the hybrid materials is fire protection. Monolithic aluminium can melt away relatively quickly in intense heat or flame conditions. In GLARE, the first glass fibre layer acts as a flame stopping barrier once the outer aluminium layer has been melted away (Hooijer 2001). The glass has a much higher melting point which corresponds to a longer burn-through time when compared with similar thickness aluminium. The long burn-through time can translate into longer allowable time for passenger evacuations.

The GLARE can also act as a protection for the internal structure of the aircraft if used as an exterior skin, which can ensure fuselage integrity for a longer duration.

4.3 APPLICATIONS

Originally developed as an improved damage tolerant material for aircraft structures, GLARE has found a number of other applications in the aerospace industry. Its flame resistance and impact strength properties make it ideally suited for cargo container construction. It has also been installed on the Boeing 777 cargo bay floors. ARALL has been used in the C-17 Globemaster III heavy-lift aircraft for the cargo door. It resulted in a 26% weight saving over traditional aluminium alloys.

The main focus of GLARE application in the future will be towards large aircraft. Airbus has recently chosen GLARE as the material for the upper crown of the ultra-high capacity A380 (Dornheim 2001). This would be the first commercial use of GLARE in primary structure (Vogelgesang & Vlot 2000).

4.4 REPAIR

The outstanding fatigue behaviour of GLARE along with its simple machining properties (Beumler & Gunnink 1995), make it an excellent candidate as a patch material for aircraft structures made from metals composites or FMLs themselves. The developed and time-proven techniques of repair of aluminium aircraft structures can be applied to GLARE.

4.4.1 Soft Patching

The use of GLARE as a patch material for traditional aluminium alloy skins has been investigated by Fredell (1994). He proposed the procedure of "soft patching" whereby a structure is repaired with a material that has just enough stiffness to carry the necessary load and does not attract load from the surrounding structure.

As the load approaches the damaged region, Figure 4-1a, it is forced to travel around the damage, causing a stress concentration around the hole. By attaching a patch over the damage, the load is given an alternate route as it can now travel over the damage. If the patch is too stiff, Figure 4-1b, load that would normally bypass the damaged region is attracted into the repaired area. This may overload the repair or cause fatigue problems. A soft patch, Figure 4-1c, is just stiff enough to allow the load to bypass the damage and not attract load from the surrounding structure. If the patch is not stiff enough, load continues to pass through the damage and the stress concentration remains as in Figure 4-1a.

Figure 4-1 Load Transfer in Repair

In addition to the appropriate stiffness, the patch material must have sufficient static strength to carry its share of the load and be fatigue resistant. With multiple variations of stiffness, comparable strength and high fatigue resistance, GLARE is a candidate for a soft patch material. Fredell (1994) suggests a change in repair philosophy from "repair to equal or better static strength" to "strength restoration with minimal impact on durability, damage tolerance or inspection." This implies that in designing a repair, the effects of stress risers, crack growth properties of the material, and inspection intervals should be considered even for simple or field level repairs.

Fredell et al (1995) also investigated GLARE as a bonded crack-stopping material. They found that crack re-initiation required hundreds of thousands of flights or manual re-introduction. Guijt (1995) reports on the installation of GLARE bonded patches on cracked portions of a C-5A fuselage. By using GLARE to patch fatigue cracks, re-skinning the aircraft was avoided which constituted a significant cost savings.

5 PROJECT DEFINITION

The short-term field repair of aircraft composite skins is usually accomplished by mechanically fastening a patch over the damaged area. Aluminium alloys, other composites and most commonly, titanium alloys, are used as the patch material. The major objective of this project was to evaluate the fibre metal laminate, GLARE, as a moderate modulus, lightweight, relatively strong, easily worked alternative to these conventional patch materials.

5.1 OBJECTIVES

The project was developed to fulfil the following objectives:

1. Evaluate GLARE as a mechanically fastened patch material.
2. Develop a design procedure for repairing composites with GLARE.

5.2 APPROACH

The objectives were approached by using finite element analysis to evaluate a number of variations of repair configurations. Parameters such as patch shape, thickness and fastener diameter were varied to investigate the effect on repair effectiveness. The finite element model was then compared to experimental measurements to verify the analytical procedure.

The finite element model was developed in two-dimensions using explicitly modelled fastener-hole interactions which allowed strain concentrations around the holes to be quantified. Composite failure theories were applied to the composite plate and isotropic material failure limits were placed on the GLARE patch. Test cases were developed to promote bearing failure in both constituents of the repair.

The model was based on an existing biaxial cruciform test specimen that was used in a previous bearing strength research project by Harrison (1998). The specimen represents a carbon fibre/epoxy wing skin laminate. Damage to the laminate was approximated by drilling large diameter (1 in. to 5 in.) holes through the test section. The damage was then repaired using GLARE patches that had been analysed through the finite element method. The specimens were assembled using blind fasteners and instrumented to allow comparison of strain gauge readings to finite element predictions.

A detailed description of the test specimen follows. The specimen is the basis of the finite element model of Section 7. The experimental verification tests are described in Section 8 and 9. The failure behaviour of the repaired specimens is investigated in Section 10.

6 SPECIMEN DESCRIPTION

The cruciform biaxial specimens used in the investigation had been developed to create a uniform strain field in the test section under equal biaxial loading (Harrison 1998). The specimen construction and layup were tightly controlled to ensure proper load introduction. The following section describes other biaxial test specimens that have been used and discusses the suitability of the existing specimen for this composite repair investigation.

6.1 OFF-AXIS SPECIMEN

A cost-effective method to induce a biaxial stress state in a directional-sensitive composite is to apply the loading at an angle offset from one of the material principal axes. The offset angle results in shear stresses in the material (Harrison 1998). The stresses, once resolved into the material axes, create a biaxial stress state, Figure 6-1.

Figure 6-1 Off-axis Biaxial Specimen (Whitney et al 1982)

Although simple and inexpensive, the off-axis specimen is limited in its application. The induced biaxial normal stresses will always be of the same sign: always tension-tension or compression-compression. Independent variation of individual biaxial components is not possible (Kobayashi 1993). In order to achieve a significantly large gauge (test) area, the specimen must be long enough to separate the test section from grip effects. This arrangement is simple and reliable, but unsuited to a larger scale repair investigation.

6.2 THIN-WALLED TUBULAR SPECIMEN

The use of internal pressure and axial loading on a tubular specimen can induce biaxial stresses in the wall of the tube. This specimen has the widest range of variation and allows the application of any biaxial ratio. The load introduction can be a complex problem. Tabs and end caps are required on the ends of the specimen to ensure the proper pressure sealing and axial load application. The presence of these discontinuities in the structure may cause stress concentrations that can result in failure outside the test section.

The introduction and control of internal and external pressure, grip pressure, axial load, and torque requires a complex system (Kobayashi 1993). The grip system must be designed especially carefully to ensure the system experiences the intended loading.

In a composite repair program, the patch repair of a curved panel would increase the complexity of the investigation. The range of loading provided by a pressurised setup is not necessary, and the complexity of the system is prohibitive.

6.3 FLAT PLATE CRUCIFORM SPECIMEN

A cruciform shaped test specimen has been used in many plane stress investigations. By applying load to each of the arms, the test section sees a biaxial stress state. The load can be varied so non-proportional loading is possible. If the principal material axes are rotated, then a variety of biaxial stress states can be achieved.

Figure 6-2 Types of Biaxial Cruciform Specimens (Ohtake et al 1998)

Creating the uniform stress field in the test section of the specimen is a persistent problem. The arms of the specimens can be a weak point or be too stiff and attract too much load. The three options presented in Figure 6-2 attempt to change the structural characteristics of the arms. The cut type reduces the cross section of the arms. The second option creates a thinner test section, and the third adds slots to the arms to reduce their stiffness. All the options have merit but the slotted specimen provides the required reduction in axial stiffness with the least reduction of axial strength.

In composite materials, an additional option is available, loosely based on the slotted arm design. Lucking (1990) developed such a cruciform specimen for AS4/3501-6 carbon fibre laminates. The arms consist of uniaxial plies which extend into the test section to create the quasi-isotropic layup. Plies not oriented in the loading direction are truncated. In this manner, the test section sees a

uniform stress field and the arms have a significantly reduced stiffness, Figure 6-3. The specimen has been used in bearing strength investigations by Hoa (1995) and Harrison (1998).

Figure 6-3 Cruciform Specimen (Lucking 1990)

This specimen design is suited to available test equipment at NRC-IAR (National Research Council – Institute for Aerospace Research) and could be used in a repair test program. The lack of curvature allows simplified patch design, manufacture, and installation.

6.3.1 Previous Bearing Test Program

As previously mentioned, the specimen described in Section 6.3 was utilized in a bearing/bypass strength investigation by Harrison (1998) at NRC-IAR. The investigation attempted to gather data on the bearing strength of the composite laminate under several different loading environments (biaxial ratios, tension-

compression) using a single, fixed fastener to transfer load in the centre of the test section.

Some specimens were sectioned after testing to observe bearing damage near the fastener hole. 12 specimens remained and contained only localised bearing damage at the centre of the test section. It was decided to use these specimens instead of manufacturing a new batch, to reduce the costs of the project. The localised damage was removed by boring a 1 in. diameter hole as shown in Figure 6-4. This hole represents a clean penetration damage through the laminate.

6.3.2 Cruciform Specimen Dimensions & Properties

In addition to the 1 in. damage, each loading arm of the specimen has four 0.5 in. diameter bolt holes, Figure 6-4. The grips permit attachment to the floating grips of the MTS biaxial load frame described in Section 8.1. The 8 in. x 8 in. test section provides sufficient area for a variety of patch configurations.

Figure 6-4 Cruciform Specimen Dimensions

The test section properties were originally chosen to represent the layup of the F/A-18 wing skin. The 0.25 in. thick, quasi-isotropic area consists of 48 plies of AS4/3501-6 carbon fibre/epoxy prepreg. FM300K adhesive is used in the arms and corner fillets to provide the necessary thickness in the absence of a prepreg ply. The adhesive does not provide stiffness or strength but maintains specimen planarity and thickness from the arms and grips to the test section. The material properties of the plies of the specimen are presented in Table 6-1.

Table 6-1 – Lamina Material Properties (Harrison 1998)

Material	E_{11} Msi	E_{22} Msi	G_{12} Msi	ν_{12}	Thickness inches
AS4/3501-6	21.32	1.31	0.73	0.30	0.0052
FM300K	0.71	0.71	0.0579	0.30	0.0052

Using classical laminated plate theory (CLPT) the above lamina properties can be converted into global material properties, giving the characteristics of the laminate as a whole. Table 6-2 and Figure 6-5 present the laminate properties of different sections of the specimen, as developed by Harrison (1998). Sections A, D, and E consist entirely of AS4/3501-6 plies while Sections B and C are a lay-up of AS4/3501-6 and FM300K adhesive. The symmetric lay-ups of Sections A, D, and E can be represented by the following:

Section A: $[45_2/-45_2/0_2/\pm 45/90_2/\pm 45/90_2/\pm 45/0_2/\pm 45/90/0_3]_s$

Section D: $[\pm 45_2/0_2/\pm 45/90_2/\pm 45/90_2/\pm 45/0_2/90_3/0_3]_s$

Section E: $[\pm 45_2/0_2/90_4/\pm 45/90_2/\pm 45/0_2/90_3/0_3]_s$

As the FM300K plies of Sections B and C are isotropic, the inclusion of the letter F to denote an adhesive ply in the code is sufficient (i.e direction is unimportant). Plies that have a direction are considered to be AS4/3501-6 which is orthotropic:

Section B: $[F_4/0_2/F_{10}/0_2/F_3/0_3]_s$

Section C: $[F_8/90_2/F_2/90_2/F_6/90/F_3]_s$

Table 6-2 Cruciform Section Properties

Section Figure 6-5	E₁₁ Msi	E₂₂ Msi	G₁₂ Msi	ν₁₂	Thickness inches
A	8.89	7.38	3.11	0.34	0.25
B	6.72	0.93	0.25	0.30	0.25
C	0.88	5.00	0.20	0.05	0.25
D	8.79	8.79	2.71	0.25	0.25
E	8.58	10.20	2.32	0.19	0.25

Figure 6-5 Material Property Sections

The cruciform specimen geometry was used as the basis for the finite element model described in Section 7.1. The material properties of Table 6-2, were applied to the element properties in the model. The subsequent experimental investigation compared the finite element results to data gathered from the instrumented specimens.

7 FINITE ELEMENT ANALYSIS

By employing the finite element method (FEM), a structural system can be discretised and analysed using numerical methods. This virtual specimen testing can be accomplished at low cost and provide significant insight into structural behaviour.

The finite element method was used in this test program to investigate the behaviour of a mechanically fastened composite repair. The pre-processor, MSC Patran was used to define the geometry and mesh while MSC Nastran was used as the finite element solver. Multiple parameters, such as fastener diameter, patch thickness and patch shape were varied. The effects of these changes on the loading of the damaged laminate and repair patch were recorded.

The modelling techniques used along with the description of the structural problem follow in Section 7.1. The test program is outlined in Section 7.2. The data produced from the tests and discussion of the results are included in Section 7.3.

7.1 FINITE ELEMENT MODEL DESCRIPTION

Composite field repair is intended to be simple, reliable, and sufficient to restore the static strength of a damaged structure. This has been accomplished in the past by utilising empirically derived guidelines developed for traditional patch materials. More recent computer-based packages such as BREPAIR (Bohlmann et al 1981) and SAMCJ (Harrison 1998), calculate fastener loading to check against joint failure. The majority of the packages cannot accurately simulate complex loading, so simulations are limited to simple cases.

Complex loading is easily handled by the finite element method. Multiple variations of geometric or material parameters can be evaluated before a repair is implemented. This may not be practical in a field situation where the hardware needed to run the finite element software would not be available. The purpose in this investigation is to determine the behaviour of GLARE as a composite patch material. Guidelines for such a repair using GLARE can be generated using FEM and arranged for field use in similar situations or structural problems.

7.1.1 Structural Problem

The modelling involves a number of assumptions or simplifications to obtain an efficient and accurate model that can be trusted to provide meaningful results. Before the solution can be interpreted, the problem must be clearly defined.

The problem being investigated is that of a patched composite wing skin laminate under in-plane, biaxial, tensile loading, Figure 7-1. The damaged laminate has lost some of its static strength and high stress concentrations occur near the damage (Tan 1987b). The stress concentration is relieved by providing an alternate load path through the patch for the load initially carried by the damaged area. In this manner, the load carrying capacity of the structure is restored.

Figure 7-1 Structural Problem

The load transfer occurs through bearing of fasteners. The situation is basically a single shear joint with in-plane tensile loading. This joint arrangement has an inherent load eccentricity which creates a bending couple in the repair. The bending is taken up by the fasteners. Any load transfer through friction at the patch-laminate faying surface is not considered.

By using accepted modelling techniques, the structural problem can be represented as collection of finite elements. These elements are then applied appropriate constraints and loads. The methodology used to accomplish this is given in the following section.

7.1.2 Methodology

FEM creates a mathematical model of a real-world structure. Obviously, the representation will not be exact for complex systems but, if certain guidelines are

followed, high levels of accuracy can be obtained. The numerical solution is arrived at by dividing a solid, or in this case a thin plate, into a number of smaller elements. These elements are assigned specific material and section properties.

The computational time required is related to the size of the model (i.e. number of elements). In order to reduce the time, a portion of the model can be constructed taking advantage of the rotational symmetry of the problem (Caffrey & Lee 1994). Each portion of the structure that is not modelled is replaced by equivalent boundary conditions. The model is then geometrically defined through the elemental mesh.

The elements join together at points referred to as nodes. Nodes have no mass or stiffness, but provide links between the elements. A collection of elements resembles a web, hence it is referred to as a mesh. The selection of element type and density of the mesh are key parameters in obtaining accurate results.

A complete finite element model must contain definitions of the following (Brauer 1993):

- Geometry, divided into finite elements
- Materials
- Excitations (applied forces or displacements)
- Constraints

These parameters, as they apply to the present model are discussed in the following sections.

7.1.2.1 Elements

The evaluation of the repair includes a combination of one and two-dimensional elements which are assigned specific properties. Each element is assigned its own coordinate system to define its position in the mesh. Figure 7-2 presents the

mesh constructed to represent a repair similar to the schematic of Figure 7-1. Section A is the test section of the composite specimen. Sections B and C are included to accurately represent the test specimen described in Chapter 6.

Figure 7-2 Specimen Mesh

7.1.2.2 Two-dimensional Elements

Two-dimensional (2D) elements are used to model thin plates or shells. Quadrilateral and triangular elements are most commonly used. 2D shell elements in NASTRAN are assigned stiffnesses in five of the six possible degrees of freedom. Rotation about the normal to the plate is not constrained (Reymond & Miller 1996).

“Quad” elements are four-sided, 2-D elements. They can be assigned four nodes at the corners (QUAD4) or be of a higher degree with mid-point nodes along the edges (QUAD8). Similarly, “tri” elements are three-sided and can have

three nodes (TRI3) or higher. TRI3 elements may experience excessive stiffness and result in stress anomalies in the output if located next to QUAD4 elements (Caffrey & Lee 1994). Tri elements are only used when geometric constraints dictate, such as in the corner fillet, Figure 7-2. QUAD4 elements are used in the plate and the patch, as well as in the fastener heads.

Once assembled, the mesh must be assigned properties to determine its stiffness. The type of properties selected can also restrain the elements to certain behaviour. Since the composite plate can be considered a thin, flat plate and will be loaded in-plane, 2D membrane properties are assigned. This gives the elements stiffness in the X and Y directions only. This arrangement is suited to the plane stress condition ($\sigma_z = 0$).

For a more general approach, shell element properties can be assigned. In this case the elements are assigned stiffness in the X, Y, Z directions and also the rotations about X and Y. This selection is used for the elements of the patch where the loading state is not as evident. Hamilton (1991) employed QUAD4 shell elements in an investigation of repair of curved panels. By modelling a curved panel with flat plate elements, the out-of-plane rotation was found to be incompatible for neighbouring elements. The Nastran software automatically constrains the rotation about Z, but this artificially increases the stiffness of the mesh. Hamilton counteracted this problem by increasing the number of elements which kept the incompatibility to a minimum. The flat panels of the present investigation do not experience this problem.

7.1.2.3 One-dimensional Elements

The explicit modelling of the fasteners has rarely been done in a repair investigation. It involves substantial modelling time and most investigations are not concerned with the stress concentrations introduced by the holes, just the

load transfer. Fredell (1994) attached the damaged structure and the patch with beam elements. Her & Shie (1998) explicitly modelled the holes but not the fasteners by representing the fastener loading as a cosine distribution in the bearing direction. In the present investigation, both the fastener beams and holes are modelled which provides a more accurate representation of the stress in both the laminate and patch.

Figure 7-3 Fastener Model

The fastener discs do not completely fill the holes modelled in the patch and plate. This tolerance gap is bridged with radial BAR2 elements that will act as non-linear springs (GAP). The gap element is assigned properties that give it a very high stiffness in its closing direction and a negligible stiffness in its opening direction. This approach allows explicit modelling of hole deformation. GAP elements have been used in joint (Bellinger & Poon 1990) and repair investigations (Shi 2001). The addition of non-linear properties to the model adds complexity to the solution but it more closely represents the interaction between the fastener and the plate. Where appropriate, one-dimensional elements are employed. The fastener model consists of two circular discs that represent the cross-section of the fastener, Figure 7-3. The discs are modelled as 2-D membranes in the same plane as the patch and plate. The central node on each of these discs acts as a connection point for a bar element oriented in the Z

direction. The bar is modelled as a beam with a cross-section equal to the diameter of the chosen fastener. The BAR2 elements transfer the load from the plate to the patch. The discs then bear against the hole edges. The element selection was based on previous work, as discussed above. These techniques are appropriate to the repair investigation.

7.1.2.4 Mesh

The connection of all the elements results in a mesh of nodes. In the creation of the elements, the location of the nodes can be biased so that a higher density exists in regions of interest, allowing greater accuracy. Using an unrepaired model (i.e. no patch, with only central damage hole), results were compared with the exact solution of the Tan-Tsai (1994) method for complex stresses around an elliptical hole in a laminate. The mesh density surrounding the central hole was increased until the FEM produced agreement within approximately 3%. Further refining resulted in negligible increases in accuracy and significantly longer computation time.

The highest densities of elements occur near the central cut-out and the fastener holes. The loading arms are meshed with a lower number as they are not of interest. Here, a compromise is made between high density and computation time. With the present model, a solution was obtained using a SGI Origin system in approximately 40 minutes per case.

In the fillet location, TRI3 elements were employed to allow the curvature needed. Use of QUAD elements in this location led to problems maintaining acceptable aspect ratios and tapers in the elements. Once the limits are exceeded the results computed are not considered accurate. The TRI3 elements are stiffer but provide the necessary geometry and meet the tolerance limits.

7.1.2.5 Materials

Since the materials involved in the investigation are composites, the material property definition is very important. The model attempts to accurately represent the test specimen discussed in Section 6.3.2. Using the finite element pre-processor, MSC Patran, the individual lamina material properties used in the construction of the specimen are assigned to the shell elements of the model. By duplicating the lay-up of the specimen sections in the software, the global properties are produced, Table 7-1. These properties are very similar to the values obtained using CLPT and presented in Table 6-2.

Table 7-1 FEA Material Properties (Msi, in) (Harrison 1998)

	AS4/3501-6	FM300K	Section A	Section B	Section C
E₁₁	21.32	0.71	8.89	6.72	0.878
E₂₂	1.31	0.71	7.38	0.925	5.00
G₁₂	0.73	0.05786	3.11	0.254	0.198
v₁₂	0.3	0.3	0.34	0.3	0.0526
thick.	0.00521	0.00521	0.25	0.25	0.25

The fasteners were assigned the properties of the titanium alloy, Ti-6Al-4V. The fasteners selected for use in the experimental verification were made of this alloy. The material of the fastener contributes little to the overall behaviour of the model and stainless steel fasteners provide the same performance. Titanium is used primarily for corrosion protection as it has a lower galvanic difference with carbon than aluminium and is lighter than steel.

A similar method to the above was used to model the fibre metal laminate GLARE. As discussed in Section 4.2, GLARE consists of alternating layers of 2024-T3 aluminium sheet and S2-glass prepreg. The GLARE3 variant, which is

the main focus of the investigation breaks the glass prepreg into an additional two layers to create a biaxial layup. Each prepreg layer actually contains two orthogonal (0-90) plies.

The properties of the prepreg and the aluminium were used, along with their thicknesses, to obtain GLARE3 laminate properties, Table 7-2, in a similar manner to the cruciform specimen. Being a biaxial layup, the moduli are equal in the 11 and 22 directions. Therefore, in the analysis, GLARE3 is treated as an isotropic material. This simplification has been used in previous research (Fredell 1994).

Table 7-2 GLARE Material Properties (MSi, in.)

	GLARE3-2/1	GLARE3-3/2	GLARE3-5/4	GLARE3-7/6
E₁₁	8.82	8.46	8.21	8.12
E₂₂	8.82	8.46	8.21	8.12
G₁₂	3.1	2.89	2.75	2.7
v₁₂	0.271	0.263	0.257	0.255
Thick.	0.033	0.055	0.098	0.1416

7.1.3 Loads and Boundary Conditions

To produce results, loads must be applied to the mesh and the elements be constrained in a manner that duplicates a specimen installation or real-world application.

7.1.3.1 Load Introduction

When a specimen was installed in the load frame, Section 8.1, load was applied by bearing through four bolts in each arm. It is assumed that all the bolts see an equal fraction of the load. This is translated to the model by omitting the grip portion of the specimen, and applying a distributed load to the outer edge of the

orthogonal arms of the specimen, Figure 7-4. The function divided the total load over the edge length and applied nodal forces to the model. Introducing the load through the grip bolt holes would require a curved distributed load on the holes to simulate the load transfer by bearing. This would add unnecessary complexity to the model.

A number of initial trials were run to determine the load required to obtain a maximum principal strain of $4000 \mu\text{in/in.}$ near the damage hole. This is the design ultimate strain for composite wing skins (Manno 1981). For the model, this corresponded to 30 kip (30,000 lbf) on each of the specimen arms. The current arrangement fixes the centre of the damage hole, as the arms are pulled in both the X and Y directions. The load indicated on Figure 7-5 results in 15 kip on each arm distributed over the 4 in. edge length. (half of the total). The use of distributed loads ensures a uniform deformation along the specimen edges.

Figure 7-4 Loading Conditions

7.1.4 Boundary Conditions

The test setup allows vertical motion through the horizontal grips and horizontal motion through the vertical grips, Figure 7-4. This ensures uniformly distributed load application and does not introduce side loads into any of the actuators. In the model, this can be represented through boundary conditions.

To simplify the model and reduce computational time, one quarter of the specimen was constructed. The removal of material necessitates the addition of boundary conditions to complete the simulations. The existing basic boundary conditions are shown in Figure 7-5 (a designation of 1 represents a translation constraint in the x direction, etc.). The fibre metal laminate patch is assigned similar boundary conditions.

Figure 7-5 Boundary Conditions

7.1.4.1 Multi-point Constraints (MPC)

With the chosen fastener loading arrangement, the fastener heads are allowed to rotate. This causes the solution to diverge or give erroneous results because of displacement incompatibilities. To limit this problem, the central point of each fastener head is allowed only to translate in the X and Y directions but no rotation is permitted. This constraint is then applied to a series of nodes on each fastener head. Therefore, they must translate the same amount that the central point

experiences. This prevents fastener rotation, but allows a certain degree of fastener deformation to occur. This linking is referred to as a multi-point constraint (MPC) in the Nastran code. The central node (independent) transfers its motion to the dependent nodes, Figure 7-6.

Figure 7-6 Multi-point Constraint

7.2 TEST PROGRAM

The finite element model was used to evaluate GLARE as a composite field repair material. An unrepaired test case, U, was run first to determine the baseline for repair effectiveness and to refine the model. The results of this test case are discussed along with the experimental data in Section 9.1.

A number of test cases were devised to investigate the effect of varying design parameters on the effectiveness of the patch. For each test case, the maximum principal strains in the laminate and patch were recorded. Table 7-3 outlines the test program.

Additional test cases were run to investigate the possible failure behaviour of the repair. These investigations are discussed in Chapter 10.

Table 7-3 FEA Test Matrix

Test Case	Number of Trials	Patch Thickness	Patch Material	Fastener Diameters	Shape
A	20	0.033"	Glare3-2/1	1/8"	Square – 15 Figure 7-7
		0.055"	Glare3-3/2	5/32"	
		0.098"	Glare3-5/4	3/16"	
		0.142"	Glare3-7/6	1/4" 5/16"	
B	12	0.033"	Glare3-2/1	5/32" 3/16" 1/4"	Square – 8 Figure 7-8
		0.055"	Glare3-3/2		
		0.098"	Glare3-5/4		
		0.142"	Glare3-7/6		
C	12	0.033"	Glare3-2/1	5/32" 3/16" 1/4"	Circular – 10 Figure 7-9
		0.055"	Glare3-3/2		
		0.098"	Glare3-5/4		
		0.142"	Glare3-7/6

Figure 7-7 Test Case A

Figure 7-8 Test Case B

Figure 7-9 Test Case C

7.3 FINITE ELEMENT ANALYSIS AND RESULTS

The finite element model described in Section 7.1 was used to obtain strain results for different test cases. The strain results were then compared to observe the effect of changes in patch thickness or fastener diameter on the maximum principal strains in the repair design.

The goal of the repair was to reduce the maximum principal strain at the damage hole while limiting the magnitude of the strain concentrations introduced by the fastener holes. The effectiveness of the repair was based on the relative magnitude of these values. The damage size was maintained at 1 in. diameter for all test cases. Maximum principal strains are reported for the damage hole, the maximum strain near the critical fastener hole in the composite and the maximum strain near the critical fastener hole in the patch.

7.3.1 Test Case A – 8 in. x 8 in. Patch, 1 in. Fastener Pitch - Results

The values obtained in the analysis are based on the mesh presented in Figure 7-10. The patch mesh is located directly above the centre section of the specimen. Loading, boundary conditions and fastener attachments are discussed in Section 7.1. The 12 tested variants of Test Case A are presented in Table 7-4.

Figure 7-10 Test Case A FEA Mesh

Table 7-4 Test Case A FEA Results

ID	Patch Material	Patch Thickness	Fastener Diameter	Max. Principal Strain		
				microstrain		
		in.	in.	Damage	Fastener Hole on Composite	Fastener Hole on Patch
1	GLARE3-2/1	0.033	0.125	3210	3170	4760
2	GLARE3-3/2	0.055	0.125	3060	3330	3980
3	GLARE3-5/4	0.098	0.125	2880	3500	2900
4	GLARE3-7/6	0.142	0.125	2820	3550	2120
5	GLARE3-2/1	0.033	0.1875	3270	3140	3620
6	GLARE3-3/2	0.055	0.1875	3050	3200	3220
7	GLARE3-5/4	0.098	0.1875	2780	3360	2620
8	GLARE3-7/6	0.142	0.1875	2590	3460	2160
9	GLARE3-2/1	0.033	0.250	3350	3630	3760
10	GLARE3-3/2	0.055	0.250	3160	3720	3390
11	GLARE3-5/4	0.098	0.250	2840	3890	2910
12	GLARE3-7/6	0.142	0.250	2610	4010	2510

Loading the mesh of Figure 7-10 creates a complex strain state within the test section. The fringe plot of the strain in the test section, Figure 7-11, shows that strain concentrations develop around the fastener holes and the damage, as expected. The outer row of fasteners sees the greatest load and, therefore, the greatest strain. These fasteners are the first to encounter the load on its path through the repair and pick up the majority of the bearing into the patch. The highest loaded fastener is located in the lower right-hand corner of Figure 7-11.

**Figure 7-11 Test Case A6 Max. Principal Strains – Composite Plate
Test Section at 30 kip**

The orthotropic nature of the composite is apparent in the non-uniformity of the strain field. With an isotropic material, the outer rows on both the top and bottom of the test section would be highly loaded with the corner fastener being the highest loaded. The vertical direction has a greater stiffness and therefore experiences less strain.

The quasi-isotropic GLARE3-3/2 patch sees a much more uniform strain field. The fringe plot of Figure 7-12 shows both the horizontal and vertical outer rows equally loaded. The slight difference in the far field strain is due to the unequal load transfer from the fasteners in the composite.

Figure 7-12 Test Case A6 – Max. Principal Strains – GLARE3-3/2 Patch at 30 kip

Generally, as the thickness of the patch increased, the strain at the damage hole and at the most critical fastener hole in the patch decreased. The strain at the most critical fastener hole in the composite increased with patch thickness. This behaviour is outlined in Figure 7-13, Figure 7-14, and Figure 7-15 with three different fastener diameters.

The patch fastener strain in Figure 7-13 is very high relative to the other values for the thinner patches. It sharply declines as the thickness increases. This

fastener diameter, 1/8 in. is not normally used in repair designs for shear strength reasons; therefore, subsequent test cases replace it with 5/32 in.

An efficient patch design is the variant 6 in Table 7-4, due to the combination of low composite strain and reasonable damage and patch strain. This design was implemented in the experimental test program.

Figure 7-13 Test Case A – 1/8 in. Fasteners

Figure 7-14 Test Case A - 3/16 in. Fasteners

Figure 7-15 Test Case A - 1/4 in. Fasteners

The magnitude of the strains on the patch vary slightly as the fastener diameter changes, as shown in Figure 7-16. There is a relatively larger difference in the composite, with the larger fasteners imposing a higher strain.

Figure 7-16 Test Case A – Strain Variation with Patch Thickness for Different Fastener Diameters

7.3.2 Test Case B – 6 in. x 6 in. Patch, 1 in. Fastener Pitch

The second test case removed the outer row of fasteners from Test Case A which reduced the total to 32, Figure 7-17. This forced each fastener to carry a higher load, increasing the peak strain near the holes. The 1/8 in. diameter fasteners were replaced in the analysis with 5/32 in. because of availability. Table 7-5 presents the strains obtained for the 12 different variants.

Figure 7-17 Test Case B FEA Mesh

Table 7-5 Test Case B – Max. Principal Strain under 30 kip Loading

ID	Patch Material	Patch Thickness	Fastener Diameter	Max. Principal Strain		
		in.	In.	microstrain		
				Damage	Fastener Hole on Composite	Fastener Hole on Patch
1	GLARE3-2/1	0.033	0.15625	3500	3810	4410
2	GLARE3-3/2	0.055	0.15625	3360	3990	3790
3	GLARE3-5/4	0.098	0.15625	3180	4200	2890
4	GLARE3-7/6	0.142	0.15625	3100	4290	2220
5	GLARE3-2/1	0.033	0.1875	3490	3920	4270
6	GLARE3-3/2	0.055	0.1875	3330	4090	3770
7	GLARE3-5/4	0.098	0.1875	3140	4400	3060
8	GLARE3-7/6	0.142	0.1875	2980	4460	2530
9	GLARE3-2/1	0.033	0.250	3550	3840	3660
10	GLARE3-3/2	0.055	0.250	3370	3950	3310
11	GLARE3-5/4	0.098	0.250	3110	4150	2810
12	GLARE3-7/6	0.142	0.250	2930	4290	2400

The strain behaviour of the composite test section followed that of Test Case A. The right side of the section shown in Figure 7-18 experienced higher strain than the upper edge of the section. This is again due to the anisotropy of the composite. As for the patch, Figure 7-19, the strain follows a fairly uniform distribution with the left side of the patch loaded slightly higher.

Figure 7-18 Max. Principal Strains – Composite Plate at 30 kip

Figure 7-19 Max. Principal Strains – GLARE3-3/2 Patch at 30 kip

The strain variation with patch thickness, is similar to that of Test Case A. Figure 7-20, Figure 7-21, and Figure 7-22 show the behaviour using 5/32, 3/16 and 1/4 in. fasteners respectively. The trend of increasing composite strain, with decreasing patch and damage strain continues. Figure 7-20 also shows that the small diameter fastener, 5/32 follows the same behaviour as that of the 1/8 fastener in Test Case A; a large drop in strain with thickness increase.

In terms of effectiveness, the magnitude of the damage strain is reduced when compared to similar variants in Test Case A. The reduction is small in magnitude but it comes with the elimination of 28 fasteners in the repair design. This is a significant weight saving, an important consideration in aircraft repair. Also, the labour and machining time are reduced.

The variant that provided the a good combination of strains was B10, and was chosen for assembly and experimental verification; however, it was later modified and utilised in a failure investigation, Section 10.1.

Figure 7-20 Test Case B - 5/32 in. Fasteners

Figure 7-21 Test Case B – 3/16 in. Fasteners

Figure 7-22 Test Case B – 1/4 in. Fasteners

As with Test Case A, the fastener diameter seems to have limited effect on the magnitude of the strains, Figure 7-23.

Figure 7-23 Test Case B – Strain Variation with Patch Thickness for Different Fastener Diameters

7.3.3 Test Case C – 8 in. diameter Patch

Test Case C was intended to investigate the effects of the fastener layout in a radial pattern as opposed to a rectangular one. The fastener pitch was reduced from 1 in., in Test Cases A and B to 0.8-0.9 in Test Case C. Edge distance remained the same at 1 in., Figure 7-24. The results for the 12 variants are given in Table 7-6.

Figure 7-24 Test Case C – FEA Mesh

Table 7-6 Test Case C - FEA Strain Results under 30 kip Loading

ID	Patch Material	Patch Thickness	Fastener Diameter	Max. Principal Strain		
		in.	in.	microstrain		
				Damage	Fastener Hole on Composite	Fastener Hole on Patch
1	GLARE3-2/1	0.033	0.15625	3470	3750	4430
2	GLARE3-3/2	0.055	0.15625	3280	3950	4140
3	GLARE3-5/4	0.098	0.15625	3090	4150	3080
4	GLARE3-7/6	0.142	0.15625	2880	4330	2720
5	GLARE3-2/1	0.033	0.1875	3390	4080	4770
6	GLARE3-3/2	0.055	0.1875	3210	4270	4310
7	GLARE3-5/4	0.098	0.1875	2940	4560	3610
8	GLARE3-7/6	0.142	0.1875	2930	4490	2320
9	GLARE3-2/1	0.033	0.250	3410	3870	3960
10	GLARE3-3/2	0.055	0.250	3247	3970	3530
11	GLARE3-5/4	0.098	0.250	3010	4120	2930
12	GLARE3-7/6	0.142	0.250	2900	4160	2260

The effect of the anisotropy in the composite was seen again in the circular repair arrangement. Figure 7-25 shows the right side of the test section experiencing the highest strains, the fastener hole on the lower right being the highest loaded. A significant effect of the change in material properties was seen near the upper edge. Here the non-uniaxial plies of the test section are truncated and replaced by plies of adhesive. This is reflected in the model through different material properties in the test section and the arms (not shown).

**Figure 7-25 Test Case C7 – Max. Principal Strains – Composite Plate
Test Section at 30 kip**

The patch fringe pattern, Figure 7-26, is fairly uniform with the fasteners nearly equally loaded. The upper left portion of the patch has a higher far field strain. This was due to the high stiffness of the composite in that area.

Figure 7-26 Test Case C7 – Max. Principal Strains – GLARE3-3/2 Patch at 30 kip

The strain variation with patch thickness, Figure 7-27, Figure 7-28, and Figure 7-29, show a similar trend to that of the rectangular patches, Test Cases A and B. The strain at the most critical fastener hole in the patch did exhibit different behaviour. A non-linear, cubic type curve as opposed to the exponential decline of Test Case A and B. As shown in Figure 7-28, the effect is most pronounced using 5/32 in. fasteners. As the fastener diameter increases, the trend follows that of the rectangular patches.

The strain magnitudes are comparable as well, Table 7-6, but the number of fasteners has been reduced to 40, down from a total of 60 in Test Case A. The radial fastener pattern offers similar performance to the rectangular patch while reducing the overall weight of the repair.

Figure 7-27 Test Case C – 5/32 in. Fasteners

Figure 7-28 Test Case C - 3/16 in. Fasteners

Figure 7-29 Test Case C – 1/4 in. Fasteners

Except in the patch, Figure 7-30c, the fastener diameter does not significantly affect the strain in the repair. Generally, the 1/4 in. fastener variants produced lower strains in all the areas. The larger fastener diameter allows the load to be transferred by bearing over a larger area, thus reducing the peak strain at the fastener holes.

Figure 7-30 Test Case C – Strain Variation with Patch Thickness for Different Fastener Diameters

8 TEST SETUP

The purpose of the experimental program was to verify the finite element model and to investigate the effect of patch assembly on load transfer in the repair design.

The experimental test program was carried out at the National Research Council (NRC) Institute for Aerospace Research – Structures, Materials and Propulsion Laboratory (IAR-SMPL). An MTS Planar Biaxial Test Frame and supporting controller software and hardware were used.

8.1 TEST FRAME

The MTS Planar Biaxial Test Frame, Figure 8-1, consists of four opposed 50 kip capacity hydraulic actuators which allow planar loading in both tension and compression. The loading grips were previously modified from a hydraulic to a floating mechanical arrangement to accommodate the cruciform specimen, Figure 8-2. The floating grips also provide the necessary boundary conditions and prevent the introduction of side loads to the actuators.

Figure 8-1 MTS Planar Biaxial Test Frame with Cruciform Specimen

The actuators can be commanded in load or displacement control. Feedback is provided through a load cell and a linear voltage displacement transducer (LVDT) on each arm.

Figure 8-2 Horizontal Floating Grip

8.2 TEST CONTROLLER

The load control was provided by an MTS TestStar II control system. The so-called “hybrid” control mode ensured the load was not affected by the specimen drifting down under its own weight. The system prevented this by applying force, but checking the feedback of the displacement of the actuators. The amount of the displacement of opposing actuators was kept equal.

A positive displacement was defined as a movement which would produce a tensile load on the specimen. Therefore, all tensile displacements were radial, Figure 8-1. The test program involved only equal biaxial ratio tests, so differential loading was not required.

The test profiles were produced using MTS TestWare SX. This software allows actuator command in load or displacement modes. Test profiles were created for each test case to load the specimen, and to gather data at specified load points. The test profiles are presented in Sections 9.1 through 10.1.2.

8.3 DATA ACQUISITION

In addition to the force and displacement feedback signals provided by the test control system, 16 differential input signals were available through an Instronet 100 data acquisition system, Figure 8-3. Strain gauge signals, wired in a quarter-bridge configuration provided values to compare with finite element predictions.

Figure 8-3 Instrunet 100 Data Acquisition Module on Test Frame

Due to the complex loading environment, 45° single-plane strain gauges rosettes were used to determine the principal strains during the test. Planar rosettes were necessary to ensure the power density on the composite material was not too high as this could result in local overheating. Stacked rosettes were excluded on this basis. During the failure investigations, Sections 10.1 and 10.2, uniaxial strain gauges were used when the loading direction was already known.

Data was only acquired continuously during the failure investigations, B and F. Otherwise, data from each gauge was taken at every hold during the loading profiles.

8.4 PHOTOELASTIC OBSERVATIONS

By bonding a thin sheet of photoelastic material to the specimen, cruciform or patch, and then applying load, surface strains can be seen through a fringe pattern (Whitney et al 1982). The fringes, when viewed through a polariscope, can be analysed to determine the principal strain difference or maximum shear

strain in the specimen. The equipment used consists of a light source, polarizer and camera to capture the images through the polariscope (Figure 8-4).

Figure 8-4 Photoelastic Equipment and Setup

Photoelastic coating was applied to several test specimens to obtain a qualitative view of the strain field. The fringes were not reduced to obtain strain at a specific point. The use of strain gauges on the opposite side of the specimen made this redundant. Also the accuracy of the system was not guaranteed. The properties of the photoelastic coating are presented in Table 8-1

Table 8-1 Photoelastic Coating Properties

Measurements Group Inc. Photoelastic Sheet	
Type	PS-1B
'K' factor	0.15
Thickness	0.082 ± 0.002 in.
Lot no.	0628
'f' value	925
Adhesive	PC-1 Bipax Adhesive/Hardener

8.5 FASTENERS

The fasteners used in the test program were Huck Fasteners Inc. Unimatic and Ti-Matic blind fasteners. They were developed specifically for use in composite materials. Unimatic blind fasteners are pull-type, single action, flush break and feature a large bulb formation on the back side (Armstrong & Barret 1998). The fasteners are constructed of A286 stainless steel. Ti-Matic fasteners are the all titanium version of the Unimatic, Figure 8-5.

Figure 8-5 Ti-Matic Blind Fastener (Anon 1999)

The fasteners are available in four diameters: 1/8 in., 5/32 in., 3/16 in., and 1/4 in. Repair operations generally use the larger diameter varieties to ensure sufficient load carrying capacity. The fastener holes are drilled slightly oversized, and as the fastener is engaged, the sleeve expands to fill the hole, while the large footprint bulb forms on the back side, Figure 8-6.

Figure 8-6 Ti-Matic Installation Sequence (Anon 1999)

The pull grooves of the fastener stem fit into the grips of a unique nose piece that attaches to Huck installation tools. The hand-held pneumatic tool, Figure 8-7, grips the stem and pulls back, acting as an anvil. As the proper tension is reached the stem is broken off by the tool and is discarded leaving a flush head.

Figure 8-7 Blind Fastener Installation Tool Model 225

Although available in countersunk versions, all fasteners used in this investigation had protruding heads. The thickness of the patches constructed did not permit countersinking as the head would extend to approximately $3/4$ of the depth of the patch, resulting in inadequate load transfer. Protruding head fasteners would present aerodynamic problems on a lifting surface; however the scenario investigated involved short-term applications and the localised aerodynamic disturbance was deemed acceptable.

9 EXPERIMENTAL VERIFICATION PROGRAM

Selected test case variations from the finite element analysis were constructed using the existing biaxial cruciform composite specimens. The specimens were instrumented using strain gauges and photoelastic coating.

Five specimens were used in the experimental test program. The first three (1,2,3) were used to compare the strain field of the analysis to that of an actual installation. The remaining two tests (4,5) were used to investigate the possible failure behaviour of the repair designs. These tests and the test results are described in Section 10. Test Case F was also used to verify the composite strains.

1. Test Case U – Unrepaired Specimen
2. Test Case A6 – 8 in. x 8 in. Rectangular GLARE3-3/2 Patch
3. Test Case C2 – 8 in. dia. Circular GLARE3-3/2 Patch

4. Test Case B – 6 in. x 6 in. Rectangular GLARE3-3/2 Patch (Modified)
5. Test Case F – 8 in. x 8 in. Rectangular GLARE3-3/2 Patch (Single Fastener)

9.1 TEST CASE U – UNREPAIRED SPECIMEN

In order to provide a baseline and to evaluate the data acquisition scheme, a damaged but unrepaired specimen was tested, Figure 9-1. The damage was simulated by a 1 in. diameter through-thickness hole. This also allowed the loading scheme to be validated and adjusted. The test section of the cruciform was coated with a photoelastic material to visualise the strain field in the damaged region. The test case parameters are presented in Table 9-1.

Figure 9-1 Test Case U – Geometry

The parameters listed are limited since the specimen does not involve a repair patch. This test case represented the structure in the damaged condition after a clean-up operation. Normally, composite penetration damage is cleaned-up by carefully drilling a larger circular or elliptical hole in place of the damage. This removes any localised delamination and exposed fibres.

Table 9-1 Test Case U - Parameters

Parameter	Value
Specimen No.	942
Damage Size	1"
Fastener Quantity	0
Fastener Diameter	N/A
Fastener Pitch	N/A
Patch Thickness	N/A
Fastener Number	N/A

The loading profile for Test Case U, Figure 9-2, consists of biaxial loading in the range of 0 – 30 kip tensile force. The load was applied equally to each arm of the specimen. The loading was halted at 7 points in order to take strain data and observe the photoelastic fringe pattern.

Figure 9-2 Test Case U – Loading Profile

The strain data was supplied by three 45° strain gauge rosettes (Micromeritics CEA-06-125UR-350) applied on the surface of the specimen, Figure 9-3. The strain from each of the three gauges that make up the rosette was then reduced to obtain the principal strains at the strain gauge locations.

The opposite face of the test section of the specimen was coated by a 8 in. x 8 in. photoelastic coating, Figure 9-1. The data from the photoelastic investigation showed the strain state in the test section after the damage and before the repair. The penetration damage was seen as a stress concentration.

Figure 9-3 Strain Gauge Locations – Test Case U

The wiring and strain gauge numbering is given in Figure 9-4. The gauges were wired in a quarter-bridge circuit using $350\ \Omega$ bridge completion resistors. The wiring led to the Instrunet module shown in Figure 8-3 and the strain signals were digitised by a PC and recorded.

Figure 9-4 Strain Gauge Wiring – Test Case U

9.1.1 Experimental and Analytical Comparisons

Figure 9-5 presents the maximum principal strain produced by the analysis and the principal strain reduced from the three component gauges of each rosette. Substantial difference is seen in gauge location R1 and R3. They show a difference of approximately 50% and 30%, respectively. These gauges are placed near the corner fillet of the test section which experiences a very high strain gradient. Strain is defined at a single point, but the strain gauge averages the value over the area of the gauge. When a measurement is being taken in an area where the strain gradient is non-linear, as in this case, the errors can be significant (Whitney et al 1993). Near the corner fillet, the strain changes by a factor of two, as determined from the FEA, which increases the error.

Figure 9-5 Test Case U – Strain Gauge Results

Due to symmetry the strain predicted for locations R1 and R3 was equal; however, the gauge readings were not. This suggests either an unequal load path through the test section or an irregularity in the material properties of the specimen. Examining the photoelastic fringes of Figure 9-6, the strain field in the specimen does not appear to be uniform. This specimen was also used to test

the loading profile and controls. The load actually seen by the specimen may be slightly different than that applied to the model. Gauge R2, near the damage hole showed good agreement with predicted values, approximately 3% at higher load levels.

The photoelastic coating provided a qualitative picture of the strain state in the test section under loading. The fringes of Figure 9-6 indicate the effect of the damage hole on strain concentration. As the load increases, the fringes, which represent the principal strain difference or maximum shear strain, become denser around the damage. The unique, non-circular pattern is in agreement with previous photoelastic investigations of laminated composites. The final image shows the coating delaminated after the specimen failure.

Figure 9-6 Test Case U – Photoelastic Fringes

At the load of 30 kip, the left arm of the composite cruciform failed by shearing away from the test section, Figure 9-7. Subsequent shear-out of the grip bolts in the other directions followed. The specimen had only been tested to 10 kip maximum in the previous investigation for which it was designed (Harrison 1998). As a result of this failure, the 30 kip value was established as an absolute upper limit for follow-on testing.

Figure 9-7 Test Case U – Post Failure

9.2 TEST CASE A6 – LARGE RECTANGULAR REPAIR PATCH

Test Case A6 consisted of a 8 in. x 8 in. repair patch of GLARE3-3/2. The fasteners were Unimatic 3/16 in. spaced at 1 in. pitch. The specimen was coated with photoelastic material on a quarter of the patch surface in order to investigate the strain field in the patch. The damage hole remained at 1 in. diameter, Figure 9-8.

The fastener holes were drilled and reamed to the specified diameter using a numerical controlled (NC) machine tool. Both the patch and specimen holes were

drilled simultaneously to ensure alignment. Plexiglass backing was provided to limit delamination on the drill bit exit side.

Figure 9-8 Test Case A6 – Geometry

The repair patch covers the entire test section of the specimen. A286 stainless steel Unimatic fasteners were used to assemble the repair, Table 9-2. The installation sequence was randomised. All bulb formations on the back side of the specimen were deemed acceptable as per Huck shop procedures (Anon 1999).

Table 9-2 Test Case A6 - Parameters

Parameter	Value
Specimen No.	967
Damage Size	1 in.
Fastener Quantity	60
Fastener Diameter	0.375 in.
Fastener Pitch	1 in.
Patch Thickness	0.055 in. GLARE3-3/2
Fastener Number	Unimatic NAS1919C06S05WU

The loading of Test Case A6 was similar to that of Test Case U; however, the maximum force applied was 20 kip, Figure 9-9.

Figure 9-9 Test Case A6 – Loading Profile

Five strain gauge rosettes were used to gather data. Three were placed on the composite specimen, Figure 9-10, and two on the patch, Figure 9-11. The patch gauges were placed back-to-back with the specimen gauges. The strain gauges were placed where the FEA predicted the highest strains.

Figure 9-10 Test Case A6 – Strain Gauge Locations on Specimen

Figure 9-11 Test Case A6 – Strain Gauge Locations on Patch

The strain gauge installations on the specimen and patch are shown in Figure 9-12 and Figure 9-13, respectively. The strain gauges were installed after the specimen was fastened together. This did not allow for the recording of any strain introduced during the assembly process. The post-installed condition was treated as the zero strain condition.

Figure 9-12 Test Case A6 - Strain Gauge Wiring on Specimen

Figure 9-13 Test Case A6 – Strain Gauge Wiring on Patch (Installed)

Photoelastic coating was applied to one quarter of the patch after assembly. This allowed the viewing of the strain field of the patch under loading. The small fringes visible around the fastener holes in Figure 9-14 are a result of the machining of the holes. The coating is extremely sensitive to heat from the drilling. The purpose of the photoelastic coating was to provide a qualitative view of the strain field, so the localised damage was acceptable.

Figure 9-14 Test Case A6 – Photoelastic Coating on Patch (Installed)

9.2.1 Experimental and Analytical Comparison

Figures 9-15, 9-16 and 9-17 represent the strain gauge results from the composite. The difference between the prediction and the gauge values is evident; however, the slopes of the curves are very similar suggesting that, if the strain in the specimen was zero prior to the test then the agreement would be better.

Figure 9-15 Test Case A6 – Strain Gauge R1

Figure 9-16 Test Case A6 – Strain Gauge R2

Figure 9-17 Test Case A6 – Strain Gauge R3

The experimental results in Figure 9-18 and Figure 9-19 show a significant non-linearity, suggesting a complex load transfer in the patch. Due explanation of this result could be the fastener fit and assembly sequence. If one fastener is gripped tighter, then it will attract a greater proportion of the load. This will cause inequalities within the strain field of the patch. As the load increases the load transfer may change, loading some fasteners and unloading others.

Figure 9-18 Test Case A6 – Strain Gauge R4

Figure 9-19 Test Case A6 – Strain Gauge R5

A qualitative view of the strain field is presented by the photoelastic fringes of Figure 9-20. It shows that the right edge outer row fasteners of the patch see the highest load, as the density of the fringes is greatest in that location. The interaction between the strain concentrations developed around the fastener holes create a very complex strain field.

Figure 9-20 Test Case A6 – Photoelastic Fringes

The strain field produced in the photoelastic coating was compared to the maximum strain fringe plot produced by the analysis, Figure 9-21 . Although this is only a qualitative comparison, the similarities are noticeable. The outer row fasteners are the highest loaded and there is significant interaction between the strain concentrations introduced by the fastener holes. The photoelastic coating had larger holes to accommodate the fastener heads.

Figure 9-21 FEA (left) and Photoelastic Fringes under 20 kip Loading

9.3 TEST CASE C10 – CIRCULAR REPAIR PATCH

Test Case C10 consisted of an 8 in. dia. repair patch of GLARE3-3/2. The fasteners were Ti-Matic 1/4 in. The damage hole remained at 1 in. diameter, Figure 9-22. Due to the directionality of GLARE3, care was taken to ensure that the 0-90 directions were oriented in the loading directions to provide maximum stiffness along the load path.

Figure 9-22 Test Case C10 – Geometry

The parameters of Test Case C10 are given in Table 9-3. The inner row fastener pitch is slightly less than that of the outer row. The Ti-Matic fasteners are of the same batch used to assemble Test Case B.

Table 9-3 Test Case C - Parameters

Parameter	Value
Specimen No.	956
Damage Size	1 in.
Fastener Quantity	40
Fastener Diameter	0.25 in.
Fastener Pitch	Inner: 0.80 Outer:0.92
Patch Thickness	0.055 in. GLARE3-3/2
Fastener Number	Ti-Matic UBP-MV08-05AC

The loading profile for Test Case C, Figure 9-23, is identical to that of Test Case A. Constant ramp commands bring the specimen to the required load and maintain the specimen at the hold points.

Figure 9-23 Test Case C – Loading Profile

Five 45° strain gauge rosettes, CEA-06-125UR-350, were applied to the specimen and patch prior to assembly. The locations are indicated in Figure 9-24 and Figure 9-25. Strains were monitored continuously during the fastener installation process and at the specified hold points in the loading profile. The gauges after installation are seen in Figure 9-26 and Figure 9-27.

Figure 9-24 Test Case C10 – Strain Gauge Locations on Specimen

Figure 9-25 Test Case C10 – Strain Gauge Locations on Patch

**Figure 9-26 Test Case C10 – Strain Gauge Wiring on Specimen
(Installed)**

Figure 9-27 Test Case C10 – Strain Gauge Wiring on Patch (Installed)

9.3.1 Experimental and Analytical Comparison

The maximum principal strain versus time plot of Figure 9-28 shows substantial variation in magnitude during the fastener installation. The installation sequence was carried out to ensure each area of the patch had approximately the same amount of fasteners at any time. The gauges on the composite saw very little change in their magnitude, while the patch gauges fluctuated ± 400 microstrain. This left a residual strain at the gauge areas of 400 and -200 microstrain for R4 and R5 respectively

Figure 9-28 Test Case C10 – Installation Strains

The strains recorded after the assembly operation were kept as the initial values for the loading profile. The strain versus load plots of gauges R1, R2 and R3, those on the composite, show the actual gauge value, the FEA prediction and a corrected value which subtracts the residual strain due to assembly from the gauge value. This curve is more appropriate for comparison as the FEA does not account for the installation strain. The corrected values of the composite gauges

differ from the FEA predictions, Figure 9-29, Figure 9-30, and Figure 9-31, by 5 – 18%.

Figure 9-29 Test Case C – Strain Gauge R1

Figure 9-30 Test Case C – Strain Gauge R2

Figure 9-31 Test Case C – Strain Gauge R3

The gauges on the patch, in particular R4, Figure 9-32, still show a significant difference from the FEA prediction. R4 differs by 80% while R5 differs by 20%. The curves are approximately linear which indicates the patch assembly has resulted in a uniform strain distribution through the patch. Test Case A showed a non-linear behaviour in the patch gauges. This also indicates that the tested strain levels are very low and puts the suitability of this cruciform specimen into question.

The slope of the load-strain curve is dependent on the material elastic modulus. The gauges are returning smaller strain values for a given load than the analysis. This implies the patch is actually stiffer than the FEA model.

Figure 9-32 Test Case C – Strain Gauge R4

Figure 9-33 Test Case C – Strain Gauge R5

10 FAILURE INVESTIGATIONS

Following the specimen failure of Test Case U, Section 9.1, the test program was modified to investigate the possible failure of the patch repair. Although the test was limited by the 50 kip capacity of the load frame actuators, the previous failure established an upper limit of 30 kip for the specimen. It was necessary to develop a patch design to promote a localised failure in the test section of the repair and not in the specimen arms. The finite element model developed for Test Case B, Section 7.3.3, was used to evaluate variants of the repair which would result in failure of the laminate or the patch. Two investigations were carried out and implemented into test specimens.

The first investigation varied the existing configuration of Test Case B in order to maximize the stress concentration near the fastener holes, which would promote failure. The second investigation involved the analysis of an intentionally deficient patch design which was intended to induce a bearing failure in the patch near the fasteners.

10.1 TEST CASE B MODIFICATION

To maintain a margin of safety and ensure failure in the test section, a maximum load of 25 kip was desired. Test Case B was chosen to be modified because the patch and fastener hole pattern had already been machined when the failure investigations started. The geometry of the specimen was set, Figure 7-8; however, the number of fasteners installed, damage size and patch thickness could still be varied. The original Test Case B consists of 32, 1/4 in. fasteners spaced at 1 in. By investigating modifications to this specimen, a condition where failure could be induced in the test section, was established.

10.1.1 Finite Element Investigation

Three possible modifications were considered:

1. Enlarging the central damage hole
2. Changing patch thickness
3. Modifying the number of fasteners

The simplest solution would be the first one, which involved one additional machining operation; however, as the diameter increases, there would be significant stress interaction with the nearby fastener holes. The patch thickness was limited by the availability of the GLARE variants. As a GLARE3-3/2 patch had already been machined, it was preferred; however, a thinner GLARE3-2/1 patch was also considered. Thirdly, removing fasteners would load the others to a higher level increasing the stress concentration around the fastener holes. If the inner row of rivets was removed, the outer row would see a larger load, that may be sufficient for failure. Additionally, combinations of the three options were considered, such as a larger diameter damage hole and reduced numbers of fasteners.

10.1.1.1 Failure Criteria

The composite laminate was deemed to fail if the Hoffman Failure Criterion, Equation 10-1, exceeded 1. This criterion is applied to orthotropic plates in a general state of plane stress with unequal tensile (X_t , Y_t), compressive (X_c , Y_c) and shear (S) strengths. The principal stresses (σ_1 , σ_2) and shear stresses (σ_{12}) are computed for each ply of the laminate. The finite element solver evaluated the expression for each ply, using first-ply-failure (FPF) as a condition for laminate failure.

$$\left[\left(\frac{1}{X_t} - \frac{1}{X_c} \right) \cdot \sigma_1 + \left(\frac{1}{Y_t} - \frac{1}{Y_c} \right) \cdot \sigma_2 + \frac{\sigma_1^2}{X_t \cdot X_c} + \frac{\sigma_2^2}{Y_t \cdot Y_c} + \frac{\sigma_{12}^2}{S^2} - \sigma_1 \cdot \frac{\sigma_2}{X_t \cdot X_c} \right]$$

Equation 10-1 Hoffman Failure Criterion

FPF means that if a single ply satisfies the criterion, the laminate is deemed to have failed. It is noted that the failure indices (values obtained from evaluation of the criterion) do not follow a linear behaviour; therefore, failure cannot be linearly extrapolated using previous runs. Separate tests must be run for each change in loading.

The GLARE patches were deemed to have failed if the stress in the patch exceeded the ultimate bearing stress of the material. Bearing failure was preferred to failure modes that result in a complete loss of load carrying capacity. Therefore, the edge distance (e/d ratio) and fastener pitch were kept high to avoid shear-out and net tension failure, respectively.

The bearing strength of GLARE3 has been characterised by Slagter (1992). Using the test procedure method described in ASTM D-953, which involves lateral constraints and employs the 4% hole elongation approach, GLARE3 was shown to exhibit behaviour similar to a plastic material in bearing. The repair configuration available does not permit the addition of lateral constraint; however, the clamping of the rivets does provide out-of-plane support. The bearing strength values used for determination of failure are given in Table 10-1.

Table 10-1 GLARE Bearing Strength (Slagter 1992)

GLARE Layup	Pin-type Bearing Strength (ksi)		Bolt-type Bearing Strength (ksi)		Compressive Yield Strength (ksi)
	Yield	Ultimate	Yield	Ultimate	
GLARE2-3/2	-	79.6	76.9	102.8	60.0
GLARE3-3/2	-	77.2	79.2	114.4	44.8
GLARE4-3/2	-	74.0	75.1	95.4	52.9

10.1.1.2 Iterative Failure Investigation

A number of variants of the geometry and repair parameters of Test Case B were modelled and analysed. The results of each run were checked against the failure criteria for the composite and the patch. Additional changes to geometry or loading were done to arrive at a variant which produced a bearing failure.

The following tables present the strain results for different patch configurations. Table 10-2 presents the results using a 0.033 in. thick GLARE3-2/1 patch. The inner and outer rows refer to the presence of installed fasteners. For example, variant 1 includes all eight fasteners (1/4 of the total an actual specimen) of the original design. Variant 9 includes only the five fasteners of the outer row.

Only variants 6 and 9 produce failure predictions within the available loading envelope. The failures occur at the edge of fastener holes in the patch and laminate. Both these tests require enlarging the damage size. A 4 in. diameter damage hole (variant 9) would interfere with the fastener pattern already machined.

Table 10-2 Test Case B – Failure Investigation – GLARE3-2/1 Patch

Variant	Load (kip)	Central Hole Dia. (in.)	Outer Row Present	Inner Row Present	Strain at Central Hole	Strain at Rivet (laminate)	Strain at Rivet (patch)	Failure?
1	30	1	Yes	Yes	3550	3840	3660	No
2	30	2	Yes	Yes	3890	5500	4480	No
3	300	2	Yes	Yes	38400	54700	44800	Laminate & Patch
4	100	2	Yes	Yes	13900	19800	18700	Laminate & Patch
5	100	2.5	Yes	Yes	18002	29300	26100	Laminate & Patch
6	30	2.5	Yes	Yes	4260	7220	5420	Laminate
7	20	2.5	Yes	Yes	2860	5090	3580	No
8	25	2.5	Yes	Yes	3150	6330	4480	No
9	25	4	Yes	No	3400	6300	7140	Patch

Another series of variants was run to determine if reducing the number of fasteners in the repair, leaving the existing holes open, would be sufficient to induce failure. Table 10-3 shows that a patch failure may occur at 25 kip if one fastener (on the quarter specimen) is used, variant 4. This would not involve enlarging the central hole and the fastener would be located in the outer row, at the corner of the patch.

Table 10-3 Test Case B – Failure Investigation – GLARE3-2/1 Patch – Limited Fasteners

Variant	Load (kip)	Central Hole Dia. (in.)	Number of Fasteners (1/4)	Fastener Locations	Strain at Central Hole	Strain at Rivet (laminates)	Strain at Rivet (patch)	Failure?
1	25	4	1	Outer Corner	3860	6650	8150	Laminate & Patch
2	20	4	1	Outer Corner	3690	6660	8000	Laminate & Patch
3	10	4	1	Outer Corner	1490	2690	3120	No
4	25	1	1	Outer Corner	3130	3390	4270	Patch
5	25	2	1	Outer Corner	3260	4960	5010	Patch
6	10	2	3	Outer Corner	1280	1900	1860	No
7	20	2	3	Outer Corner	2520	3810	3750	No

Additionally, a number of variants were considered using GLARE3-3/2 as the patch material. These runs, Table 10-4, all included a single fastener in the outer row, corner position. By loading the model to 27 kip, a patch failure was predicted near the critical fastener hole. As the patch pattern was already machined in the GLARE3-3/2 material, variant 4 was selected as the repair design to be implemented and tested in the load frame.

Table 10-4 Test Case B – Failure Investigation – GLARE3-3/2 Patch – Limited Fasteners

Variant	Load (kip)	Central Hole Dia. (in.)	Number of Fasteners (1/4)	Fastener Locations	Strain at Central Hole	Strain at Rivet (laminate)	Strain at Rivet (patch)	Failure?
1	10	1	1	Outer Corner	1260	1440	1590	No
2	20	1	1	Outer Corner	2520	2870	3180	No
3	25	1	1	Outer Corner	3140	2570	3950	No
4	27	1	1	Outer Corner	3480	3960	4390	Patch
5	30	1	1	Outer Corner	3820	4350	4820	Patch
6	40	1	1	Outer Corner	5050	5740	6360	Patch
7	10	2	1	Outer Corner	1310	1980	1860	No
8	20	2	1	Outer Corner	2890	3960	3720	No
9	25	2	1	Outer Corner	3590	4920	4630	Patch

10.1.2 Test Method

Variant 4 from Table 10-4 was assembled using the pre-machined patch and laminate. The final geometry left four fasteners in the pattern, Figure 10-1. As bearing was considered the critical case, a method for detection of bearing failure onset was required. Four methods were considered: ultimate sustainable load, first peak load, percent hole elongation, and bearing strain.

An ultimate sustainable load condition is applied while observing the load/displacement curve of the material. The failure point is taken as the maximum point of the curve. Here the material exceeds its bearing limits and the curve flattens or drops off. This load is then divided by the bearing area to obtain the bearing ultimate strength. This is the method suggested for isotropic metal testing in ASTM E-238, pin-type bearing.

First peak failure involves a similar observation of the load/displacement curve. Before the maximum load is reached, the curve will see a number of peaks that indicate local yielding in the material. The first peak to occur is taken as the failure point.

Bearing strength has also been defined as the bearing stress at which the hole diameter is deformed to a percentage of its original size. ASTM D-953, bolt-type bearing, takes this to be 4%. It is determined by applying a template to the completed load/displacement curve and locating the offset. Subsequent tests on various laminates have shown different behaviour with bearing onset occurring as early as 0.4%.

The bearing strain approach detects a first ply failure. The failure of a ply results in a loss of stiffness in the local area. The load applied is the same, therefore a local strain increase should be observed. The difficulty lies in the detection of this step in the load/strain curve. The failure is very localised, hence small strain

gauges must be used and the load application must be steady and constant. The gauges should be placed along the bearing direction. Harrison (1998) used this method to varying success, as the step in the curve was difficult to identify. The method is suited to the specimen as no special coupon is needed and limited setup is required compared with the other methods.

The bearing strain method has not been used with GLARE before. Slagter (1992) used ASTM E-238 and D-593 to obtain the values of Table 10-1. The specimen does have a degree of out of plane constraint, so the bolt type bearing values will be used for the failure limit. For GLARE, one of the aluminum or glass fibre plies must fail in order for the laminate failure to be detected. Strain data from two uniaxial gauges placed at 45° to the loading direction near the fasteners was recorded continuously, Figure 10-2.

* Only corner rivets installed

Figure 10-1 Test Case B – Geometry

Figure 10-2 Test Case B – Strain Gauge Locations on Patch

Photoelastic coating was also applied to one quarter of the patch prior to assembly in order to observe the strain field during the installation and loading. The test case parameters are summarised in Table 10-5. The change in fasteners from Unimatic to Ti-Matic was due to availability.

Table 10-5 Test Case B - Parameters

Parameter	Value
Specimen No.	955
Damage Size	1 in.
Fastener Quantity	4
Fastener Diameter	0.25
Fastener Pitch	5
Patch Thickness	0.055 in. GLARE3-3/2
Fastener Number	Ti-Matic UBP-MV08-05AC

Figure 10-3 Test Case B – Load Profile

The load profile of Figure 10-3 shows a constant ramp test up to 27 kip where failure was predicted by the model. An additional portion was added to permit loading up to 30 kip, if necessary. The loading procedure was identical to the model verification tests of Section 9. The installed specimen with photoelastic coating is shown in Figure 10-4.

Figure 10-4 Test Case B – Strain Gauge Wiring (Installed)

10.1.3 Experimental Results

As the gauges and coating were applied to the patch prior to assembly, monitoring of the strain during assembly was possible. The photoelastic coating did not provide any useful information, as it debonded from the patch during the installation of the fasteners. The installation resulted in a very high transient strain reading gauge U2. This strain rate may have been responsible for the debond.

The strain-time traces are plotted in Figure 10-5. The spikes represent very high strain rates. After the installation, a slight buckle was observed between the patch and the composite. This may have lead to out-of-plane bending in the repair. During subsequent assemblies, Test Case C and F, precautions were taken to avoid this buckling by securely pressing on the patch during assembly.

Figure 10-5 Test Case B – Installation Strains

Figure 10-6 shows the load and strain in the patch during the test. The discontinuity visible at 27 kip was a hold in the load profile and not a failure

indication. No step, as prescribed by the bearing strain failure detection method, was observed, so the load was increased to approximately 30 kip.

At 29.3 kip loading, the right arm of the specimen sheared away from the test section of the cruciform specimen, Figure 10-7. The high stress concentrations in the fillets of the corners and the brittleness of the composite led to the rapid failure of the arm. A purpose-built failure investigation specimen was needed.

Figure 10-6 Test Case B – Strain and Load vs. Time

Figure 10-7 Test Case B – Post-failure

10.2 TEST CASE F – LARGE DAMAGE, SQUARE REPAIR PATCH

Test Case F arose from the desire to design another purposely deficient patch which would experience failure within the loading envelope available. Test Case B was constrained by modifying an existing specimen and patch. Test Case F was designed specifically to induce a bearing failure in a GLARE3-3/2 patch. Using the trends of the investigation of Test Case B, a simple repair design of a 8 in. x 8 in. rectangular patch with four fasteners was chosen, Figure 10-8. The damage size was increased to 5 in. diameter to further reduce the strength of the composite. This would require the patch and fasteners to carry more load.

Figure 10-8 Test Case F – Geometry

10.2.1 Finite Element Investigation

The mesh of Figure 10-9 which shows both the laminate plate and patch, was loaded to 20 kip. The resulting compressive stress was compared with the GLARE3-3/2 strength values of Table 10-1. The peak stress at the hole edge was observed to be 119 ksi. This value exceeds the ultimate bearing strength of the GLARE3-3/2 material. Slagter (1992) obtained these values by loading the material in its orthogonal directions (0-90). The main bearing direction in Test Case F was at 45° to the L and LT directions of the material. As the glass fibre plies of the GLARE are oriented orthogonally, they would contribute little to the bearing strength in the direction being investigated. The bearing load here was being carried by the aluminium.

Figure 10-9 Test Case F FEA Mesh

The fringe plot of the strain near the fastener hole at 20 kip, Figure 10-10, highlights the localised bearing effect of the fastener hole. The edge distance of 1 in. results in a e/d ratio of 4 which is necessary to avoid edge effects. Lower edge distance could cause a shear tear-out failure, undesirable for a repair. The bearing failure results in a reduction of load carrying capacity, which is preferable.

Figure 10-10 Test Case F – Minimum Principal Stress (ksi) Near Fastener Hole at 20 kip

10.2.2 Test Method

The parameters of the repair are given in Table 10-6. The geometry differs from Test Case B, with a much larger damage hole and a larger patch. The test program is very similar, with an identical loading profile, but with additional strain measurements.

Table 10-6 Test Case F - Parameters

Parameter	Value
Specimen No.	945
Damage Size	5 in. diameter
Fastener Quantity	4
Fastener Diameter	0.25
Fastener Pitch	7 in.
Patch Thickness	0.055 in. GLARE3-3/2
Fastener Number	Ti-Matic UBP-MV08-05AC

The loading profile, Figure 10-11 for Test Case F was identical to Test Case B, the previous bearing test investigation. This specimen was specifically chosen for a failure investigation and is therefore more suitable to test. Failure was predicted to occur near 20 kip, which would place it well within the loading envelope of the specimen and load frame.

Figure 10-11 Test Case F – Loading Profile

Four strain gauges were placed in the bearing direction of the four fasteners on the patch, Figure 10-13. It was anticipated that the bearing failure may be detected by the bearing strain method, Section 10.1.2. On the specimen, three 45° strain gauge rosettes were placed to record the composite material behaviour during the test, and for further model verification, Figure 10-12.

Figure 10-12 Test Case F – Strain Gauge Locations on Specimen

Figure 10-13 Test Case F – Strain Gauge Locations on Patch

10.2.3 Experimental Results

The strain in the specimen during installation was monitored, Figure 10-14. As before, the strain in the composite and the centre of the patch after assembly was minimal. The strain near the fasteners in the bearing direction (U1,U2,U3,U4) varied significantly during the installation.

Figure 10-14 Test Case F – Installation Strains

A similar procedure to that of Test Case B was used to determine if bearing failure had occurred in the patch. The bearing strains were monitored during the load profile, Figure 10-15, to detect a distinctive step change in slope. The plots did not show evidence of this. Detailed examination of the fastener holes would be necessary to determine if any deformation had occurred. During the fastener removal

Figure 10-15 Test Case F - Load and Strain vs. Time

For further model verification, two rosettes were placed on the composite (R2,R3) and one on the centre of the patch (R1). The patch corrected gauge values show the same trend as in the previous tests, Figure 10-16. The curve is linear but at a lower slope.

Figure 10-16 Test Case F – Strain Gauge R1

The gauges on the composite show good agreement, Figure 10-17 and Figure 10-18. The strain levels are higher due to larger damage; however, the gauges are in different positions so they cannot be compared directly.

Figure 10-17 Test Case F – Strain Gauge R2

Figure 10-18 Test Case F – Strain Gauge R3

11 DISCUSSION OF RESULTS

Both the analytical and experimental test programs were geared to evaluating the suitability and behaviour of GLARE as a patch material for carbon-epoxy laminates. Various configurations were tested to determine the most effective design of repairs. The following discusses the choice of GLARE in composite repair, the behaviour of a repair in terms of fastener size and patch thickness, and the suitability of the finite element model as a repair design tool.

11.1 GLARE AS A COMPOSITE REPAIR PATCH MATERIAL

The strain reduction near the damage comes about as the load that was carried by the missing composite material is now passed through the patch. By redirecting this load, the stress near the damage is reduced. Traditional patch materials, such as titanium, are normally selected to match the stiffness of the composite. This permits the patch to carry its share of the load and limits the attraction of additional load from outside the damage area. By selecting a material less stiff than the damaged structure, more load is passed through the patch and it does not locally stiffen the damaged area. Essentially, soft patching results in a higher loaded patch and a less loaded damage area.

A comparison study was carried out to compare the performance the titanium alloy, Ti-6Al-4V, with GLARE3-3/2 in the field repair scenario being investigated. Both patches were of equal thickness, 0.055 in., and 1/4 in. fasteners were used. Figure 11-1 shows that titanium performed better in reducing the strain near the damage, and also gave lower strain in the patch. At the critical fastener location in the composite, GLARE3 outperformed the titanium. The less stiff material does not attract as much load into the repaired area meaning the fastener region of the composite is less loaded. The higher strain in the patch when using GLARE3 is acceptable as the material is sufficiently strong for temporary use in the field.

More permanent repairs require damage tolerant design. In this case, even though the patch would experience a higher load, its fatigue life would be greater due to the very low crack growth rates in GLARE, Section 4.2.1. In general, mechanically fastened repairs can be fatigue critical structures. The stress concentrations around the fasteners can lead to the initiation of cracking. By reducing this concentration or using material, such as GLARE, that is not as susceptible to cracking mechanically fastened repairs can be designed to longer lives.

Figure 11-1 Titanium Alloy – GLARE3-3/2 Comparison

Another source of stress concentration in repair are the fasteners themselves. Typically, fasteners are countersunk to provide a flush surface. None of the specimens tested included countersunk fasteners as the thickness of the GLARE did not provide sufficient depth to avoid a knife edge condition in the patch. The knife edge would increase the stress concentration and drastically reduce the life of the repair. The protruding head rivet cause an aerodynamic penalty; however,

they provide the necessary load transfer and since the repair is designed to be temporary, they are sufficient for field use.

Working the material in the field can be a problem. Titanium requires much more machining precautions and specialised equipment than does GLARE. Traditional workshop procedures that are used for aluminium can be used for GLARE. This simplifies the work and skill required to assemble a fibre metal laminate patch. The material has been qualified for a number of uses and the data available for analysis and design is nearing that of the traditional metals. As the material is implemented in more applications, familiarity may overcome reluctance to use the newer material in repairs.

Another advantage of GLARE is its possible orthotropy. GLARE3 was used in the investigation because of the complex loading environment and uncertainty of the stress field. The quasi-isotropy of the layup provided adequate stiffness in all directions. If the principal directions of the loading are known, GLARE2 or GLARE4 variants can be chosen to provide stiffness in a specific direction. This permits a certain degree of tailoring the material to suit the situation.

The “soft-patching” concept proposed by Fredell and Gunnink (1992) suggests the use of a moderate stiffness material in patch repairs to avoid attracting additional load into the damaged area. The relatively low stiffness GLARE3 provides the necessary effectiveness in the investigation. Compared to titanium alloy, the damage strain reduction coupled with the fastener strain introduction, the GLARE performs better. The true benefit of GLARE in repair designs would come in longer term repairs where its excellent fatigue behaviour would extend the life of repairs beyond existing levels.

11.2 EFFECTIVENESS OF GLARE PATCH

Through the test cases studied using the finite element model, the effectiveness of the repair changed. The variation of patch thickness, fastener diameter, and patch shape are just three of many parameters that can be adjusted in the design but they indicate that by changing aspects of the design, performance can be altered.

In general, as the patch thickness increased, the strain at the damage hole decreased as shown in Test Cases A and C. This was coupled with an increase in strain at the critical fastener hole in the composite. The acceptable balance of these two values makes an effective patch. These two values tend to asymptotic levels as the patch thickness reaches 0.142 in. Extrapolating, subsequent increases in thickness will not have large effects in decreasing the damage strain. On the other end, small gauge patches result in a large damage strain but smaller composite strains.

These results provide a starting point to outlining a design procedure for repair of composites using GLARE. The study showed significant differences in strain levels for different patch thickness and shapes. These values when combined with an accurate failure model, could lead to the creation of a set of rules for repairing with GLARE. More work is needed to investigate further variations such as fastener pitch or patch tapering. The experimental testing and finite element results provide some direction, but they cannot be carried out for every repair scenario; generalised guidelines are needed.

11.3 FINITE ELEMENT MODEL ACCURACY

Finite element analysis was used as a tool to investigate the effect of parametric changes to the repair design. This involved small changes to the geometry of the model and repeated analyses to provide strain predictions of the completed repair. In order for the finite element model to be used as a design tool, it must

provide reasonably accurate predictions of the behaviour of the repair. Predictions were compared to experimental results and a number of observations were made:

1. Strain predictions for composite were reasonably accurate (15-20%).
2. Strain predictions for patch were relatively inaccurate (30-80%).
3. Patch was stiffer than the model predictions.
4. Repair assembly introduces strain which is not accounted for in the model.

The composite cruciform was accurately represented by the finite element model. The repaired specimens of Test Case A and C, along with the composite gauges on Test Case F show good agreement (5-18%). Once the strains are corrected to include installation effects, the prediction agrees well with the experimental data. This agreement supports the modelling procedures and material definition for the composite material.

The patch strain predictions differed from the gauge readings by as much as 80%. This is traced either to the material definition or the load transfer mechanism in the model. The load transfer from the composite to the patch is simulated by the fastener arrangement of beams and gap elements. In the majority of the tests, the strain measured on the patch was greater than the prediction.

GLARE3 was modelled as an isotropic material. This is not entirely accurate, as the stiffnesses in the orthogonal directions are equal, but the off-axis stiffness is lower. A sensitivity analysis was carried out which modelled the GLARE3 as a composite laminate, similar to the carbon-epoxy specimen. The resulting strains proved to be within 3% of the isotropic material results.

There are other methods to represent the material. It can be modelled as discrete layers instead of one layer of elements with a defined thickness. Special elements have also been developed to represent laminated materials. As an

example, Hashagen et al (1995) developed a geometrically and physically non-linear solid-like shell element for laminated materials. The 16 noded element was applied to GLARE and results showed good agreement with published data and experiments. The incorporation of such an element in a repair investigation may provide a more accurate picture of the behaviour of the patch.

12 CONCLUSIONS

From the finite element and experimental results, and observations during testing, the following conclusions were drawn.

On GLARE as a composite patch material:

1. GLARE3 has been shown to be an appropriate material for short-term, small scale, mechanically fastened composite plate repairs. The patches reduce the strain at penetration damage in a composite laminate.
2. The technique of "soft patching" has shown promise in composite repair.
3. Increasing thickness of the patch reduces the strain at the damage; however, this is accompanied by an increase in strain near the fastener holes.
4. Fastener size has a limited effect on effectiveness of the patch design.
5. GLARE is easily worked using standard machine shop procedures.

On mechanically fastened composite repair:

6. Installation of blind fasteners must be done with care to ensure that unwanted residual strains are not introduced into the repair during assembly.
7. Blind fasteners are a very simple, efficient tool for quick field repair operations.

On the finite element modelling:

8. The finite element model produced a reasonable prediction of strain in the composite portion of the repair.
9. The strain in the patch was over-predicted due to a lower than actual stiffness in the model.
10. Additional Work in Defining an accurate GLARE material model is necessary.

13 FUTURE WORK

A number of issues arose during the research, which require additional time and effort to resolve. Also, areas have been identified which have potential for further research projects:

1. With the coming introduction of GLARE as a primary structural material, there will be a definite requirement for repair guidelines and procedures. The procedures should be geared to maximising the damage tolerance benefits of the material.
2. Following from 1, the availability of the biaxial test setup at NRC-IAR-SMPL can provide a means to further characterise GLARE behaviour in complex loading environments. The development of a GLARE biaxial cruciform test specimen could be extremely valuable not only for repair investigations but possibly for joint or tension-compression tests.
3. To obtain more useful failure data, a new composite specimen should be designed to permit a repair failure within the limited loading envelope of the load frame. This would likely involve a thinner laminate.
4. The technique of soft patching has shown promise in this application. Further work may look at its use in the repair of traditional aluminium alloy aircraft structures.

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