Consequences and Challenges of GLARE for Structural Repair and Newly Designed Fuselage Structure

PROEFSCHRIFT

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ABSTRACT

Concerns over the safety and the maintenance cost have been raised over the last years in the airline. The important elements in maintenance of aircraft structural parts are the inspection threshold, repair classification, minimization of the structural damage (e.g. floor panel) and economic consideration of repair. The thesis focuses on the testing and analysis: The testing consists of several types of repair and new floor panel configurations. The analysis considers a new aircraft design in view of maintenance and Life Cycle Cost.

The repair configurations consist of current riveted repair, bonded-riveted repair and the usage of Glare as a repair material. The application of Glare sheet on fuselage repair is appropriate for damage tolerance of riveted repair. Adhesive as a faying surface in riveted repair improves the fatigue life significantly as well as the static strength. The combination of Glare sheet with bonded-riveted repair results in the highest fatigue life.

To determine the class of repairs, implementation of the Repair Assessment Program (RAP) is necessary in aircraft maintenance. However, it will result in considerable man-hours for evaluations, especially for assessing the existing repairs. To help the maintenance engineering to reduce the engineering-hours and to maintain a sufficient level of safety, a computer program is developed, named the Repair Calculation and Clasification Program (RCCP).

The investigation of new floor panel configurations were carried out by varying the top faces and core materials. The testing programs were impact, bending, compression and corrosion. The choice of Glare sandwich for a cargo floor proved to be the best solution. It results in a significant longer service life.

Extensive studies of the application of Glare sheet in newly designed aircraft concerning aircraft operations and maintenance, show that fuel savings and maintenance cost reductions could be obtained through weight saving and damage tolerance of the advance Glare structure.

An economical analysis is an essential requirements in order to confirm the technical advantages of Glare repair. The Life Cycle Cost analysis has confirmed that Glare as a repair material is more economical as compared to aluminum alloy.
Consequences and Challenges of
GLARE for Structural Repair and Newly Design Fuselage Structure

Key word: Repair, bonded-riveted repair, damage tolerance, inspection threshold, repeat inspection, fatigue basic, interval basic, Glare, Life Cycle Cost, scatter factor, rivet flexibility, fuel saving, inspection-hour.
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<td>AAWG</td>
<td>Airworthiness Assurance Working Group</td>
</tr>
<tr>
<td>AC</td>
<td>Advisory Circular</td>
</tr>
<tr>
<td>AD</td>
<td>Airworthiness Directive</td>
</tr>
<tr>
<td>AIA</td>
<td>Aerospace Industries Association</td>
</tr>
<tr>
<td>Al</td>
<td>Aluminum</td>
</tr>
<tr>
<td>ALI</td>
<td>Airworthiness Limitation Items</td>
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<tr>
<td>ASTM</td>
<td>American Standard Testing Material</td>
</tr>
<tr>
<td>ATA</td>
<td>Air Transport Association of America</td>
</tr>
<tr>
<td>CA</td>
<td>Constant Amplitude</td>
</tr>
<tr>
<td>CBS</td>
<td>Cost Break-down Structure</td>
</tr>
<tr>
<td>CPACP</td>
<td>Corrosion Prevention and Control Program</td>
</tr>
<tr>
<td>DAS</td>
<td>Designated Alteration Station</td>
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<tr>
<td>DER</td>
<td>Designated Engineering Representative</td>
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<tr>
<td>DS</td>
<td>Discard</td>
</tr>
<tr>
<td>ED</td>
<td>Environment Damage</td>
</tr>
<tr>
<td>EUAW</td>
<td>Equivalent Uniform Annual Worth</td>
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<tr>
<td>FAA</td>
<td>Federal Aviation Administration</td>
</tr>
<tr>
<td>FAR</td>
<td>Federal Administration Regulation</td>
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<tr>
<td>FD</td>
<td>Fatigue Damage</td>
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<tr>
<td>FMLs</td>
<td>Fiber Metal Laminates</td>
</tr>
<tr>
<td>FOD</td>
<td>Foreign Object Damage</td>
</tr>
<tr>
<td>HT</td>
<td>Hart Time</td>
</tr>
<tr>
<td>IATA</td>
<td>International Air Transport Association</td>
</tr>
<tr>
<td>ICAO</td>
<td>International Civil Aviation Organization</td>
</tr>
<tr>
<td>IRR</td>
<td>Internal Rate of Return</td>
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<tr>
<td>JAR</td>
<td>Joint Airworthiness Requirement</td>
</tr>
<tr>
<td>LCC</td>
<td>Life Cycle Cost</td>
</tr>
<tr>
<td>LT</td>
<td>Long Transverse</td>
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<tr>
<td>LU</td>
<td>Lubrication</td>
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<tr>
<td>MIL</td>
<td>Military Specification</td>
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<tr>
<td>MRB</td>
<td>Maintenance Review Board</td>
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<td>MSD</td>
<td>Multiple Site Damage</td>
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<td>MSG 3</td>
<td>Maintenance Steering Group 3</td>
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<td>MSI</td>
<td>Maintenance Significant Items</td>
</tr>
<tr>
<td>MTOW</td>
<td>Maximum Take Off Weight</td>
</tr>
<tr>
<td>MZFW</td>
<td>Maximum Zero Fuel Weight</td>
</tr>
<tr>
<td>NDT</td>
<td>Non Destructive Testing</td>
</tr>
<tr>
<td>NPV</td>
<td>Net Present Value</td>
</tr>
<tr>
<td>OMP</td>
<td>Operator Maintenance Program</td>
</tr>
<tr>
<td>OP</td>
<td>Operational Check</td>
</tr>
<tr>
<td>R</td>
<td>Stress Ratio (\frac{\sigma_{\text{gross max}}}{\sigma_{\text{gross min}}})</td>
</tr>
</tbody>
</table>
List of Symbol

<table>
<thead>
<tr>
<th>Symbol</th>
<th>Description</th>
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<tbody>
<tr>
<td>RAP</td>
<td>Repair Assessment Program</td>
</tr>
<tr>
<td>RCCP</td>
<td>Repair Calculation and Classification Program</td>
</tr>
<tr>
<td>RS</td>
<td>Restoration</td>
</tr>
<tr>
<td>RTG</td>
<td>Repair Assessment Task Group</td>
</tr>
<tr>
<td>SB</td>
<td>Service Bulletin</td>
</tr>
<tr>
<td>SFAR</td>
<td>Supplemental Federal Administration Regulation</td>
</tr>
<tr>
<td>SRM</td>
<td>Structure Repair Manual</td>
</tr>
<tr>
<td>SSI</td>
<td>Structural Significant Items</td>
</tr>
<tr>
<td>STC</td>
<td>Supplemental Type Certificate</td>
</tr>
<tr>
<td>SV</td>
<td>Servicing</td>
</tr>
<tr>
<td>$[C]$</td>
<td>Flexibility matrix</td>
</tr>
<tr>
<td>$[P]$</td>
<td>Matrix of the rivet's force</td>
</tr>
<tr>
<td>$[\Delta \text{eff}]$</td>
<td>Compatibility displacement matrix</td>
</tr>
<tr>
<td>$[C]$</td>
<td>Flexibility matrix</td>
</tr>
<tr>
<td>$[F]$</td>
<td>Rivet force matrix</td>
</tr>
</tbody>
</table>

Greek Symbols

$\mu$ A constant
$\theta$ Bearing distribution factor
$\beta$ Hole filling factor
$\alpha$ Hole surface condition factor
$\lambda$ Minimum incremental for each method of NDT
$\Delta L_{d1}$ Displacement of the doubler
$\Delta L_{s1}$ Displacement of the skin
$\Delta R_I$ Rivet displacement
$\sigma_{BT}$ Local bending moment
$\delta_{cf}$ Displacement due to concentrated force
$\sigma_{\text{doubler}}$ Ultimate tensile strength of repair doubler
$\delta_{\text{eff}}$ Displacement due to far-field stress,
$\delta_{pc}$ Displacement of patch
$\delta_{rad}$ Radial displacement,
$\sigma_{\text{ref}}$ Applied gross stress
$a$ Crack length
$A$ Cross section area of adhesive,
$a_0$ Minimum detectable crack length
$C_b$ Rivet flexibility
$C_D$ Coefficient of drag
$C_L$ Coefficient of lift
$D$ Rivet diameter
$d$ Rivet shank diameter
$E_a$ Adhesive modulus
$E_a$ Impact energy
List of Symbols

\[ E_r \quad \text{Young modulus of rivet} \]
\[ E_s \quad \text{Young modulus of Skin} \]
\[ f_s \quad \text{Form factor} \]
\[ G \quad \text{Adhesive shear modulus} \]
\[ g \quad \text{Gravity} \]
\[ h_{re} \quad \text{Height of the rebound of the impactor} \]
\[ I \quad \text{Moment inertia of adhesive} \]
\[ I_{basic} \quad \text{Interval basic} \]
\[ KN \quad \text{Kilo Newton} \]
\[ K_{ib} \quad \text{Bearing stress concentration factor} \]
\[ K_{ig} \quad \text{Stress concentration factor due to bypass load} \]
\[ L \quad \text{Distance between rivet rows} \]
\[ M_i \quad \text{Mass of the impactor} \]
\[ MPa \quad \text{Mega Pascal} \]
\[ N_{basic} \quad \text{Fatigue basic} \]
\[ N_{fr} \quad \text{Number of fastener rows} \]
\[ POD \quad \text{Probability of Detection} \]
\[ r \quad \text{Nominal shank radius} \]
\[ R \quad \text{Fuselage radius} \]
\[ R_{NDT} \quad \text{Repeat inspection interval NDT} \]
\[ Scf \quad \text{Scatter factor} \]
\[ SF \quad \text{Severity factor} \]
\[ T \quad \text{Inspection threshold} \]
\[ t \quad \text{Thickness of the adhesive layer} \]
\[ t_{av} \quad \text{Average thickness of skin and patch (doubler)} \]
\[ t_{doubler} \quad \text{Doubler thickness} \]
\[ t_{pc} \quad \text{Patch thickness} \]
\[ t_s \quad \text{Skin thickness} \]
\[ V_i \quad \text{Velocity of the impactor} \]
\[ W \quad \text{Pitch length} \]
INTRODUCTION

INTRODUCTION 1

Background 1

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Background

Knowledge and progress, on aircraft design can be said to be excellent, but in spite of what we know, the aircraft operator is still experiencing maintenance problems, which cost many of millions US dollars for repair and modification, not yet including indirect penalties such as delays, cancellation, unscheduled downtime and decreased availability of aircraft for service. As shown in many references the maintenance cost over twenty years may exceed the price of the aircraft itself by approximately 3 to 4 times. Furthermore the delay cost of the wide-body aircraft an average international airport may reach $ 200 per minute. This costs could be drastically reduced if the manufacturers had given due consideration to the maintenance issue in the beginning of the design. Improvement of maintenance techniques and procedures are also significant for reducing costs cost of repairs and modifications of the aircraft structure as a consequence of fatigue cracks, corrosion, impact damage and other causes.

Since 1990, the market for passengers and freight increased, but the air-fares were dramatically reduced. It caused a world wide exhausting competition due to the deregulation in USA and Europe. As a result there was a lot of pressure on the airlines and aircraft manufacturers to reduce maintenance costs.

In view above problems, the new aircraft sheet material Glare is of great interest. Glare is a so-called fiber metal laminate, built up from thin Al-alloy sheets bonded to a laminated sheet by adhesive bonding. The adhesive layers are the matrix material for uni-directional glass fiber. Glare was developed by the Structures and Materials Laboratory of the Faculty of Aerospace Engineering at Delft University of Technology. It was primarily developed to obtain a sheet with superior fatigue properties and good damage impact resistance. It has been amply shown that these goals were achieved. It then appears that Glare must also be an interesting material for maintenance and design:
INTRODUCTION

1. Repair of fatigue cracks and other aircraft skin damages. This can be a noteworthy problem for pressurized fuselage structures.

2. Replacement of aluminum sheet material in structural components. Where impact damage is the most critical issue. This is very much true for cargo floor sandwich panels.

3. A longer inspection interval of Glare stricture as compared to aluminum, will reduce essentially the man-hours in aircraft maintenance.

It is obvious that both repairs and replacements have to meet requirements with respect to safety and reliability, and more particular the damage tolerance requirements [FAR 25-45, 1978]. Attention must be paid to these questions. The theme of the research in the present thesis can be defined in general terms as the application of Glare to improve aircraft maintenance with respect to repair and replacement.

The thesis consist of four parts. The first two parts are:

I. The applicability of Glare as a repair material for the aircraft fuselage skin, with a bonded-riveted repair as an option for additional improvements.

II. Application of Glare as facing material in cargo floor sandwich panels or other application, e.g. cargo liner in A 330.

These two parts cover the problems defined before. As a result of the experience of the research on those problems, it was considered to be worthwhile to address design issues of new aircraft with respect to maintenance. This is done in the third part of the thesis:

III. Application of Glare in aircraft design with respect to aircraft maintenance.

A final part of the thesis is covering economic aspects of repairs. Good repair design are not necessarily the most cost effective solutions. Aspects of this question are considered in part IV.

IV. Application of life cycle cost for trading of Glare versus aluminum in maintenance.

Some more introductory comments on the subjects of each part of the thesis are given below.
Aircraft repair

The need for good repair methods is essential for the airlines in view of reducing the direct operating costs. Since the introduction of the damage tolerance requirements in 1978, repair and inspection significantly contribute to maintenance costs.

In service, lack of time and insufficient space in aircraft structures have frequently led to inadequate repairs and to fatigue problems afterwards. Small changes of the current repair procedures are necessary to improve the airworthiness of the aircraft without increasing the cost of the repair.

The objective of our experiments is focusing on the improvement of the current repair method, through the application of the cold bonded-riveted repair and the use of the new Glare sheet material. Extensive static and fatigue tests were carried out, which has led to determined the improvement in fatigue life after repair. This repair research is discussed in part 1.

Existing repair techniques and new repair options should be evaluated ,also with respect to damage tolerance requirements. For this purpose a Repair Assessment Program (RAP) was outlined, which leads to a large saving of engineering-hours. It provides a computer program, called Repair Calculation and Classification Program (RCCP). The objective of this computer program is to maintain a sufficient level of safety. The RCCP may reduce the engineering-hours up to 70 percent depending on the complexities of the repairs. The RCCP is also discussed in part 1.

Aircraft Cargo Floor

The high failures rate of the cargo floors in service is due to corrosion and impact damage, which forced the airlines to obtain a solution rather than replacements or repairs. This damage could have a direct affect on the operations of the aircrafts as well as the maintenance costs due to its low reliability. Extensive experimental research, described in part 2, has given relevant information to improve the impact properties and the corrosion behaviour of a new floor panel.

Aircraft Design with Respect to Aircraft Maintenance in Service

When the aircraft structure design engineer and the airline structure engineer are looking at the same time to the aircraft structure, they have a different perspectives. The aircraft designer will be proud of their excellent design in terms of competitiveness, weight saving and productivity. However, the airline engineer would be very much concerned to see the locations can prone to fatigue, corrosion or accidental damage, where these occur and how to inspect and repair. The most suitable moment for solving maintenance problems is in the design stage. In part 3 options of a new aircraft design were considered. In view of the good fatigue, corrosion and impact properties of Glare sheet, the study on fuselage skin application
of Glare sheet in the MD-11 fuselage could show that Glare has a significant saving in the inspection-hours.

**The Life Cycle Cost**

The introduction of a new material to the existing aircraft must be approved technically and economically. Investment costs for new applications are the major obstacle for aircraft industries. In order to confirm the above technical advantages, an economic evaluation is required. To perform this analysis, a Life Cycle Cost (LCC) has been calculated to compare each repair configuration in terms of competitiveness, as shown in part 4. It shows that Glare repair results into lower LCC as compared to aluminum repair.
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INTRODUCTION

Part 1: GLARE REPAIR ON ALUMINUM FUSELAGE SKIN

1. INTRODUCTION

1.1. Field Repair Experiences

1.2. Current Repair Procedure

1.3. Objective

1.4. References

1.1. Field Repair Experiences

In recent years, there have been a lot of structural problems caused by Accidental Damage, Environmental Deterioration (Corrosion) and Fatigue Damage to aircraft in service. To prevent these failures, a step by step method for fracture detection, non-destructive inspections, designs, modifications and repair procedures is developed by the manufacturers, governments, airlines and universities to guide and regulate the engineers. The purpose of this is to ensure the continuing airworthiness of the aircraft.

As mentioned in introduction before, maintenance cost may reach up to 3 to 4 times the price of an aircraft itself and the delay cost for the average international airport may reach $200 per minute. The major job in aircraft overhaul is the repair of the airframe, which could reach up to 70 percent of the total jobs. Therefore, the improvement in repairs could considerably be reduced in the maintenance cost.

For many reasons, a good repair recommendation with regard to fatigue, corrosion and accidental damage is often difficult to incorporate in field repair. Lack of spaces, equipment, time or complex repairs can limit the possibilities to produce a good repair. However, poorly performed inadequate repairs may considerably degrade the reliability of the aircraft structure which can lead to the loss of an aircraft. A Boeing 747 of Japan Air Line crashed due to an inadequate repair on the aft pressure bulkhead. Since that time, all aircraft manufactures, authorities and airlines have pointed out that those repair methods, materials and inspections needed to be improved to meet the safety and economical requirement.
The situation above has forced the airline engineers to obtain a solution in improving the current repair without too many changes in method, equipment, inspection, etc. Thus, by the beginning of 1995, Garuda Indonesia and Delft University of Technology has developed a research program that lead to design a new repair method which may improve the safety and economy. This research has focused on increasing the level of repairs to Category A or better with regards to safety and economy.

1.2. Current Repair Procedure

Before an airframe is committed for repair, it should be submitted for damage assessment by an engineer or technician. This assessment involves a series of inspections and a damage evaluation to determine whether the repair is justified. A careful examination of the extent of the damage allows for a correct assessment of the damage. The result of this assessment will classify the damage into a minor or major repair. This statement will affect the approval process of this repair. A minor repair is usually covered in SRM of each type of aircraft, model and manufacture serial number. This repair will be automatically approved as long as that repair is met and followed the SRM. For a major repair, the approval process and accomplishment can be seen in chapter 2.

In the current repair method, the use of one gage or thicker of a doubler than the skin being repaired, could easily lead to a static overdesign, the consequences could be a considerable loss in fatigue strength as mentioned by Swift [11]. This philosophy would not meet the damage tolerance requirement for the new repair concept.

The use of the current repair method with sealant as a faying surface on the former countersink holes at the run out row leads to unsafe repair, then it must be categorized as a temporary repair which needs to be replaced or a frequent inspection. This situation would significantly increase the maintenance cost while this type of repair occurs frequently in maintenance. The use of cold bonded-riveted in this case could increase the repair category to an acceptable level or better.

1.3. Objective

The objective and the scope of this research are to extend the life of current repairs throughout the application of cold bonding in riveted repair instead of sealant as faying surface and the use of Glare sheet as a repair material. This research also included the investigation of the effect of the use of the former countersink holes at the run out row of rivet in fatigue strength, from reference [4], it could reduce the fatigue strength about 33 to 66 percent. The standard repair method which exactly follows the instructions given in the Structure Repair Manual (SRM) has been carried out in order to compare the results of new repairs. The development of this new repair method uses the current repair procedure and production support such as surface treatment, inspection method, tooling, etc. The purpose of this was to avoid a massive investment when it would be applied. Hence, it will become more valuable for the industries to use this new repair method developed in the laboratory of Delft University of Technology. To determine a repair should be considered as a major
repair or a minor repair, it is important in term of the approval process of this repair. Chapter 2 discussed about the approval process of repair or alteration concerning the maintenance of the aircraft structures.

Cold bonding adhesive in riveted repair which will act as an anti fretting and to maintain airtightness. Unlike sealant, adhesive contributes significantly in the load transfer of a joint. The relative displacement between two skins might become smaller, therefore the load transfer through the rivets could also be lower. This effect has also resulted in lowering the bearing stress of the rivet holes, consequently it increases the fatigue strength. The analytical of the load transfer in bonded-riveted lap joint is discussed in chapter 4 of this part. The chapter 5 shows the static test results of the external doubler on patch and lap joint repairs with varies of the doubler materials and faying surfaces. In order to determine the fatigue basic and the interval basic of a repair, the fatigue test programs of patch and lap joint repairs have been carried out for the configurations which passed the static requirement. These fatigue test results discussed in chapter 6. In reference [2], the cold bonding in riveted repair of external patch repair increased the fatigue life with about 20 to 30 percent as compared to the sealant as a faying surface. Fokker F-28 has used the cold bonded-riveted on the fuselage lap joint for more than 20 years of successful results in service experiences.

Since the Aloha accident, the Airworthiness Assurance Working Group (AAWG) addressed the operators to perform a repair assessment program especially on an aging aircraft. It was stated that the existing repairs needed to be reviewed and might require additional inspection or replacement to terminate the inspection task. While most repairs in service were designed for static strength and fail safe criteria for the aircraft, certified prior to FAR 25-45 (1978) which added the damage tolerance methodology. The incorporation of the damage tolerance methodology in repair practises is required in order to ensure their continued airworthiness, but it is less clear in practice.

Repair Assessment Program (RAP) is a guideline for the operator to assess the existing repairs or the new repairs which incorporate the damage tolerance concept. RAP could also be used to determine the inspection threshold and repeat inspection interval of a repair. If the current inspection interval in the Operator Maintenance Program (OMP) or Structural Inspection Program satisfies the required interval found by RAP, it means that repair meets the damage tolerance criterion. Thus, no supplemental inspection is necessary to be done [4]. Therefore, the improvement of the repair should match with the existing maintenance program interval, otherwise it would be not beneficial. The application of the RAP in the maintenance will result a significant increasing the engineering-hours. In this part provides a computer program, called Repair Calculation and Classification Program (RCCP). This computer program is intended to help the airline engineers to assess the existing repairs or the new repairs which can maintain a sufficient level of safety and reduces the engineering-hours. The RCCP discussed in chapter 7.
1.4. References


[8] Aircraft Inspection and Repair, Acceptable Methods, Techniques, and Practices, Department of Transportation Federal Aviation Administration, United States of America, *AC 43.13 - 1A*.


2. FIBER METAL LAMINATE

2.1 Advantages of Glare®

In the early eighties, the development of fiber metal laminate in Delft University of technology was started. The purpose of this development was to obtain a new material with superior fatigue properties. Therefore the application of Glare was addressed for fatigue critical areas of the aircraft structures.

There are many types of Glare that have been developed in the recent years. The choice of Glare 3 for repairing existing aircrafts, is due to the strength in longitudinal (L) and long transverse (LT) directions which are approximately the same. In Glare 3, the fibers are laid up in cross-ply with a 50 percent in two directions. The fiber glass epoxy layers between the aluminum layers have a favorable effect on corrosion as well as the impact properties. So the application of Glare for fuselage skin repair could give a significant improvement.

The slightly lower elastic modulus of Glare as compared with the aluminum, results in lowering the load transfer in the skin and less load attraction into the repair area. This idea is called “Soft Patching” [4]. The conventional aluminum repairs create a stiff area in the structure which may attract an additional load into a repaired area; and as a result this repair could lead to a fatigue problem. Since the patch material must satisfy static and fatigue strength requirements, it can be difficult to use a conventional patch material such as monolithic aluminum alloy.

2.2 Glare®

Glare® is a family of Fiber Metal Laminates (FMLs), which are advanced hybrid aerospace materials composed of thin high strength aluminum alloy sheet, adhesively bonded by fiber/epoxy prepregs as shown in Figure 2-1.

Fiber Metal Laminates were primarily developed for fatigue prone areas such as fuselage skin. This material consists of alternating thin aluminum alloys layers (0.2 - 0.4 mm) and uniaxial or biaxial glass fibers prepeg. In Glare 3, the fibers are laid up
biaxially (cross-ply), therefore, the strength in longitudinal (L) and long transverse (LT) are approximately the same. The aluminum layer is coated with the phosphoric acid anodizing and protected with the corrosion inhibiting primer BR 127 to provide both good adhesion and excellent long-term corrosion behavior. Although, extremely aggressive environments would eventually cause surface corrosion on the outer surface of the FML, just as it would for monolithic aluminum, deep pitting and perforation, is precluded in practices because of the barrier function of the (inert) prepeg layers.

Figure 2-1, Fiber Metal Laminate (FML), typical 2/1 lay-up.

The superior fatigue behavior of FML in comparison with monolithic aluminum alloy, is the primary reason for its application. In the cracked of a FML area, the intact fiber will bridge the loads continuously, as a result, the crack growth rate would remain low, whereas the monolithic aluminum showed progressively increasing cracks growth rates. In Figure 2-2 shows the fatigue performance of Glare as compared with the monolithic aluminum alloy, reference [6].

In a standardized burn test, Glare shows that an 1100° C oil burner does not penetrate through the thickness even after 15 minutes of a direct flame. The temperature measurement on the non-exposed side showed a significant heat reduction while this side never reached 300° C. On the contrary, the monolithic aluminum sheet was burned away in 20 - 30 seconds. Glare passed the FAR 25.855 flame test requirement. Recently, Glare has also been accepted by Boeing and Airbus Industries for use in cargo floors.

The code following the Glare name means, e. g. *Glare 3 3/2 0.3*, the number 3 is the Glare type in Fiber Metal Laminate families and the fiber orientation is in cross-ply (biaxial) and the 3/2 means that Glare is made of three layers of thin Al 2024 T3 sheet and two layers of glass fiber. The 0.3 means that the thickness of the aluminum layers are 0.3 mm. While the types of Glare, that are commercially available, are shown in Table 2-1.

The use of Glare in aircraft structures will provide several advantages over the monolithic aluminum which are as follows:
- low density,
- high strength,
- almost fatigue insensitive,
- high impact resistance,
- burn-through resistance,
- good corrosion behavior,
- traditional workshop properties, and
- low crack growth rate.

Figure 2-2, Fatigue performance of Glare laminate [6].

Table 2-1, Glare types which commercially available.

<table>
<thead>
<tr>
<th>Type</th>
<th>Config</th>
<th>Aluminum Alloy</th>
<th>Prepreg Constituent</th>
<th>Prepreg Orientation</th>
<th>Surface Condition</th>
</tr>
</thead>
<tbody>
<tr>
<td>Glare 2</td>
<td>2/1, 3/2, etc.</td>
<td>2024 T3</td>
<td>Glass/ Epoxy</td>
<td>Unidirectional</td>
<td>Bare/ Clad</td>
</tr>
<tr>
<td>Glare 3</td>
<td>2/1, 3/2, etc.</td>
<td>2024 T3</td>
<td>Glass/ Epoxy</td>
<td>Cross-play</td>
<td>Bare/ Clad</td>
</tr>
<tr>
<td>Glare 4</td>
<td>2/1, 3/2, etc.</td>
<td>2024 T3</td>
<td>Glass/ Epoxy</td>
<td>Cross-play</td>
<td>Bare/ Clad</td>
</tr>
<tr>
<td>Glare 5-F1</td>
<td>2/1</td>
<td>5052 H34/7075 T6</td>
<td>Glass/ Epoxy</td>
<td>Cross-play</td>
<td>Bare</td>
</tr>
<tr>
<td>Glare 5-F2</td>
<td>2/1</td>
<td>2024 T3</td>
<td>Glass/ Epoxy</td>
<td>Cross-play</td>
<td>Clad</td>
</tr>
<tr>
<td>Glare 5-FW</td>
<td>2/1</td>
<td>5052 H34</td>
<td>Glass/ Epoxy</td>
<td>Cross-play</td>
<td>Bare</td>
</tr>
</tbody>
</table>

Note: - Glare 3 was developed primarily for fuselage skin application and Glare 5 for cargo floor application.
- The difference of the Glare 3 and the Glare 4 in above are the fiber fraction, where the Glare 3 is 50 by 50 and the Glare 4 is 60 by 40.
2.3 Available Data on Glare As a Repair Material

As shown by many experimental studies of aircraft manufactures and the Delft University of Technology, Glare has a future as an aircraft material. In 1997, Airbus Industries mentioned that Glare was one of the candidate material for fuselage skin of A 3XX, meaning that Glare is accepted for usage in aircraft structures. Flight testing of Glare repair was also being carried out by US Air-force on C 5 Galaxy; this repair was very successful.

To carry a repair on an existing aircraft, some data of static and fatigue testing in combination with aluminum is needed. The reason for this, where an existing aircraft fuselage skin is made of aluminum alloy, therefore the test data concerning this issue is important to support the calculation and approval process of a Glare repair. The experiment of the combinations of Glare and aluminum sheets in patch and lap joint repairs have been carried out (see chapter 6 of part 1). The test results show a significant improvement on the fatigue strength while there is no static strength reduction noted.

The Life Cycle Cost (LCC) calculation for Glare repair has also been carried out by Salim [9], it showed that a significant saving could be achieved. The saving is due to the man-hours needed for Glare repair are less than the aluminum repair. In the aluminum repair, the doubler should consist of two skins but for the Glare repair, the doubler consists only one sheet and does not necessary to be tapered. The calculation of this LCC is shown in chapter 3 of part 4.

From the data above, it shows that there is no doubt as to the use of Glare as a repair material in aircraft maintenance. The only thing we need is the approval process before it can be used.
2.4 References


3. REGULATIONS AND REPAIR APPROVAL IN AIRLINE MAINTENANCE

This chapter gives an overview of the regulations and repair approval procedure in airline maintenance within the civil aviation. The International Civil Aviation Organization (ICAO) establishes standards for approval of new aircrafts, aircraft maintenance, aircraft repair and many other important issues related to air safety that is required in "International" action. The objective of the ICAO is to achieve the highest practical, uniform air regulation, standards and procedure for aircraft, personnel, airways, and aviation services throughout the world. This guidance is important especially when we do an operation or maintenance in an international area. Therefore, the inspector responsible for the operation and maintenance in the international activities must be familiar with the content and the details of the ICAO standards and recommended practices.

This overview is mainly based on FAA procedures and regulations. The reason for this, is that most of the regulations and procedures for the civil aviation in a country who is a member of the ICAO are adopted from FAA. For civil aircrafts the basic sources of informations are contained in the Joint Airworthiness Requirement (JAR) which referenced are made from the Federal Aviation Regulation (FAR). Therefore, the use of the FAA procedure in this case, will generally represent a repair approval procedure which is applicable for most countries.

The general procedure discussed in this chapter is considered as a general guideline; only for detailed procedures and approval requirements, it must be referred to the local authority regulations.
3.1 Repair Categorization and Approval

A repair categorization is needed to determine a repair which should classify as a major repair or as a minor repair. This categorization of repair will directly be involved in the approval process of such repair, whereas finally will also affect the cost and time of the repair significantly. A repair is categorized as a major or minor, it will not change the type design approval of the original structure. Whereas a repair changes to the type design approval of the original structure, these changes shall be categorized as a major alteration.

This section provides a guidance in determining the categorization of a repair or alteration, ensuring that the aircraft can be returned to service in accordance with the approved technical data. This categorization follows SFAR 36 [8] that can be defined as follows:

1. **Major alteration**: An alteration that is not listed in the aircraft, aircraft Structure Repair Manual (SRM), or other accepted manual where that:
   
   - might appreciably affect to weight and balance, structural strength, performance, flight characteristics, or other qualities affecting airworthiness of the aircraft.
   
   - is not done according to accepted practices or can not be done by elementary operations.

2) **Major repair**: A repair that is:

   - If improperly done, might appreciably affect weight and balance, structural strength, performance, flight characteristics and other qualities affecting airworthiness of the aircraft.

   - is not done according to accepted practices or can not be done by elementary operations.

3) **Minor alteration or minor repair**: Any alteration or repair that is not classified as a major alteration or major repair.

Where the accepted practices are: Airworthiness Directive (AD), Service Bulletin (SB), Structure Repair Manual (SRM), etc.

Minor repairs or minor alterations which are covered by the accepted practices, and when it is done and inspected properly, then this minor repair or alteration will be automatically approved.

Under the Federal Aviation Regulation FAR 121.379(b), 127.140(b) and 145.51 it has been stated that major repairs and major alterations should be accomplished according to technical data approved by the administrator. While under SFAR 36
permits to certain FAR Part 121 (air operator/ airline) and Part 145 (repair station) operators to develop their own technical data for performing major repair on aircraft, airframes and/ or other appliances (covered in their qualification), when the approved data does not exist. The approved data which is used to approve major repairs or major alterations includes the following:

- Type Certificate Data Sheets,
- Supplemental Type Certificates (STC),
- Airworthiness Directive (AD, issued by the authority),
- Manufacturer FAA approved data,
- Designated Engineering Representative (DER) approved data, and
- Designated Alteration Station (DAS) approved data developed for alteration performed by that repair station only.

While the acceptable data that may be used to approve major repairs or major alterations on an individual basis are:

- FAA Advisory Circular (AC) 43.13-1A and 43.13-2A,
- manufacturer's technical information, e.g., Structure Design Manual, bulletin, etc.,
- MIL Specs, and
- FAA Field Approval.

FAA Field Approval is an approval done by an authorized inspector of a major repair or major alteration. This approval should be accomplished by:

- Examination of the data that is only for one aircraft,
- physical inspection, demonstration, testing, etc., which is for one aircraft only, and
- examination of the data only if that a duplication of an identical aircraft.

3.2 FAA Approval Form 337 Used for a Major Repair and a Major Alteration

FAA form 337 is needed when the changes are categorized as a major repair or major alteration. The FAA form 337 contains the following:

- Data approval issued for one aircraft is applicable only for the aircraft described in block 1 of the FAA form 337. This data can not automatically be used as approved data for an other aircraft. The data may be used only with the approval by the local authority as the basis for obtaining approval for an other aircraft.

- Data approval may use for approval of the other aircraft when the identical alteration is performed on an aircraft which has an identical make, model, and series to the original modifier.

- When the alteration or repair has been performed by any other person than the original modifier, then this data may be used as a reference for obtaining approval of other aircrafts as well as for another country when it is accepted by the local
authority.

- Approval for return to service (block 7 of FAA form 337) by a flight standards airworthiness inspector will be performed only when the operator designated personnel are not available.

### 3.3 The Paperwork Processes

This section discusses the process for obtaining the approval for major alteration or major repair. In order to comply with the regulation, it is important to understand each step of the approval processes carefully. The steps are as follows:

1) The inspectors must determine that the data supplied is complete enough to proceed for further evaluation of the proposed alteration or repair. Therefore, the inspector must review and evaluate:

   - The formal application form FAA 337,
   - Other forms that are used by the manufacturer or operator which can be accepted by the administrator,
   - The detail description and design of the proposed alteration or repair, which include sketches, drawings, stress analysis, photographs, etc.
   - The testing procedure and method to meet certification such as static strength, fatigue strength, demonstration for damage tolerant, etc.
   - The detail standard which ensures that the operator considers the effects of major alteration or repair on the structural integrity of the aircraft, inspection procedure and method, or any other factor that may affect the airworthiness.

   If in this step the data is not complete, then the operator must supply the additional information needed.

2) Evaluate the proposal to determine compatibility with the current aircraft configuration to make a preliminary evaluation of the proposed alteration or repair. The inspection to the aircraft is required at least for aircraft data recording, review maintenance and inspection procedures which may be affected by the alteration or repair. If the inspector determines that the assistance from engineering is needed for approving a major repair or major alteration, then the operator shall coordinate with their authority.

3) After the data for major alteration or major repair has been approved, the inspector will schedule a conformity inspection with the operator to verify workmanship which complies to the accepted or approved data. These conformity inspections include: The inspection during and after alteration or repair and the jobs which should be done according to the accepted or approved data.

4) A review of the approval for return to service, the aircraft must be approved by a person authorized by FAR 43.7. They must be completing the block 7 of FAA
form 337 and make a maintenance record entry.

Figure 3-1-A and Figure 3-1-B show the example of FAA form 337.

3.4 The Effects of Using Glare on a Repair of An Existing Aircraft On the Approval Process

Before we continue to evaluate the effect of using Glare on approval process, first of all we shall look into the history of the approval process of the aircraft. When an aircraft is first designed and built, then their features and performance are all recorded on drawings, reports and on the Type Certificate Data Sheet (TCDS). All these data records are an integral part of the approval process for the aircraft. The approval which is given by the Airworthiness Authority, is required by the government order before it is commercially used.

A repair or alteration will be classified as a minor or major according to the nature and extent of the airworthiness inspector investigations in connection with its proposed changes. In the case of Glare being used for repair on fuselage skin on existing aircraft, the inspector would determine that this change would not be in accordance with the accepted practices. While, this was changed the original design of the structure, however small, shall be approved by the Airworthiness Authority, otherwise it is not legal for further flight. The use of Glare for repair on fuselage structure could change the static strength, fatigue performance, inspection method, damage tolerant, etc., while this change might also affect to the structural integrity of the aircraft. Therefore it should be categorized as a major alteration. To approve this repair, it could be obtain in two ways as follows:

- The approval can be obtain as an individual approval for one aircraft only by the local airworthiness authority. The approval processes are needed that:
  
  - prepares the FAA form 337 as mentioned in section 2.2,
  - completes the data required for the approval process as mentioned in section 2.3 such as static strength calculation and test data, drawing sketch, fatigue calculation and test data, demonstration of damage tolerance requirement, etc.,
  - prepares a conformity inspection by the inspector to verify workmanship which complies to the approved data during and after alteration, and
  - completes the approval for return to service of block 7 of FAA form 337 by authorized personal.

- The approval can also be obtain throughout the aircraft manufacture by issuing a SB, as mentioned before that SB is an accepted practice which can use as a basis for approval. In SB, which includes information about the extension of this modification, aircraft type, model, aircraft effectivity, aircraft manufacture serial number and the effect of this changes to the aircraft weight and balance manual, aircraft performance, etc. When a SB has been issued, then it can be used to approved this modification. The SB could also be used to approved this
modification for all fleet in the world as long as the modification within the extended of this SB.

3.5 Conclusions and Remarks

1. The approval process of the application of Glare on the fuselage skin repair of the existing aircraft is classified as a major modification due to the changes of the original design on the primary structure.

2. The approval of Glare as a repair material for existing aircrafts could be obtained from the aircraft manufacture by issuing a Service Bulletin (SB).
3.6 References

[1] Aircraft Inspection and Repair, Acceptable Methods, Techniques, and Practices, Department of Transportation Federal Aviation Administration, United States of America, AC 43.13-1A.


[12] Federal Aviation Regulation, Maintenance, Preventive Maintenance, Rebuilding, And Alteration, FAR Part 43.


# REGULATION AND REPAIR APPROVAL IN AIRLINE MAINTENANCE

## MAJOR REPAIR AND ALTERATION
(Airframe, Powerplant, Propeller, or Appliance)

**INSTRUCTIONS:** Print or type all entries. See FAR 43.9, FAR 43 Appendix B, and AC 43.9-1 (or subsequent revision thereof) for instructions and disposition of this form. This report is required by law (49 U.S.C. 1421). Failure to report can result in a civil penalty not to exceed $10,000 for each such violation (Section 901 Federal Aviation Act of 1986).

### 1. Aircraft
   - **Maker:**
   - **Model:**
   - **Serial No.:**
   - **Nationality and Registration Mark:**

### 2. Owner
   - **Name (As shown on registration certificate):**
   - **Address (As shown on registration certificate):**

### 3. For FAA Use Only

### 4. Unit Identification
   - **Unit:**
     - **Airframe**
       - (As described in item 1 above)
     - **Powerplant**
     - **Propeller**
     - **Appliance**
       - **Type:**
       - **Manufacturer:**

### 5. Type
   - **Repair**
   - **Alteration**

### 6. Conformity Statement
   - **A. Agency’s Name and Address:**
   - **B. Kind of Agency:**
     - U.S. Certificated Mechanic
     - Foreign Certificated Mechanic
     - Certified Repair Station
     - Manufacturer
   - **C. Certificate-No.:**

   - **D. Identify that the repair and/or alteration made to the unit(s) identified in item 4 above and described on the reverse of attachments hereto have been made in accordance with the requirements of Part 43 of the U.S. Federal Aviation Regulations and that the information furnished herein is true and correct to the best of my knowledge:**

   - **Date:**
   - **Signature of Authorized Individual:**

### 7. Approval for Return To Service
   - Pursuant to the authority given persons specified below, the unit identified in item 4 was inspected in the manner prescribed by the Administrator of the Federal Aviation Administration and is:
     - **APPROVED**
     - **REJECTED**

   - **By:**
     - FAA Pit, Standards Inspector
     - FAA Designee
     - Repair Station
     - Person Approved by Transporation Canada Airworthiness Group
     - Certificate of Designation No.

   - **Date of Approval or Rejection:**
     - Signature of Authorized Individual

---

*Figure 3-1-A, Sample of FAA form 337*
NOTICE
Weight and balance or operating limitation changes shall be entered in the appropriate aircraft record. An alteration must be compatible with all previous alterations to assure continued conformity with the applicable airworthiness requirements.

5. Description of Work Accomplished
(If more space is required, attach additional sheets, identify with aircraft nationality and registration mark and date work completed.)

☐ Additional Sheets Are Attached

Figure 3-1-B, Sample of FAA form 337
4. RIVETED AND BONDED-RIVETED JOINTS

4. RIVETED AND BONDED-RIVETED JOINTS

4.1 Load Transfer Mechanism

4.2 Rivet Flexibility

4.3 Load Transfer in Lap Joint Repair

4.4 Load Transfer of the Bonded-Riveted in Lap Joint Repair

4.5 Adhesive Materials

4.6 Conclusions

4.7 References

Structural joints are the most common source of failure in the aircraft structure. Because a repair introduces new joints, it is important that all aspects in repair design shall be considered when making a structural analysis. Failures may occur for various reasons but generally it can be caused by secondary bending stress due to eccentricities, stress concentrations, excessive deflections, etc., or combinations of them. These are difficult to evaluate to same an exact degree. These factors do not only affect the static strength, but have a great influence on fatigue strength of the joint or the adjacent structure.

The use of riveting or adhesive bonding as a joining method in aircraft construction is an acceptable means for attaining a high structural efficiency. The combination of riveted-bonded joint is an improvement to achieve a higher fatigue strength. Extensive use of these two types of joints have been employed in the aircraft primary structure has not been implemented yet in civil aviation, but the bonded-riveted joint was used extensively in the Fokker 28 and Fokker 27 for lap joints of the fuselage skin.

References [3], [10] and [13] show that the bonded-riveted joints have has a better fatigue strength as compared with the merely riveted joint. The reason of this is, because, the load transfer trough the rivet reduces significantly as shown in load transfer calculations in this chapter. Load transfer in thin sheet joints is an major topic of this chapter.

4.1 Load Transfer Mechanism

The load transfer mechanism is to be considered when determining the fatigue strength of the structural part. In most cases of the riveting joint, the load is
transferred through the rivets, but the friction between the mating surfaces is usually neglected in the calculation. Hartman [1] mentioned that friction between mating surfaces has an important effect on fatigue strength. To include this in the calculation on load transfer gives more complexities. The faying surface has also a big influence in the load transfer mechanism. Schijve [2] reported that the use of LPS-3 can penetrate into the joint and prevents the friction between the mating surfaces. The LPS-3 acts as an anti-fretting, as well as a lubricant. It reduces the static strength by 16 percent as well as the fatigue strength of about 10 percent in lap joint.

The use of cold bonded adhesive in riveted repair can reduce the bearing load of the rivet hole and the relative displacement between the mating surfaces. The adhesive will also prevent from fretting. Therefore, it increases the fatigue and static strength. In static strength, the adhesive may increase the strength of about 10 percent in lap joint (see also the results in chapter 4). The fatigue strength of patch repair increases about 20 to 23 percent as compared to the sealant one [3].

Load transfers in the riveted joint are carried out by the following:

- Load through the rivet (bearing load), and
- friction/shear (adhesive faying) between two mating surfaces.

In the riveted joint, the rivet flexibility is an important parameter in load distribution along the rivet rows. Müller [4] said that the squeezing force has a significant effect in rivet flexibility. It is caused by the residual stress around the rivet hole including the interference between the rivet and the hole. Figure 4-1 shows the schematic load transfer in riveted joint.

In the case of mating surfaces, filled by an adhesive, the load transfer by the adhesive in the bonded -riveted joint shall be taken into account.

![Figure 4-1, Schematic load transfer mechanism](image-url)

1. Bypass Load,
2. bearing load, and
3. friction/shear load (adhesive faying).

Load transfer of a repair in fuselage skin can be divided into two situations. First, a repair which has a cut-out smaller than a half frame distance in flight direction, the so
called patch repair. Second, a lap joint repair in which a repair has a cut-out larger than a half frame distance or over a lap joint, where the middle of the cut-out will act as a lap joint. Therefore, the load transfer of this type of repair should be determined as a lap joint. Figure 4-2 shows the patch repair and the lap joint repair on fuselage skin.

![Diagram of patch and lap joint repair](image)

Figure 4-2. The patch repair and the lap joint repair.

### 4.2 Rivet Flexibility

It is recognized that the modeling of the load transfer in riveting joint is important in order to predict the fatigue life of a repair. There are several methods that can be used to calculate load transfer in a riveted joint, but most of them disregard the friction between the mating surfaces. Jarfall [17] and Swift used the assumption that the rivets behave as linear elastic members. When the rivet is loaded in shear, then the rivet is allowed to deflect as a member. This deflection of the rivet is called rivet flexibility.

The rivet flexibility can be defined by theoretical or experimental methods. There are several empirical equations for the rivet flexibility $C_b$ that can be used in load transfer calculations.

According to Jarfall [7],

$$
C_b = \frac{8}{t_{av} \cdot E_r} \left\{ 0.13 \left( \frac{t_{av}}{D} \right)^2 \left( 2.12 + \left( \frac{t_{av}}{D} \right)^2 \right) + 1.0 \right\}
$$

4-1

Where:

$t_{av}$ is average thickness of skin and patch,
$E_r$ is Young modulus of the rivet and,
$D$ is rivet diameter.

According to Swift [4],

$$C_b = \frac{1}{E_s \cdot D} \left[ 5.0 + 0.8 \left( \frac{D}{t_s} + \frac{D}{t_{pc}} \right) \right]$$ \hspace{1cm} (4-2)

Where:

$E_r$ is Young modulus of skin,
$D$ is rivet diameter,
$t_s$ is skin thickness and,
$t_{pc}$ is patch thickness.

According to Vlieger/ Broek [8]

$$C_b = \frac{1}{E_r \cdot D} \left[ 5.0 + 0.8 \left( \frac{E_r \cdot D}{E_s \cdot t_s} + \frac{E_r \cdot D}{E_{pc} \cdot t_{pc}} \right) \right]$$ \hspace{1cm} (4-3)

Where:

$E_r$ is Young modulus of the rivet,
$E_s$ is Young modulus of skin,
$E_{pc}$ is Young modulus of patch,
$D$ is the rivet diameter,
$t_s$ is skin thickness and,
$t_{pc}$ is patch thickness.

For our purposes, the Vlieger/ Broek [8] model has been chosen because it can be applied for different moduli of patch and skin materials.

Table 4-1 shows the rivet flexibilities of several joint configurations with NAS 1097 rivet as found by calculation. The rivet flexibility for a Glare 3 4/3 (1.95 mm) with Al 20204 (1.6 mm) joint results a similar rivet flexibility as for the merely aluminum joint with sheet thickness as of 1.6 mm, it means that the thicker Glare sheet did not result in an increase of the rivet flexibility. This is an advantage of Glare when it is used for repair of an aluminum skin. A comparable joint of a 2 mm Al 2024 T3 sheet to 1.6 mm Al 2024 T3 sheet lead to a lower rivet flexibility. It should be noted that a lower rivet flexibility is unfavorable for the distribution of the load transmission by the different rivet rows of a joint on a repair patch.
4.3 Load Transfer in Lap Joint Repair

The lap joint of Figure 4-3 can be modeled as shown in Figure 4-4. Each rivet is simulated as a linear elastic member under shear load. The portion of skin and doubler strip can be modeled as a bar, so the spring constant $C_b$ of skin and doubler can be obtained as follows:

$$C_b = \frac{L}{t_s \cdot W \cdot E_s}$$ \hspace{1cm} 4-4

Where:

$t_s$ is the skin thickness,
$W$ is the pitch length,
$E_s$ is the skin or doubler modulus, and
$L$ is a distance between the rivet rows.

![Figure 4-3, Model of riveted lap joint.](image)
Table 4-1, Rivet flexibility found by calculation in K.g⁻¹.mm²

<table>
<thead>
<tr>
<th>Skin</th>
<th>Douller</th>
<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Al 2024 T3</td>
<td>Glare 3 3/2</td>
<td>4.060 10⁻⁸</td>
<td>3.765 10⁻⁸</td>
<td>3.769 10⁻⁸</td>
</tr>
<tr>
<td>(1.4 mm)</td>
<td>(1.95 mm)</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Al 2024 T3</td>
<td>Al 2024 T3</td>
<td>3.631 10⁻⁸</td>
<td></td>
<td></td>
</tr>
<tr>
<td>(2 mm)</td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

To evaluate the load transfer in a lap joint, the simplifications should be made as follows:

1. It is considered as a two-dimensional problems by neglecting the deformation normal to the skin,
2. the rivet flexibility is calculated with the analytical method developed by Vlieger/Brock [8],
3. the lap joint is simplified as a number of strips in the loading direction, and
4. load transfer by friction or sealant is neglected.

Displacements in the skin on each rivet must be in compatibility with those in the other skin after accounting for the rivet displacement. From Figure 4-4, it can be written as follows:

\[ P_{s1} - P_{s2} = F_1, \quad P_{s2} - P_{s3} = F_2 \quad \text{etc.,} \]

and

\[ P_{d1} - P_{d1} = -F_1, \quad P_{d1} - P_{d1} = -F_2 \quad \text{etc.} \]

![Figure 4-4, Modeling of riveted lap joint.](image)

Where s and d denote skin and doubler.
Figure 4-5, Compatibility model of rivet displacement.

Figure 4-5, representing the compatibility model of the riveted lap joint implies:

\[ L + \Delta L_{d1} + \Delta R_2 = L + \Delta L_{sl} + \Delta R_f, \ldots \]  

Where:

- \( L \) is a distance between the rivet rows,
- \( \Delta L_{d1} \) is displacement of the doubler,
- \( \Delta L_{sl} \) is displacement of the skin, and
- \( \Delta R_f \) is the rivet displacement.

By deriving the displacement compatibility above, the relation of displacement and force can be expressed as follows:

\[ [\Delta f] = [C][F] \]  

Where:

- \([\Delta f]\) is compatibility displacement,
- \([C]\) is flexibility matrix, and
- \([F]\) is the rivet force.

By inverting the flexibility matrix \([C]^{-1}\), it can be derived that the load transfer from each rivet is:

\[ [F] = [C]^{-1}[\Delta f] \]
Figure 4-10 to Figure 4-18 show the load transfer calculation results for different doubler thicknesses, doubler materials and the rivet rows. A lap joint (three rivet rows) of Glare 3 4/3 (1.95 mm) to Al 2024 T3 (see Figure 4-10) with no faying material resulted in a favorable load transfer, i.e. the load transfer at the first and the third rivet row has an equal in load transfer. While for the thinner Glare 3 3/2 (1.45 mm) resulted in a little higher load transfer at the first rivet row as compared to the merely aluminum joint of 1.6 mm, as shown in Figure 4-10 and Figure 4-13. This is due to a thinner skin and it is also due to its little lower modulus of Glare as compared to the aluminum.

4.4 Load Transfer of the Bonded-Riveted in Lap Joint Repair

The strength characteristics of a bonded-riveted joint were considered for a further investigation of repair options. It increases the fatigue and static strength. To clarify which bonding conditions give the optimum fatigue and static strength, test programs for fatigue and static should be conducted to determine the effectiveness of the type of the bonding material to be used as a faying surface, before it is widely used in practice.

To determine the load transfer in bonded-riveted joints has become important, because it gives a better fatigue life. In obtaining the solution for calculating the load transfer, the assumptions are to be taken as follows:

1. The cement layer or adhesive can be expressed as a rivet, as shown in Figure 4-6,
2. it is considered as a two-dimensional problem by neglecting the deformation normal to the skin,
3. the rivet flexibility is calculated with the analytical method developed by Vlieger/Broek [8], and
4. the lap joint is simplified as a number of strips in the loading direction.

The displacement of the adhesive rivet is caused by a bending moment and a shear force. So the adhesive can be assumed as a cantilever beam with a concentrated load at the free end. From Gere/Timoshenko [9], the rivet flexibility of the adhesive can be determined as follows:

\[ C_b = \frac{t^3}{3.E_a.I} \left( 1 + \frac{3.f_s.E_a.I}{G.A.t^2} \right) \]

Where:

\( t \) is the thickness of the adhesive layer,
\( E_a \) is the adhesive modulus,
\( I \) is the moment inertia of adhesive,
\( A \) is the cross section area of the adhesive,
\( G \) is the adhesive shear modulus,
\( f_s \) is the form factor, for the rectangle is equal 1.2.
Equation 4-10 can be used to find the load distribution in the riveted-bonded joint by substituting this rivet flexibility into equation 3-9. Table 4-2 shows the calculation of the rivet flexibility for several types of adhesive varying with types of the doubler material.

From Figure 4-19 to Figure 4-30, it can be seen that the loads transfer throughout the rivets are much smaller than that of the merely riveted joint. It shows that the rivet hole is no longer critical.

<table>
<thead>
<tr>
<th>Doubler</th>
<th>Type of adhesive</th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>EC 9323</td>
<td>BA 2216</td>
<td>AW 106</td>
</tr>
<tr>
<td>Glare 3 1.4 mm</td>
<td>5.708 x 10^-10</td>
<td>1.042 x 10^9</td>
<td>6.108 x 10^-10</td>
</tr>
<tr>
<td>Glare 3 1.95 mm</td>
<td>5.708 x 10^-10</td>
<td>1.042 x 10^9</td>
<td>6.108 x 10^-10</td>
</tr>
<tr>
<td>Al 2024 1.6 mm</td>
<td>3.769 x 10^-8</td>
<td>3.769 x 10^-8</td>
<td>3.769 x 10^-8</td>
</tr>
<tr>
<td>Al 2024 2 mm</td>
<td>3.631 x 10^-8</td>
<td>3.631 x 10^-8</td>
<td>3.631 x 10^-8</td>
</tr>
</tbody>
</table>

Note Skin Al 2024 T3 1.6 mm

## 4.4.1 Adhesive Materials

There are many potentials in cold bonding adhesives existing today, but in our experiment, only three types of adhesive have been selected. The criteria for adhesive selection should include the operational flight, loading direction, exposure to extreme environment. Further criteria to be considered are:

- Adequate for long term durability,
- adequate for static shear strength,
- good fatigue property,
• creep resistance,
• long storage life,
• ease of handling,
• long application time or working life at room temperature, and
• minimum toxic release during the curing process.

The three types of adhesive used in this analytical as well as for static and fatigue experiment, are the following:

**Adhesive EC 2216**

Manufacturer : 3 M - Company  
Components : 2, Gray hardener paste A, white resin paste B.  
Mixing ratio by weight 140 : 100  
Storage : 1 year at < 45°C  
Pot life : 1 hour  
Bonding pressure : Contact pressure is sufficient  
Curing cycle : 3-7 days at room temperature (25°C) or 2 hours at 65°C  
Young modulus (E) : 1711 MPa  
Shear Modulus (G) : 586 MPa  
Reason : This adhesive has been used in fuselage bonded-riveted lap joint by Fokker with good experience.

**Adhesive AW 106**

Manufacturer : CIBA  
Components : 2, Hardener paste HV-953 U and resin paste AW 106  
Mixing ratio by weight 80 : 100  
Storage : 1 year at < 20°C  
Pot life : 1 hour  
Bonding pressure : Contact pressure is sufficient  
Curing cycle : 5 days at room temperature or 0.5 hours at 90°C  
Young modulus (E) : 2725 MPa  
Shear modulus (G) : 1000 MPa  
Reason : This adhesive has widely been used for general application in composite repair or metal bonding in maintenance.

**Adhesive EU 9323**

Manufacturer : 3 M - Company  
Components : 2, Red-violet hardener paste, white resin paste.  
Mixing ratio by weight 27 : 100  
Storage : 1 year at < 23°C  
Pot life : 1 hour (50 grams, 23°C room temperature)  
Bonding pressure : Contact pressure is sufficient  
Curing cycle : 15 days at room temperature (23°C) or 2 hours at 65°C  
Young modulus (E) : 2870 MPa  
Shear modulus (G) : 1070 MPa  
Reason : This newly adhesive has a higher shear strength.
In the load transfer calculation of a bonded riveted joint, there are two models of the joint used for each type of adhesive, which are a lap joint with two rivet rows and with three rivet rows. Figure 4-19 to Figure 4-30 show the results of the load transfer calculations for an Al 2024 T3 1.6 mm and Glare 3 in a bonded-riveted joint, where the upper joint is called “doubler”. In this calculation, the doubler is Glare 3 3/2 (1.4 mm) and Glare 3 4/3 (1.95 mm) or Aluminum 2024 T3 with thicknesses of 1.4 mm, 1.6 mm, 1.8 mm and 2 mm. From these load transfer calculations, it can be seen that the adhesive is taken the larger part of the load transfer, while the rivet is almost carrying no load at all. So from the fatigue point of view, the use of adhesive in the riveted joint as a faying material implies that the first rivet row will no longer be fatigue critical as long as the bonding is still intact. The difference of the load transfer in the first rivet row of the two rivet rows and the three rivet rows is very little for each type of adhesive.

4.5 Load Transfer in Patch Repair

There are many ways to carry out a repair on a damaged structure, e. g. by using hot/cold bonding, riveting or a combination of bonding and riveting. The main purpose of a patch is to transfer the load away from the damaged area. Consequently the stress level in the damaged area will be decreased to an acceptable level, but of course the repair should meet static and dynamic strength criteria.

There are two faying surfaces used in our experiments which are silicon rubber (sealant) as normally used in production and cold bonded adhesive. Due to the complexities of the bonded-riveted in load transfer mechanism, only experimental results are considered for this type of joint. The analytical analysis of a riveted repair with a sealant as faying surface will be evaluated in this section. Figure 4-7 shows the schematic drawing of a patch repair.

To evaluate the load transfer in patch repair, simplifications should be taken into account due to the three-dimensional characters of the problem which are as follows:

1. It is considered as two-dimensional problems by neglecting the deformation normal to the skin,
2. the rounded central square hole is simplified by a circular hole,
3. the rivet flexibility is calculated with an analytical method developed by Vlieger/Broek [8],
4. the beam point approach has been chosen to model the rivet holes,
5. the patch is simplified as a number of strips in the loading direction, and
6. load transfer by friction or sealant, is neglected.

The load distribution on the rivets in the riveted patch repair depends on the displacement compatibility between the skin, the rivet and the patch. A compatibility equation for a patch model according to Mahardika [6] is as follows:

\[ \delta_{ff} = \delta_{rad} + \delta_{cf} + \delta_{pc} \]  

4-9

where:
\( \delta_{ff} \) is displacement due to far-field stress,
\( \delta_{rad} \) is radial displacement,
\( \delta_{cf} \) is displacement due to concentrated force, and
\( \delta_{pc} \) is patch displacement in each rivet position.

The load distribution can be solved by inverting the following equation:

\[
[\delta_{ff}] = [C][P]
\]

Where:

\( [C] \) is the flexibility matrix, and
\( [P] \) is the rivet force.

Figure 4-8 and Figure 4-9 show the calculation results of the fastener load distribution in patch repair with aluminum and Glare sheets for an applied gross stress of 120 MPa. From Figure 4-8, it can be seen that an aluminum patch results in a higher load transfer in the first rivet row as compared to Glare 3 patch (Figure 4-9). It means that the a Glare patch attracts less load into the repair area than the aluminum, consequently the aluminum patch may result more critical for the fatigue stand point.

Figure 4-7, Schematic and dimension of patch repair, dimension in mm, rivet NAS 1097 AD 5, and cut-out 50x50 mm.
4.6 Conclusions

**Lap joint**

1. The load transfer of Glare 3 4/3 with 1.95 mm thickness is comparable with the aluminum 2024 T3 doubler of 1.6 mm. This is due to the low stiffness of Glare as compared to aluminum.

2. A Glare doubler which is 20 to 30 percent thicker than the skin being repaired shows that the first rivet row of the skin is no longer the critical area. It means that this repair is more damage tolerant. When the crack will show up at the doubler, then it will be visible to the inspector by means of visual inspection.

3. A significant load transfer reduction in the first rivet row has been found for a bonded riveted joint. Therefore these rivets row are no longer the fatigue critical area. The load transfer in the first rivet row is only about 0.5 to 2 percent of the total load, while these results are identical for the three types of adhesive as a faying surface. As a result, a longer fatigue life will be obtained by this type of joint.

4. For the two and the three rivet rows of bonded riveted joint, the difference of load transfer in the first rivet row is very small and can be assumed to be equal. But for a repair, three rivet rows will always be desired from a damage tolerance point of view.

5. As shown in the load transfer calculations three rows of rivets are adequate for a repair. An additional row of rivets in lap joint with sealant as faying surface, does not give any great benefit, it will result in an additional cost of repair.

**Patch repair**

1. The difference of the load transfers of the rivets at the first rivet row is very little for both type of patches material.

2. For a small cut-out, the load transfer of the inner row of the rivets or the rivet row adjacent to the cut-out is very small. This means that the inner rivet row of a patch repair is not a critical area.

3. A Glare patch is with a little lower elastic modulus relatively to the aluminum, results in less load attraction to the repair area as compared to the aluminum one. A thinner Glare of 90 percent of the skin being repaired does not give a significant static strength reduction (see also chapter 5). Furthermore, the result of the fatigue testing shows that Glare patching has a significantly higher fatigue strength than the aluminum one (see also chapter 6).
4.7 References


Figure 4-8, Fastener load distribution of Aluminum patch 1.6 mm.

Figure 4-9, Fastener load distribution of Glare patch 1.4 mm.
Figure 4-10, Fastener load distribution of Glare 3 doubler with three rivet rows

Figure 4-11, Fastener load distribution of Glare 3 doubler with four rivet rows
Figure 4-12, Fastener load distribution of Glare 3 doubler with five rivet rows

Figure 4-13, Fastener load distribution of aluminum doubler with three rivet rows
Figure 4-14, Fastener load distribution of aluminum doubler with four rivet rows

Figure 4-15, Fastener load distribution of aluminum doubler with five rivet rows
Figure 4-16, Fastener load distribution of Glare skin 1.4 mm and Glare doubler with three rivet rows.

Figure 4-17, Fastener load distribution of Glare skin 1.4 mm and Glare doubler with four rivet rows.
Figure 4-18, Fastener load distribution of Glare skin 1.4 mm and Glare doubler with five rivet rows.

Figure 4-19, Fastener load distribution of Glare doubler with two rivet rows, and AW 106 as a faying surface.
Figure 4-20, Fastener load distribution of Glare doubler with two rivet rows, and EC 2216 as a faying surface.

Figure 4-21, Fastener load distribution of Glare doubler with two rivet rows, and EU 9323 as a faying surface.
Figure 4-22, Fastener load distribution of Glare doubler with three rivet rows, and AW 106 as a faying surface.

Figure 4-23, Fastener load distribution of Glare doubler with three rivet rows, and EC.2216 as a faying surface.
Figure 4-24, Fastener load distribution of Glare doubler with three rows rivet, and EU.9323 as a faying surface.

Figure 4-25, Fastener load distribution of Aluminum doubler with two rivet rows, and AW 106 as a faying surface.
Figure 4-26, Fastener load distribution of Aluminum doubler with two rivet rows, and EC.2216 as a faying surface.

Figure 4-27, Fastener load distribution of Aluminum doubler with two rivet rows, and EU. 9323 as a faying surface.
Figure 4-28, Fastener load distribution of Aluminum doubler with three rivet rows, and AW 106 as a faying surface.

Figure 4-29, Fastener load distribution of Aluminum doubler with three rivet rows, and EC 2216 a faying surface.
Figure 4-30, Fastener load distribution of Aluminum doubler with three rivet rows, and EU 9323 as a faying surface.
Most repairs in service today were designed for static strength and fail safe criteria for aircrafts certified prior FAR Amendment 25-45 (1978), reference [1]. In the past, the design philosophy of repair was based on equal or better static strength only, without too much care about the fatigue life. A good repair design should have an equal static strength and no too over the static strength as compared with the original structure. The over-design of static strength can lead to degrade the fatigue life as compared to the original structure [9].

This chapter describes the results of static tests of the external doubler repairs. There are two repair conditions used in these static tests, i.e. a repair with a small cut-out (less than half frame distance), called patch repair and a repair with a larger cut-out (larger than a half frame distance), called lap joint repair, as explained in chapter 4 of part 1.

Many types of patch materials and faying surfaces are used in this experiment to obtain the optimum configuration for the static strength of a repair. Cold bonding adhesive and sealant are used as the faying surfaces.

5.1 Ultimate and Limit Loads
Limit loads are the maximum loads anticipated once in the aircraft operation life. Ultimate loads are limit loads multiplied by a safety factor. The structure of the aircraft shall be capable to withstand the limit load without suffering detrimental permanent deformation. The safety factor is 1.5, except for special areas, e.g. cockpit area, wing to fuselage attachment, etc.

Although an aircraft is not supposed to undergo loads larger than its limit load, in the case of aircrafts going beyond this limit or until its ultimate load, the structure must be strong enough to withstand the load once. For another reasons, a certain amount of reserve strength against a complete structural failure is necessary for any part of the structures, because of some factors; 1) The approximation involved in aerodynamic and structural analysis theory, 2) Variations in physical properties of materials, and 3) Variation in manufacturing and inspection. This reserve strength is most important, when the aircraft is slightly exceeding its ultimate load in an emergency condition. This condition might cause a permanent deformation, which requires a repair before a further flight after inspection.

5.2 Fuselage Stresses

The fuselage pressurization load creates two major stresses in the fuselage structure, which are the hoop tension stress (circumferential direction) and the longitudinal tension stress. If the fuselage is not circular, then the non circular cross-section will tend to become circular under internal pressure which imposes bending loads on the fuselage frame. This bending will increase the local stress in a single lap joint. Figure 5-1 shows the schematic diagram of pressurization loads on the fuselage. The hoop tension stress and the longitudinal tension stress can be written as follows:

- The hoop tension stress is:

\[
\sigma_{HT} = \frac{p \cdot R}{t} \tag{5-1}
\]

If there are no longitudinal stiffeners, then:

- the longitudinal tension stress is:

\[
\sigma_{LT} = \frac{p \cdot R}{2 \cdot t} \tag{5-2}
\]

Where:

\( p \) is the pressure differential,
\( R \) is the fuselage radius, and
\( t \) is the fuselage skin thickness.
In addition to the pressurization, fuselage bending loads occur due to moment (e.g. bending). A local bending stress $\sigma_{BT}$ should be added to the pressurization stress system. Therefore in determining the stress of a repair of the fuselage skin it is important to know, which stress is critical one for fatigue. According to experience, most of the repairs occurred at the sections below the windows due to accidental damage or corrosion.

![Diagram](image_url)

Figure 5-1, Fuselage skin stress due to the cabin pressure

The maximum shear flow $K_{allowable}$ transmitted to a repair must be equal or a little higher than the maximum static strength of the original structure. Usually the static strength is written as the running load ($K_{maximum}$) of the local skin thickness of an aircraft type. Table 5-5 is the maximum specific running load of the Fokker 28 taken from reference [4]. The maximum shear flow of a repair $K_{allowable}$ is:

$$K_{allowable} = \sigma_{doubler} \cdot \frac{t_{doubler} \cdot (P - d)}{P}$$  

5-3

Where:

$\sigma_{doubler}$ is the ultimate tensile strength of a repair doubler,
$t_{doubler}$ is the doubler thickness,
$P$ is the rivet pitch, and
$d$ is the rivet shank diameter.

The running load as given in references [3] and [4] for a specific skin thickness of 1.6 mm in the circumferential direction (see Table 5-5) is about 350 N/ mm. The difference between the pre-SB (pre Service Bulletin/ before modification) F 28/ 21-16 and the post-SB (after modification) F 28/ 21-16 is the thickness of the fuselage skin, while for the post SB the skin thickness is 1.2 mm, therefore this aircraft can fly at a higher altitude as compared with the pre-SB. The shear flow in the longitudinal direction from reference [4] for the specific skin thickness of 1.6 mm is about 200 N/ mm.

The shear flow in longitudinal direction is less critical than in the circumferential direction for the pressurization load as shown in equations 4-1 and 4-2. Therefore, the load in the circumferential direction is used as the basis for determining the static strength of a repair. Furthermore the repair should be adequate to resist the fatigue loads and static ultimate load caused by pressurization.

The frames of the fuselage will reduce the hoop stress in the skin by up to 14 percent [1]. The mid-bay skin between two frames of the fuselage will carry more load, and as a result this area will be more sensitive to fatigue and static loading.

### 5.2.1 Gross Stress and Net Stress

The maximum gross stress due to the hoop stress on the fuselage skin is also the maximum gross stress of the longitudinal lap joint in the circumferential direction. The maximum gross stress and the maximum net stress of the repair can be determined, based on the maximum running load of the lap joint as follows:

$$\sigma_{gross} = \frac{K_{maximum}}{t_{doubler}}$$  \hspace{1cm} 5-4

$$\sigma_{net} = \frac{K_{maximum}}{t_{doubler}} \cdot \frac{P}{P - d}$$  \hspace{1cm} 5-5

Due to the notch sensitivity of Al 2024 T3 or Glare, the maximum gross stress should be calculated, based on the blunt-notch strength. Table shows the material properties of the aluminum and Glare sheets used for analytical calculations of the static strength taken from references [5] and [26].
Table 5-1, Properties of Al 2024 T3 and Glare 3, [5] and [26].

<table>
<thead>
<tr>
<th>Sheet</th>
<th>Lay-up</th>
<th>Thickness (mm)</th>
<th>$E_{11}$, $E_{22}$ (GPa)</th>
<th>$\sigma_{0.2}$ (MPa)</th>
<th>$\sigma_{ult}$ (MPa)</th>
<th>$\sigma_{blunt}$ (MPa)</th>
<th>$\sigma_{br}$ (MPa)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Al 2024</td>
<td>monolithic</td>
<td>variable</td>
<td>72</td>
<td>359</td>
<td>455</td>
<td>440</td>
<td>903</td>
</tr>
<tr>
<td>Glare 3</td>
<td>3/2-0.2</td>
<td>1.1</td>
<td>54</td>
<td>265</td>
<td>695</td>
<td>404</td>
<td>695</td>
</tr>
<tr>
<td>Glare 3</td>
<td>3/2-0.3</td>
<td>1.40</td>
<td>58</td>
<td>280</td>
<td>645</td>
<td>397</td>
<td>740</td>
</tr>
<tr>
<td>Glare 3</td>
<td>4/3-0.3</td>
<td>1.95</td>
<td>57</td>
<td>275</td>
<td>660</td>
<td>387</td>
<td>725</td>
</tr>
</tbody>
</table>

Note: $\sigma_{br}$ is bearing strength ultimate with $e/D = 2$

5.3 Fastener Strength

The maximum shear flow of the original structure must be smaller than the fastener strength ($K_{fastener}$) of the repair. If the maximum shear flow is greater than the flow which the repair fasteners can carry, then the repair has an insufficient static strength. Thus this type of repair should be categorized as category “D repair” (see also repair categorization in chapter 6). It means that the repair must be upgraded to category C or better before a further flight.

The rivet shear strength $P_{su}$ is:

$$P_{su} = \sigma_{su} \cdot \pi \cdot r^2$$  \hspace{1cm} 5-6

Where:

- $\sigma_{su}$ is the ultimate shear allowable of the fastener, and
- $r$ is the nominal shank radius.

So the fastener static strength ($K_{fastener}$) of a repair is:

$$K_{fastener} = \frac{N_{fr} \cdot P_{su}}{P}$$  \hspace{1cm} 5-7

Where:

- $N_{fr}$ is the number of fastener rows, and
- $P$ is the fastener pitch.

The maximum shear flow for a specific skin thickness and a specific aircraft type should be provided by the aircraft manufacturer.

5.4 Specimen Manufacturing

The skin material used in these experiments was Clad Al 2024 T3 1.6 mm and the patch materials were made of Clad Al 2024 T3 1.6 mm, Glare 3 3/2 (thickness 1.4
mm) or Glare 3 4/3 (thickness 1.95 mm). As mentioned in chapter 2 of this part, Glare 3 has been developed for fuselage skin applications, e.g. crown fuselage skin which is a fatigue sensitive area. The typical properties of those materials used in this experiment are shown in Table 5-1.

The materials are cut to final outer dimensions using a cutting machine, after which the central cut-out for the patch repair specimen and the rivet pilot holes are milled on a numerically controlled machine. The countersunk holes are made by a drill press with a conventional countersunk tool. All patches and some skins were countersunk to simulate the external patch repair. The countersunk skin is used to determine the influence of the countersink holes underneath the external doubler concerning the static strength. Only the normal countersunk hole with the minimum cylindrical part of 1/3 of the skin thickness was observed. Special care has been taken to avoid the “knife edge effect”. Figure 5-2 and Figure 5-3 show the cross section of the repair at the rivet hole with the countersunk and the non countersunk flaws. The dimensions of the patch repair and the lap joint are shown in Figure 5-4 and Figure 5-7 respectively. In the lap joint, the upper side represents the external doubler and the lower side represents the fuselage skin.

There are four types of faying surfaces used as mentioned in chapter 4. Before applying the faying surface and installation of the repair doubler, the area of the skin or patch to be jointed must receive a good surface preparation. While more details about surface preparations are presented in the next section.

The rivet type used in this experiment is NAS 1097 AD 5. They are carefully milled to the “final length” as needed. The extended shank through the skin and the patch is equal to 6 mm (1.5 times the rivet diameter) ± 0.1 mm. Riveting is performed on the force controlled riveting machine of the Structures and Materials laboratory. The squeezing force of NAS 1097 AD 5 was about 16500 N to achieve a manufacturing head of 1.5 times the shank diameter, as normally used in current repairs.

The specimen configurations of the patch repairs and the lap joint repairs are presented in Table 5-2 and Table 5-3 respectively. The number of specimen to be tested for each configuration is 2 and all specimens have been carefully inspected visually before the test.

Table 5-2, External patch repair configurations.

<table>
<thead>
<tr>
<th>Skin</th>
<th>Patch Material</th>
</tr>
</thead>
<tbody>
<tr>
<td>Clad Al 2024 1.6 mm</td>
<td>Glare 3 3/2 1.45 mm</td>
</tr>
<tr>
<td>Clad 2024 T3 1.6 mm</td>
<td>Spc # 1.1 Sealant</td>
</tr>
<tr>
<td></td>
<td>Spc # 1.2 Sealant</td>
</tr>
<tr>
<td></td>
<td>Spc # 1.3/ AW 106</td>
</tr>
</tbody>
</table>

Note, No faying surface for all patch specimens.
Table 5-3, Lap joint configurations.

<table>
<thead>
<tr>
<th>Skin</th>
<th>External doubler material</th>
</tr>
</thead>
<tbody>
<tr>
<td>Al 2024 1.6 mm, sealant</td>
<td>Al 2024 1.6 mm, Bonding</td>
</tr>
<tr>
<td>Spc # 2.1</td>
<td>Spc # 2.2 / 2216 B/A</td>
</tr>
<tr>
<td>Spc # 2.6 / (ctr)</td>
<td>Spc # 2.7 / AW 106</td>
</tr>
<tr>
<td></td>
<td>Spc # 2.3/(ctr) EU 9323</td>
</tr>
<tr>
<td></td>
<td>Spc # 2.8/ EU 9323</td>
</tr>
<tr>
<td>Al 2024 1.6 mm</td>
<td></td>
</tr>
<tr>
<td></td>
<td>Glare 3 3/2, sealant</td>
</tr>
<tr>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td></td>
</tr>
<tr>
<td>Glare 3 4/3, sealant</td>
<td></td>
</tr>
</tbody>
</table>

Note: - The Al 2024 T3 sheets were in clad condition.
- ctr means countersunk

5.4.1 Surface Preparation

For a good corrosion protection and a good adhesion on the damaged surfaces, the metal surfaces must receive a good surface preparation. Recommendations for surface treatment are given in Mil-A-5090 D or DTD 915 B. The surface treatment should preferably be followed by an anodizing process. If the anodizing process can not be carried out, the alternative of this surface treatment is by Alodizing (see par 5.4.2).

The anodizing process is commonly used during field repair or workshop repair. The process has been widely used in the aircraft structural production for a long time. The anodizing process especially for field repair, is a very simple process which can be carried out with a minimum support and within a very short time.

The use of Alodine 1200 S surface treatment has been approved by most aircraft manufacturers and has been accepted by airworthiness authorities. The process has also been shown to be reliable in service. Therefore it is used in our experiments for surface treatment of both types of repair.

5.4.2 Alodine 1200 S

Alodine is a conversion coating on the aluminum surface. This process produces an aluminum oxide coating on the surface, which will protect the base metal in contact with the atmosphere. The surface treatment processes of the Alodine 1200 S are as follows:

1. Preparation of Alodine solution; Prepare the Alodine solution in a container which is acid resistant, preferable a polyethylene container. Put the Alodine 1200 S powder to tap water with a ratio of 22 gram of powder for one liter of the final solution. Stir the solution until the powder has been dissolved. Then after dissolving of the powder, add 3.3 cm³ concentrated nitric acid for each liter of the solution.

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2. Cleaning: Degrease for oil, dirt, etc., the patch and the skin surfaces on both sides which to be jointed with clean cloths damped with the solvent, or if possible by oil degreasing, deoxidine, and etching then drying. As a substitute of deoxidine may be used the Scotch Brite type S, which is also very fine to remove the oxide layer, and then clean with the cloths dampened with solvent and drying. This process must be done for both surfaces.

3. Alodizing: Immediately after the cleaning process above, apply the prepared alodine to all surfaces. The reason of this, is to avoid the oxidizing process after the cleaning, which may influence the quality of the treatment. The Alodine solution should thoroughly be wetting the surface, but shall also be aware to avoid to dry up the surface. After that allow the alodine to develop an acceptable aluminum oxide coating, which is ranging from clear light iridescent yellow color to clear gold or brown. The immerse time is from 50 seconds until 3 minutes depending on the temperature and condition of the solution.

4. After treatment; Rinse the parts thoroughly with clean water, and then dry the surface. To speed up the drying, the surface can be blown with dry air or using a soft cloth. Please be aware that the coating is very soft before its completely dry.

5.4.3 Preparation of Adhesive Mixture and Curing

A correct adhesive mixture is decisive for a good cold bonded joint, utmost care has to be taken when weighing and mixing the adhesive components. As usual, the adhesive is stored in a refrigerator, hence they shall be allowed to become at ambient temperature before opening the tin. Care should be taken for keeping the adhesive and hardener container closed to avoid the moisture absorption.

The adhesive can be mixed by weight or volume as mentioned by each manufacturer. The mixing ratio of the adhesive used in this experiment can be seen in chapter 3.5.1. Use different spatulas for both components, stir the adhesive and hardener for at least 5 minutes to obtain a homogeneous paste and uniform in color. Apply the adhesive on both surfaces to be bonded within its application time. After that, join the two surfaces and then immediately apply the rivets before its “tack free”. Then allow the adhesive to cure as mentioned on its specification. This installation is called “wet installation”.

5.5 Test Result and Discussion

External Patch

Table 5-4 and Figure 5-5 present the results of the static test of the external patch repair. Most specimens fractured in the skin at the first rivet row. The difference in the static strength between the aluminum and the glare patch is very small. The running load and the fastener strength of both types of patches were beyond the minimum stresses found by the calculation using equations 4-4 and 4-5.
Cold bonding adhesive (AW 106) as the faying surface did not increase the static strength of the patch repair. During the test when the load reached about 70 percent of the maximum static strength, the delamination always occurred at the area close to the first rivet row. This delamination can be identified by the sound when the delamination occurred. It can also be detected visually by a magnifying glass as a relative displacement of the skin and the patch at the end of the overlapping.

The test results prove that in patch repair the surrounding structure can still easily carry the ultimate load. Therefore, the cut-out area is no longer critical. Increasing the patch thickness in patch repair will also not increase the static strength, because the failures occurred along the first rivet row.

**Lap joint**

All static test results of the lap joint specimens satisfied the minimum static strength, found by the calculation using equations 4-4 and 4-5. The test results are shown in Table 5-6 and Figure 5-6.

Most failures of the specimens with sealant as the faying surface occurred by rivet shear because the shear flow, was higher than the fastener strength. This type of failure occurred on both types of doubler (Glare and aluminum) with sealant as a faying surface.

The static strength of the combination joint between Glare and aluminum is somewhat lower as compared to the merely aluminum joint with sealant as a faying. A merely aluminum joint with sealant is used as the basis for comparison in this experiment. While the specimens with Glare 3 3/2 1.4 mm as an upper skin has a little lower static strength or is reduced about 15 percent as compared to the reference specimen. From Figure 5-6 and Table 5-6 it can be seen that a thinner Glare used as the upper skin results in a lower static strength as compared to the other thicker Glare specimens, even with the same failure mode (rivet shear). This is due to the countersunk hole on the thinner skin which resulted in a higher bearing stress to the skin. Therefore a thinner Glare with the countersunk hole on top of it has higher bearing stress, which leads to the rivet shear.

The countersunk hole underneath the upper skin or doubler of the aluminum specimen (# 2.6) with sealant as a faying, reduced the static strength with about 5 percent as compared to the reference specimen. The cold bonding adhesive as a faying, increased the static strength with about 10 percent as compared to the reference specimen. The failures modes of these specimens change from rivet shear to skin fracture. The increasing bearing stresses due to countersunk hole in both skin and patch reduces the static strength, while this condition of the repair frequently happened in the field repair.

The specimens with cold bonding adhesive as a faying have a higher static strength than the sealant one, also the failure mode changed from the rivet shear to the skin or doubler fractures at the first rivet row. From Figure 5-6 and Table 5-6, it can be seen
that there are no big differences of the static strength by using different types of adhesives.

From the static strength point of view, there is no significant indication to use those three types of adhesive as a faying surface. But it should be noted that the static strength is not the only parameter to be considered. Other important parameters have to be considered when evaluating the type of cold bonding adhesive, e.g., the durability of the bonding, the deterioration due to the environment, surface treatment, etc. The EC 2216 BA manufactured by 3 M has been used by Fokker in fuselage lap joint of F-28 for already more than 20 years with successful experience in service. From the airworthiness stand point, the durability test of the adhesive shall be conducted before it is used in the bonded-riveted joint as a part of the certification.

From the static test results, it can be seen that the adhesive as the faying surface, has a significant effect in the static strength. The delamination of the adhesive occurs roughly at 60 to 80 percents of the maximum load. This delamination can be noticed by the sound when the delamination occurs.

5.6 Conclusions

Patch Repair

1. The static test results of the external patch repair for both types of patch materials showed that the repairs satisfy the minimum static strength caused by the hoop load in the circumferential direction.

2. A thinner Glare patch repair with a thickness of about 90 percent of the skin, did not degrade the static strength of the repair. It can be said that the Glare patch has met the static strength requirements for a small cut-out (less than a half frame distance).

3. Cold bonding AW 106 as a faying surface did not increase the static strength of the external patch repair.

Lap Joint

1. The static test results of the lap joint repair proved that the repair has satisfied the minimum static strength caused by the hoop load in circumferential direction.

2. Sealant as a faying surface, resulted in a lower static strength as compared with the cold bonding.

3. The use of the cold bonding as a faying surface in the bonded-riveted repair increased the static strength significantly.
4. The use of an adhesive in a riveted repair reduces the bearing stresses significantly in the rivet hole, and as a result it change the failure mode from the rivet shear to the skin failure.

5. There are no significant differences of the static strength due to the use of the three types of cold bonding of the lap joint.

6. The combination of a thinner Glare sheet to an aluminum sheet in a lap joint has a lower static strength than the merely aluminum lap joint of the reference specimen # 2.1.

7. A thinner Glare of 1.4 mm on specimen # 2.4 resulted in a lower static strength as compared with Glare of 1.95 mm on specimen # 2.5, while the failure was the rivet shear for both types of Glare specimens. This result shows that there is a relationship between the bearing stress and the shear strength of the rivets.
5.7 References


[22] Aircraft Inspection and Repair, Acceptable Methods, Techniques, and Practices, Department of Transportation Federal Aviation Administration, United States of America, *AC 43.13 - IA*.


Table 5-4, Static test result of the external patch repairs with a rivet pitch 23 mm.

<table>
<thead>
<tr>
<th>Spec #</th>
<th>Result in (N)</th>
<th>Gross Stress Min 219 (MPa)</th>
<th>Net Stress Min 276 (MPa)</th>
<th>$K_{\text{allowable}}$ Min 350 (N/mm)</th>
<th>Remark of failure</th>
</tr>
</thead>
<tbody>
<tr>
<td>Spec # 1.1-1 Al-Al</td>
<td>180067</td>
<td>375</td>
<td>421</td>
<td>600</td>
<td>At first rivet row</td>
</tr>
<tr>
<td>Spec # 1.1-2 Al-Al</td>
<td>182066</td>
<td>379</td>
<td>426</td>
<td>607</td>
<td>At first rivet row</td>
</tr>
<tr>
<td>Spec # 1.2-1 Glare-Al</td>
<td>174093</td>
<td>363</td>
<td>407</td>
<td>580</td>
<td>At first rivet row</td>
</tr>
<tr>
<td>Spec # 1.2-2 Glare-Al</td>
<td>182073</td>
<td>379</td>
<td>426</td>
<td>607</td>
<td>At first rivet row</td>
</tr>
<tr>
<td>Spec # 1.3-1 Al-Al, Bond</td>
<td>180258</td>
<td>376</td>
<td>422</td>
<td>601</td>
<td>At clamping</td>
</tr>
<tr>
<td>Spec # 1.3-2 Al-Al Bond</td>
<td>184133</td>
<td>384</td>
<td>431</td>
<td>614</td>
<td>At first rivet row</td>
</tr>
</tbody>
</table>

Note: Min is minimum value from calculation above

Table 5-5, The maximum shear flow of Fokker F-28.

<table>
<thead>
<tr>
<th>Skin thickness $t_{\text{skin}}$ in mm</th>
<th>$K_{\text{maximum}}$ in (N/mm)</th>
<th>Pre SB F28/21-16</th>
<th>Post SB F28/21-16</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Longitudinal rows</td>
<td>Circumferential rows</td>
<td>Longitudinal rows</td>
</tr>
<tr>
<td>0.8</td>
<td>165</td>
<td>200</td>
<td>N/A</td>
</tr>
<tr>
<td>1.0</td>
<td>200</td>
<td>250</td>
<td>200</td>
</tr>
<tr>
<td>1.2</td>
<td>200</td>
<td>250</td>
<td>200</td>
</tr>
<tr>
<td>1.2 $\leq$ 2.8</td>
<td>200</td>
<td>350</td>
<td>200</td>
</tr>
<tr>
<td>$&gt;2.8$</td>
<td>200</td>
<td>415</td>
<td>200</td>
</tr>
</tbody>
</table>

Figure 5-2, The cross section with non countersunk flaw.
Table 5-6, Static test result of the lap joint repairs.

<table>
<thead>
<tr>
<th>Spec #</th>
<th>Result in N</th>
<th>Gross stress Min 219 (MPa)</th>
<th>Net stress Min 276 (MPa)</th>
<th>$K_{\text{allowable}}$ Min 350 (N/mm)</th>
<th>Type of Failure</th>
</tr>
</thead>
<tbody>
<tr>
<td>2.1-1</td>
<td>74398</td>
<td>332</td>
<td>418</td>
<td>531</td>
<td>Rivet shear</td>
</tr>
<tr>
<td>2.1-2</td>
<td>74901</td>
<td>334</td>
<td>421</td>
<td>535</td>
<td>Rivet shear</td>
</tr>
<tr>
<td>2.2-1</td>
<td>81833</td>
<td>365</td>
<td>460</td>
<td>585</td>
<td>Patch fracture</td>
</tr>
<tr>
<td>EC 2216</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>2.2-2</td>
<td>81658</td>
<td>365</td>
<td>459</td>
<td>583</td>
<td>Skin fracture</td>
</tr>
<tr>
<td>2.3-1</td>
<td>82881</td>
<td>370</td>
<td>465</td>
<td>592</td>
<td>Skin fracture</td>
</tr>
<tr>
<td>EU 9323</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>2.3-2</td>
<td>79093</td>
<td>353</td>
<td>444</td>
<td>565</td>
<td>Skin fracture</td>
</tr>
<tr>
<td>2.4-1</td>
<td>64109</td>
<td>286</td>
<td>360</td>
<td>458</td>
<td>Rivet shear</td>
</tr>
<tr>
<td>Sealant</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>2.4-2</td>
<td>63582</td>
<td>284</td>
<td>357</td>
<td>454</td>
<td>Rivet shear</td>
</tr>
<tr>
<td>2.5-1</td>
<td>69508</td>
<td>310</td>
<td>390</td>
<td>496</td>
<td>Rivet shear</td>
</tr>
<tr>
<td>Sealant</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>2.5-2</td>
<td>70858</td>
<td>316</td>
<td>398</td>
<td>506</td>
<td>Rivet shear</td>
</tr>
<tr>
<td>2.6-1</td>
<td>71100</td>
<td>317</td>
<td>399</td>
<td>508</td>
<td>Rivet shear</td>
</tr>
<tr>
<td>Sealant</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>2.6-2</td>
<td>70304</td>
<td>314</td>
<td>395</td>
<td>502</td>
<td>Rivet shear</td>
</tr>
<tr>
<td>2.7-1</td>
<td>79023</td>
<td>353</td>
<td>444</td>
<td>564</td>
<td>Patch fracture</td>
</tr>
<tr>
<td>AW 106</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>2.7-2</td>
<td>79077</td>
<td>353</td>
<td>444</td>
<td>565</td>
<td>Patch fracture</td>
</tr>
<tr>
<td>2.8-1</td>
<td>79415</td>
<td>355</td>
<td>446</td>
<td>567</td>
<td>Skin Fracture</td>
</tr>
<tr>
<td>EU 9323</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>2.8-2</td>
<td>80758</td>
<td>361</td>
<td>453</td>
<td>577</td>
<td>Skin at Clamp</td>
</tr>
</tbody>
</table>

Note: *Skin or patch fractured at first rivet row*

![Diagram](image.png)

Figure 5-3, The cross section with the countersunk flaw.
Figure 5-4, The external patch repair.

Figure 5-5, Failure strength of external patch repair with rivet pitch of 23 mm.
Figure 5-6, Failure strength of lap joint with rivet pitch of 20 mm.

Figure 5-7, Lap joint specimen.
6. FATIGUE STRENGTH OF A REPAIRED FUSELAGE STRUCTURE

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From an economical point of view, the ideal aircraft should cost the least and remain operative for the longest time. It should also be capable of carrying a larger payload to a destination in shorter time and using less fuel than competitive aircraft. These combined factors, ideally realized, will result in the lowest operating cost.

Assume, there are two aircraft manufactures who build an aircraft of a similar type, size, range and payload. An analysis of the operating cost, performance, technical aspects and other statistical data might reveal that both aircrafts are comparable. However, upon evaluating the potential fatigue life, one aircraft may excel the other one for less inspection and repair for thousands of flying hours. Less inspection and repair mean low direct operating cost.

Long fatigue life is dependent upon superior design, material, production and quality of workmanship, one without the other means a shorter fatigue life. Long fatigue life is something that commercial customers expect from today's aircraft.

A single load cycle will do no harm to the structure, because the load is far bellow the static failure load. The effects of fatigue are cumulative. In this regard, we might think of metal as having a history. For example, if the pressurization load actions on a fuselage lap joint are repeated many times, crack can be initiated at each of the first rows of the fastener holes. This situation is called Multiple Site Damage (MSD). If this type of failure occurs in a the fuselage skin, a catastrophic damage or accident may result. The Aloha accident [33] is an example of an MSD occurrence.

Fatigue failures can be divided into three stages:

1. fatigue damage occurs on a submicroscopic scale,
2. a crack occurs and grows, and
3. final rupture takes place.

In restoring a damaged part, a repair has to be performed to reduce the stress level at
the damaged area. The common repair method used today is the riveted repair to join a doubler to a skin. A repair on an airframe structure can easily degrade the fatigue life if extreme care is not taken. In the past, repair calculations were based on equal or better static strength only, without too much care about the fatigue life. This philosophy may lead to static strength over design, which can result in considerable loss in fatigue life, as compared to the original structure.

Sealants or another material may be used as a faying surface to prevent fretting between the two mating surfaces in a joint and thus crack initiation due to fretting can be avoided. In fatigue test described in this chapter, an adhesive and sealants were used on the faying surfaces. The purpose of the adhesive is to reduce the stress level around the rivet hole due to by-pass load and bearing load as explained in chapter 4. Safety and economy are the primary concern in the aircraft repair. The quality of a repair with respect to fatigue problems, involves different aspects as follows:

1. Repair location (Primary Structural Element or not),
2. present number of flight cycles,
3. present number of flight hours,
4. present number of years (age),
5. reason of the defect (dent, crack, wrinkling, corrosion, loose fastener or combinations of them),
6. accessibility to the repair area,
7. availability of ground time,
8. surface preparation quality,
9. qualified personnel for doing the repair, and
10. careful inspection to ensure that the repair is done in a good manner.

Since 1978, Airworthiness Authorities mandates the evaluation of fatigue and damage tolerance analysis of repair. Therefore, the purpose of our experiments and analysis is to improve the current repair method. This improvement might be achieved through the use of cold bonded-riveted repair or using a new repair material, Glare 3. Glare is known as a material which is almost insensitive to fatigue damage which will result in a better damage tolerance of a repair. By using existing procedures and product support for aluminum repairs in our experiment, massive investments for tooling, procedure, training, etc., may be avoided when it will be implemented in practice These give indication about what is going to come in this chapter.

6.1 Pressurization Load and Other Loads on Fuselage Repairs

The pressurization load of the cabin is most important in fatigue of the cabin. Aluminum alloys with a high yield to ultimate strength ratios are prone to rapid tearing at low stress levels. This is especially important in view of possible cracks, accidental damage, corrosion or inadequate repair, which may trigger off a disastrous rupture of the entire fuselage.

Pressurization creates two major stresses in a fuselage, the hoop tension stress \( \sigma_{HT} \) and the longitudinal tension stress \( \sigma_{LT} \). These two stresses can be rewritten from equations 5-1 and 5-2 as follows:
The Hoop tension stress is:

\[ \sigma_{HT} = \frac{p.R}{t} \]  

The longitudinal tension stress is:

\[ \sigma_{LT} = \frac{p.R}{2.t} \]

The location of the repair is an important aspect to be considered with regards to the loads applied on this area, e.g. in the crown section, the longitudinal load is higher due to the fuselage bending. The repair location will also affect the inspection method, the visible/detectable crack size, the inspection threshold and the repeat inspection interval. Careful assessment of the damage area is required in order to define an appropriate repair design.

### 6.2 The Inspection Threshold and the Repeat Inspection Interval

In order to ensure the airworthiness of the aircraft, the fatigue damage must be detected before it reaches its fail safe design limit. Therefore it is necessary to establish the inspection threshold and the repeat inspection interval program to meet the damage tolerant requirement. Those items must be included in the maintenance program. To establish a maintenance program, the environmental deterioration (ED), accidental damage (AD), fatigue damage (FD), and program implementation guidelines from the aircraft manufacturer (e.g. Corrosion Prevention and Control Program) must then be taken into account during the evaluation. Fatigue damage very much depends on the minimum detachable crack length by NDT and the critical crack length of the structural part. These are again a matter of the design, stress level, and material properties. Repair and inspection are two important aspects of the safety in aircraft operation. It is essential that all aspects concerning a repair shall be properly recognized, e.g. repair cost, damage tolerance aspect, aircraft down time, etc.

There is experience that an improper inspection could lead to an accident. Apparently, when a crack could be undetected during an inspection, while the crack, is probably present, it means that such an inspection program is not effective and needs to be updated.

The inspection threshold is the time until the first fatigue inspection of a repair must be carried out within a specific period (i.e. flight cycle for fatigue or calendar time for corrosion). The repeat inspection is the continuation of the inspection threshold, within a specific flight cycle or calendar time. Figure 6-1 shows the specific time for each critical period of structural part.

The inspection threshold is;
\[ T = \frac{N_{\text{NDT}}}{\text{Scf}} \]

The repeat inspection interval is:

\[ R_{\text{NDT}} = \frac{N_{\text{Critical}} - N_{\text{NDT}}}{\text{Scf}} \]

\[ R_{\text{visual}} = \frac{N_{\text{Critical}} - N_{\text{visual}}}{\text{Scf}} \]

Where:

\( N_{\text{NDT}} \) and \( N_{\text{visual}} \) are numbers of cycles, where the crack is detectable by NDT or visual inspection. In our experiments, the Eddy current test is used for NDT because this method can detect the crack in the skin underneath a doubler. \( N_{\text{critical}} \) is the critical cycle where the crack length becomes unstable, and \( \text{Scf} \) is the scatter factor. Figure 6-2 shows the inspection threshold and a repeat inspection interval with a scatter factor 3.

![Figure 6-1, Inspection schedule](image)

The inspection threshold \( T \) and the repeat inspection interval \( R_{\text{NDT}} \) and \( R_{\text{visual}} \) of a repair, will be affected by the following factors:

- Location of a repair (different stress levels and spectra),
- size of cut-out (half frame distance or larger),
• doubler thickness (stress distribution and secondary bending),
• doubler modulus of elasticity (stress distribution and load attraction),
• the rivet diameter (shear stress and bearing stress),
• the rivet/bolt type (fastener flexibility),
• the use of former countersunk holes underneath the doubler at the run-out rows (increased bearing stress),
• the rivet pitch (stress distribution),
• the edge margin (shear stress at edge of cut-out or doubler),
• number of the rivet rows (stress level at first fastener row),
• aircraft utilization (fatigue cycles),
• countersunk depth on the doubler (knife edge effect), and
• row distance (bending stress level at the fastener hole).

![Figure 6-2](image.png)

Figure 6-2, Inspection threshold and repeat inspection interval.

In maintenance, the probability to detect a damage or crack in an inspection depends on the several factors, i.e.:

• damage size,
• method of inspection (visual, Eddy current, etc.),
• view distance,
• lighting condition,
• surface condition (clean/dirt area),
• accessibility to area (easy/difficult), and
• construction form (simple construction/complex).

Concerning the damage tolerance analyses, the detectable crack length plays a dominant role to assess the inspection threshold and repeat inspection interval. Therefore, the informations concerning the $N_{NDT}$, $N_{visual}$ and $N_{critical}$ are
important in order to prepare an inspection program for a repair. In our experiments, the fatigue cycles are recorded and the crack is checked by NDT (Eddy current) and visual inspection under the maintenance facility conditions.

The relation of the Probability of Detection (POD) and the crack length [10] is:

\[
POD = 1 - e^{-\left[ \frac{(a-a_0)}{(\lambda-a_0)} \right]^\mu}
\]

6-6

Where:

\(a_0\) is the minimum detectable crack length by each method of NDT in mm (see Table 6-1),
\(a\) is a crack length of a structural part in mm,
\(\lambda\) is the minimum incremental for each method of NDT, in mm (see Table 6-1), and
\(\mu\) is a constant (see Table 6-1), from reference [10].

Equation 6-6 shows the probability of a detection in one inspection by an inspector in relation with the crack length, but in this case, an inspector may not detect a crack, then the probability is 1 - POD. Figure 6-3 shows the relation between the probability of detection and the crack length. The three parameters of \(a_0\), \(\lambda\), and \(\mu\) depend on the method or equipment used in the inspection. These values are shown in Table 6-1.

![Figure 6-3, Typical probability of one time inspection, [10].](image)

As part of the continuing airworthiness of an aircraft, the structural inspections are the major task in aircraft overhaul, in order to reduce, or even to delete the inspection task, it is essential to improve the design of the structural parts. The inspection man-hours, e.g. the inspection of fuselage skin lap joints of Boeing 737-100/200 could cost $14,000 - 37,000 for each inspection, reference [31].
Table 6-1, Probability parameters.

<table>
<thead>
<tr>
<th></th>
<th>Ultrasonic</th>
<th>Penetrant</th>
<th>Eddy Current</th>
<th>X-Ray</th>
<th>Visual</th>
</tr>
</thead>
<tbody>
<tr>
<td>$\mu$</td>
<td>0.5</td>
<td>0.5</td>
<td>0.5</td>
<td>0.5</td>
<td>0.5</td>
</tr>
<tr>
<td>$a_0$</td>
<td>0.12</td>
<td>0.13</td>
<td>0.15</td>
<td>0.3</td>
<td>0.4</td>
</tr>
<tr>
<td>$\lambda$</td>
<td>0.04</td>
<td>0.06</td>
<td>0.06</td>
<td>0.12</td>
<td>0.2</td>
</tr>
</tbody>
</table>

In regard with the above issue, there are wishes from the airlines for NDT inspection, which are as follows:

1. the airframe structure should be easy to inspect, to access and to repair,
2. visual inspection rather than NDT, and
3. reliable NDT without necessity of any special knowledge or experience is desired.

6.3 Fatigue Life Prediction

Fatigue life prediction of a riveted joint is a simple process faced with several complexities. Load magnitude and load transfer mechanism are very important elements of the process. Environmental damage and accidental damage are other complexities in determining the fatigue life of the actual structure. The application of damage tolerance requirement is essential, to ensure that the fatigue damage must be covered by an appropriate inspection program.

The scatter of the fatigue life of identical specimens could be caused by several factors, i.e. riveting quality, hole quality, squeezing force, etc. Other factors which influence the fatigue life of a simple single lap joint or an aircraft joint are, the curved geometry, loading conditions, bending and torsion. To include these into a simple coupon test, is not easy. Müller [11] mentioned that the scatter factor used today varies from 2 to 5. This factor should be applied, when determining the fatigue life of the actual aircraft joints to account for scatter. The scatter factor is intended to adjust the result carried out by the experiment due to its imperfection, during manufacturing or other hidden factors, which are not encountered in the fatigue diagram (S-N curve) which resulted from experiments. Furthermore, if there is any reason that can prove the practical relevance of the test results, then the scatter factor can be reduced to a more realistic value. The influence of the countersunk holes underneath a doubler at the run-out row of the rivets of the external patch repair, can reduce the fatigue life by 30 to 60 percent depending on the depth of the countersunk itself [19]. Although extensive fatigue testing will improve the position to judge the fatigue problems, a consistent relation between test results and fatigue life predictions by analytical analysis, may not be expected.

- With respect to fatigue in repair of the riveted joints, are many fatigue prediction techniques used by industries today, e.g. Jarfall’s stress severity factor [15].

This method try to predict the fatigue life of the riveted joint. However, it does not include the effect of corrosion or accidental damage, while the crack growth is also
not included in the prediction.

This fatigue life prediction is limited to the total life of a specimen. In view of the limitations, it is clear that an inspection program is required to ensure the structural integrity of the aircraft.

The severity factor developed by Jarfall can be used to analyze the fatigue life of a repair in a qualitative way, i.e., a fastener with a higher severity factor will have a lower fatigue life. This severity factor is used in the Repair Calculation and Classificiation Program (RCCP) in chapter 7. The reason of the using the severity factor, is because this method account for a more elaborate calculation of the load transmission. The complex joint does not allow such simplifications as used by Swift [8].

6.3.1 Jarfall's Stress Severity Factor

Jarfall [15] has developed a method to predict the fatigue life of a riveted joint, the so-called Stress Severity Factor Concept. This method compares the fatigue life of several types of joints which are obtained experimentally, and at the end, it uses the stress severity factor in a qualitative comparison of fatigue lives. The Severity Factor (SF) is the ratio of the peak local stress in the fastener hole ($\sigma_{max}$) with the reference gross stress. The peak stress is corrected for the hole surface condition factor $\alpha$ and the filling factor $\beta$. The peak stress $\sigma_{max}$ of the fastener hole is derived from the local stresses caused by the load transfer $\Delta P$ (see Figure 6-4) and the stress caused by the bypass load $P$ (see Figure 6-5). Jarfall made a more elaborated load transfer calculation, which includes the rivet flexibility (see the load transfer in chapter 4 of part 1). While the local stress caused by the load transfer $\Delta P$ is affected by the bearing distribution factor $\theta$ as a function of the skin thickness and the rivet diameter. As a result, SF is a fatigue factor that accounts for:

- Fastener type,
- Hole filling (interference, hole surface quality, etc.),
- Fastener load distribution,
- Bearing stress distribution due to rivet rotation, and
- The stress concentration factor due to the load transfer and bypass load.

$$SF = \frac{\alpha \cdot \beta}{\sigma_{ref}} \left( K_{th} \cdot \theta \cdot \frac{\Delta P}{d \cdot t} + K_{tg} \cdot \frac{P}{w \cdot t} \right)$$  \hspace{1cm} (6-7)

Where:

$\alpha$ is the hole surface condition factor, e.g. $\alpha = 1$ for standard drilled hole,
$\beta$ is the hole filling factor, e.g. $\beta = 0.75$ for a rivet and 1 for an open hole,
$\theta$ is the bearing distribution factor,
$K_{th}$ is the bearing stress concentration factor,
$K_{tg}$ is the stress concentration factor due to bypass load,
$d$ is the rivet diameter,
$t$ is the skin thickness,
$w$ is the rivet pitch distance, and
$\sigma_{ref}$ is the applied gross stress.

Figure 6-4, Local stresses due to load transfer $\Delta P$

Figure 6-5, Local stresses due to the bypassing load $P$

The three factors of $\alpha$, $\beta$ and $\theta$ vary with the skin thickness, fastener type and installation process. These factors can be found in ref [2]. However, for a design or a repair of an aircraft joint, a more extensive test is a necessity, to achieve more relevant results.

Once the load distribution is obtained from the load transfer calculation in chapter 4 of part 1 and $\alpha$, $\beta$ and $\theta$ value adopted are the stress severity factor can be calculated for the proposed joints at the most critical rivet row, by using equation 6-7. A qualitative
comparison can then be made to find the most critical rivet row with the highest $SF$. The method could also be used to predict the fatigue life of the proposed joints by comparison with the baseline model (qualitative comparison, e.g. lap joints with equal severity factors at the critical rivet row, will have equal fatigue lives). When a joint configuration has a higher severity factor as compared to the baseline configuration, then it will have a lower fatigue life. As an illustration, Figure 6-6 shows the $SF$ for the aluminum skin at the 3rd rivet row from three different repair joint configurations as calculated with the equation 6-7. Note that the $SF$ for the three repair doublers are rather similar.

The severity factor concept ignores the load transmission caused by friction between the mating surfaces, where in reality it gives a significant contribution to the load transfer, as mentioned by Schijve [22].

![Diagram](image)

Figure 6-6, Severity factor ($SF$) of several joint configurations in a repair.

6.4 The Effect of the Riveting Quality to the Fatigue Life on the Riveted Joint

This section introduces the riveting problems concerning the effect on the fatigue life of riveted joint which frequently occurred. Failure to make good riveting quality, leads to lowering the fatigue life. Very often in field repair, when the space is very restrictive, the rivets are not bucked properly which result in a clinched installation (see Figure 6-7). When it occurs, the hole is not correctly filled, and the effect of the rivet swelling does not appear. Thus, the benefit of residual compressive stresses due to the radial pressure, does not exist. Figure 6-8 illustrates the radial pressure due to rivet swelling. When a rivet was bucked, it would create an equivalent of an
interference fit pin in the hole and produced a radial pressure to the hole boundary. Hence, the material around the hole will become plastically deformed, called local hardening. This effect gives a significant improvement in the fatigue life. Therefore, the estimation of the fatigue life of this type of riveting installation should be assumed as an open hole and loaded loose fit hole, as shown by Swift [8].

Figure 6-7, Rivet clinching installation

Figure 6-8, Radial pressure due to rivet swelling

The stress concentration factor $K_f$ of an open hole is approximately 3.0 for wide thin sheet with $\sigma_{br} / \sigma_{gr} = 0$, reference [8]. While the filled hole is approximately 2.0. This reduction is due to the empty hole and is allowed to deflect from circle to ellipse as shown in Figure 6-9. But for the filled hole, the pin or rivet will restrain the deformation of the hole boundary, and then it reduces the stress concentration factor at the rivet hole.
6.5 Fatigue Experiments on External Doubler Repairs

To establish information on external doubler repairs of fuselage skin, a test program has been conducted in Structures and Materials Laboratory of the Faculty of Aerospace and Engineering of the Delft University of Technology. As mentioned before, a repair of a fuselage skin can be considered as a patch repair or as a lap joint repair, depending on the severity of the damage. There is a general guide-line to determine whether a repair should be considered as a patch repair or a lap joint repair. When a repair has a cut-out smaller than a half frame distance in flight direction, then it should be considered as a patch repair [16]. For a repair with a cut-out, larger than a half frame distance, or over a lap joint, it must be considered as a lap joint, repair. Both types of repairs have been tested in our experiment. The results will be used as initial information for further development of the repair method.

Three types of the fatigue tests have been carried out, i.e. (1) external patch repair, (2) single lap joint repair, and (3) a curved panel with a single lap joint loaded by internal pressure in the so called Pressurized Fuselage Skin Test System (PFSTS) [34]. The purposes of the external patch and the single lap joint fatigue tests are to obtain the baseline fatigue lives and to establish fatigue data for future repair assessment.

The purpose of the third type of test with internal pressure is to find-out the effect of panel curvature and secondary bending on the fatigue life. These test results are then used to update the fatigue data found in a single lap joint of the plane sheet. Furthermore the updated data will be used to determine the repair threshold and the repeat inspection interval.

6.5.1 Fatigue Performance of External Patch Repair

The external patch doubler is fitted to the skin made of Aluminum 2024 T3 with 1.6
mm thickness. The fatigue tests were performed on a computer controlled servo-hydraulic fatigue machine. The constant amplitude (CA) uniaxial loading was employed to give the fatigue loading. The maximum gross stress $\sigma_{max}$ was 120 MPa and the stress ratio R was 0.05. The frequencies of 5 to 10 Hz were chosen, according to Hartman [12], that frequency between 0.1 to 66 Hz had very little effect on fatigue life. This rule is not without an exception; a very low frequency of loading is coupled to a shorter fatigue life due to corrosion. Most of the specimens were tested to failure. The tests were done at room temperature in a laboratory climate. The number of specimens for each test configuration was of 2. The specimen geometry is shown in Figure 6-11.

The patch repair configurations are shown in Table 6-2. The variables are the patch materials, the faying surface material and the countersunk hole underneath the repair doubler. The crack initiation cycles were recorded, to determine the inspection threshold by visual inspection. The location of the crack initiation and the type of failure were also recorded.

<table>
<thead>
<tr>
<th>Patch</th>
<th>Skin</th>
<th>Sealant</th>
<th>Faying</th>
<th>Faying</th>
<th>Glare 3 3/2</th>
<th>Glare 3 3/2</th>
<th>Al 2024</th>
<th>Glare 3 3/2</th>
</tr>
</thead>
<tbody>
<tr>
<td>Al 2024</td>
<td>1.6 mm</td>
<td>1.45 mm</td>
<td>1.45 mm</td>
<td>1.6 mm</td>
<td>1.45 mm</td>
<td>1.45 mm</td>
<td>None</td>
<td>None</td>
</tr>
<tr>
<td>Faying:</td>
<td>Faying:</td>
<td>Faying:</td>
<td>Faying:</td>
<td>Faying:</td>
<td>None</td>
<td>None</td>
<td>None</td>
<td>None</td>
</tr>
<tr>
<td>AW 106</td>
<td>Sealant</td>
<td>Sealant</td>
<td>AW 106</td>
<td>None</td>
<td>None</td>
<td>None</td>
<td>None</td>
<td>None</td>
</tr>
<tr>
<td>Non-csk</td>
<td>spc #1</td>
<td>spc #2</td>
<td>spc #3</td>
<td>spc #4</td>
<td>spc #9</td>
<td>spc #10</td>
<td>spc #5</td>
<td>spc #6</td>
</tr>
<tr>
<td>Csk</td>
<td>spc #5</td>
<td>spc #6</td>
<td>spc #7</td>
<td>spc #8</td>
<td>spc #9</td>
<td>spc #10</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

Note: - Nonesk is non countersunk, and Csk is countersunk

A countersunk hole is needed for aerodynamic smoothness to minimize the aircraft drag. Therefore, fuselage repairs are frequently dealing with this problem. A countersunk hole increases the bearing stress in the rivet hole. Furthermore, the existing countersunk holes underneath the repair doubler at the run-out row (see fig 7-3), can not always be avoided. To investigate the consequences of such countersunk holes, the skin was countersunked, see Figure 6-10.

The aluminum patch repair on specimen #1 is exactly following the procedure of the Structure Repair Manual (SRM) issued by the aircraft manufacturer. It is used in order to obtain the basic fatigue data for comparison with new repairs.

Fretting damage at the mating surfaces could easily initiate a crack if there is no faying surface applied. The purpose of the faying surface in the riveted joint is to avoid fretting and to ensure airtightness. As to the airtightness, it will prevent an opening the crevice between sheets to the atmosphere. Eventually it is also protected against the crevice corrosion. Cold bonded adhesive in a bonded-riveted repair as a faying, is to increase the fatigue life as mentioned above. The bonding reduces the bearing stresses in the rivet hole as shown in the load transfer calculation (see chapter 4 of part 1), and it thus will delay the crack to initiate from the rivet hole. In the load transfer calculation, it could be seen that, in cold bonded-riveted lap joint, the
critical rivet row transfers only about 1 to 2 % of the total load. Therefore this area is no longer the critical area for fatigue, as long as the bonding is intact.

![Diagram showing non-countersunked and countersunked skin]

**Non countersunked skin**

**Countersunked skin**

Figure 6-10, Countersunk flaw in a repair

### 6.5.1.1 Manufacturing of Patch Repair Specimens

The materials used were clad Al 2024 T3 1.6 mm for the skin. The patch materials were Glare 3 3/2 1.4 mm or clad Al 2024 T3 1.6 mm. The manufacturing process of the fatigue test specimens were done in the same manner as for the static test. The details of this process and the typical properties of these materials can be seen in chapter 5 of part 1.

There were two types of faying surface materials, namely AW 106 manufactured by Ciba as cold bonding adhesive and sealant PR 1422 B2 (silicon rubber) manufactured by Courtaulds Aerospace South East Asia. No faying surface were applied to specimens # 9 and # 10. The purpose was to comprehend the effectiveness of the application of a sealant or an adhesive as a faying surface to prevent fretting during the fatigue loading.

Figure 6-11 and Figure 6-12 show the specimen dimension and test set-up installation, respectively.
6.5.1.2 Test Results and Discussion of Patch Repair

There were three types of fractures occurring during the fatigue testing:

- type 1: skin fracture at the first rivet row (see Figure 6-11),
- type 2: skin fracture at the edge of the patch (see Figure 6-11),
- type 3: clamping fracture, and

The fatigue lives shown in tables are the average value of two specimens. The three types of fractures above will be discussed for their significances as follows:

![Diagram of a patch repair showing three types of fractures](image)

Figure 6-11, The external patch repair, cut-out 50 by 50 mm.

**Fracture Type 1**

Most fractures of type 1 were initiated at a rivet hole, but for the specimen with no accompanied by a faying surface; the crack initiated from the mating surface in between the rivet pitch.

For bonded riveted specimens, the fracture of the skin is through the rivet holes and it occurs, when the bonding between the end of the overlap and the first rivet row failed. Because this type of failure occurred, the bonding quality is apparently important. The failures of the bonding may occur between the alodine coating and the metal or between the alodine coating and the adhesive. Figure 6-13 shows the schematic of the
bonding. Both types of bonding failure occurred.

![Fatigue test set-up of patch repair.](image)

**Fracture Type 2**

This type of fracture occurred in bonded-riveted specimen only. The fracture occurred in the skin at the edge of the patch. It is a favorable type of fracture, because it means that the bonding quality is goods, the adhesive is effective in increasing the fatigue life of the repair. It shows that under these circumstances, the fatigue life has come up to our expectation and that the dominant factor of the fatigue life on the patch repair is the skin itself. Inevitably, the abrupt change of the cross section at the end of the patch, causes a stress raising due to the eccentricity. When this type of fracture happened, it usually occurs at very high fatigue life. Cold bonding in riveted repair is expected to have this type of fracture.

**Fracture Type 3**

Clamping failure were due to the discontinuity of the thickness at the reinforced specimen clamping area. In order to prevent this type of failure, it was decided to reduce the thickness of the reinforcing plates. Then this type of fracture was eliminated.
6.5.1.3 Effect of Faying Materials on fatigue strength

**Aluminum Patch on Non Countersunk Skin**

An aluminum patch with no faying surface resulted in a lower fatigue life as compared to the specimens with a faying surface, as shown in Figure 6-28. When the crack initiated at the mating surfaces due to fretting, it was found that the initiation did not occur at the rivet hole edges but between the rivets of the first rivet row. Employment of the PR 1422 B2 sealant as a faying surface, increased the fatigue life with some 90 percent as compared to the non sealed one. Specimen #1 with a single crack had a longer fatigue life as compared to another one, as shown in Table 6-8.

![Figure 6-13, Bonding system](image)

For the cold bonding adhesive (Araldite AW 106) as a faying surface, the fatigue life increased about 11 percent compared to result for the sealant as a faying surface. The failure was type 1 with multiple site cracks. The crack growth between the first visible crack (visual inspection) and the failure is about 10k to 20 kc. When a single crack occurred, the crack growth is sometimes considerably longer, i.e. about 80 kc, as shown in Table 6-8.

**Aluminum Patch on Countersunk Skin**

The effect of the cold bonding on the countersunk skin was increasing the fatigue life about 20 percent as compared to the sealant one. It means that the cold bonding as a faying surface has a significant effect on the fatigue life of the repair. All failures were of type 1. The crack initiated from the rivet holes at the first rivet row. While the crack growth between visible cracking (visual inspection) and failure was about 14 kc. Figure 6-29 and Table 6-8 show the results of the fatigue testing.

**Glare Patch on Non Countersunk Skin**

A Glare patch with no faying surface, also resulted in a shorter fatigue life as compared to the faying one. Crack initiation was mainly due to fretting between the mating surfaces. Multiple site cracks were detected. The use of PR 1422 B2 sealant as a faying surface extended the life with about 80 percent as compared with the non faying surface (see Figure 6-30). It means that the sealant was effective as an anti fretting material. The failures were type 1 at the first rivet row. The crack growth between the first visible crack (visual inspection) and the failure was about 30 kc.
When the cold bonding (AW 106) is used as a faying surface in bonded-riveted repair, the fatigue life increased another 30 percent as compared to the sealed repair. Again, the adhesive effectively worked as an anti fretting and contributed significantly in taking a part in the load transfer. One of the cold boned-riveted Glare patch specimens failed with type 2 and the crack growth period was very short, about 10 kc. The other specimen failed with type 3 at the clamping. Nevertheless it should be noted that both specimens failed at a very high fatigue life as compared to the basic specimen # 1, see Figure 6-30 and Table 6-8.

After renewing the holes for the clamping of the specimen with failure in the clamping area, the test was continued until 1,846 kc, and then the test was stopped for a residual strength test. The residual strength was of 165 kN (384 MPa) or 100 percent as compared to the static test result found in chapter 5 of part 1.

The difference of the fatigue life of the bonded-riveted Glare patch specimens was due to the spew fillet (over-filling the adhesive at the edges of the patch), as shown in Figure 6-14. The specimen with a shorter fatigue life was no spew fillet due to unintentionally being cleaned during the manufacturing process. It shows that the spew fillet is acting as a significant role in bonded-riveted joint.

**Glare Patch on Countersunk Skin**

Araldite as a faying surface on the countersunk skin, extended the fatigue life with about 20 percent as compared to the sealant one, as shown Table 6-8 and Figure 6-31. Again, the cold bonding improved the fatigue life. Multiple site cracks always occurred, and the visible crack growth period was about 50 kc. The failure types of these specimens were type 1.

![Diagram](image)

**Figure 6-14, Spew fillet or over filling**

**6.5.1.4 Conclusions the Effect of Faying Materials**

1. Both faying surface materials (sealant and adhesive) as a means against fretting in the riveted joint, have a favorable effect on the fatigue life. For the Specimens without anti fretting, the fatigue life was significantly lower. Table 6-3 shows summary of the influence the faying surface on the fatigue life of the patch repair.

2. Cold bonding as a faying surface on countersunk and on non countersunk skin, increased the fatigue life significantly as compared to the sealant one.

3. The crack growth period between the first visible cracking and the failure for both
types of the faying surfaces was approximately equal.

4. For the non countersunk skin, the use of a cold bonding, changed the failure mode from the rivet hole to the end of the patch. It means that the rivet hole is no longer the fatigue critical area. These fatigue test results confirmed that the adhesive has a significant effect in the load transfer of the bonded-riveted repair, as mentioned in chapter 6.4.1 of part 1.

Table 6-3, The influence of faying material in fatigue life on patch repair.

<table>
<thead>
<tr>
<th>Patch materials</th>
<th>Non faying (kc)</th>
<th>Sealant, PR 1422 B2 (kc)</th>
<th>Adhesive, AW 106 (kc)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Al 2024 1.6 mm</td>
<td>200</td>
<td>381</td>
<td>424</td>
</tr>
<tr>
<td>Glare 3 3/2 1.45 mm</td>
<td>356</td>
<td>628</td>
<td>832</td>
</tr>
</tbody>
</table>

6.5.1.5 Effect of Countersunk Holes Underneath an External Patch Repair

As mentioned before, the existing countersunk holes in the skin of the rivet holes at the run-out row (see also Figure 7-3), will have a significant effect on the fatigue life due to the high bearing stress in the rivet hole. For this reason, the present section shows the results of test to determine the reduction factor due to the usage of the former countersunk holes, at the run-out row of the rivet in a repair.

The Effect of the Countersunk Hole for an Aluminum Patch with the Sealant PR 1422 B2

Table 6-8 and Figure 6-32 show the fatigue test results. The use of the existing countersunk holes at the rivet holes of the run-out row, resulted in a significant reduction of the fatigue life. The fatigue life of the aluminum patch repair was reduced with about 50 percent as compared to the non countersunk specimen. This reduction is in agreement with the factor given by reference [16], i.e. between 30 - 60 percent. Fatigue was initiated by a single crack from a rivet hole at the mating surface, but within a very short time it was followed by the multiple site cracking. The crack growth period between the first visible crack (visual inspection) and failure was about 40 kc.

The Effect of Countersunk Hole for an Aluminum Patch with Araldite

Table 6-8 and Figure 6-33 show the fatigue test results. The reduction of the fatigue life, due to the countersunk holes at the run-out row of the aluminum patch, was about 50 percent as compared to the non countersunk case. The type of failure was type 1. The crack was initiated from a rivet hole by a single crack, followed by multiple site cracking within a very short time. The crack growth period was about 35 kc.

The Effect of Countersunk Hole for an Glare Patch with the Sealant PR 1422 B2

Table 6-8 and Figure 6-34 show the fatigue test results. The countersunk holes
underneath the Glare patch, reduced the fatigue life with about 50 percent as compared to the non countersunk case. The type of failure was type 1. The crack initiated from a rivet hole by a single crack at the mating surface, then again followed by multiple site cracking within a very short time. The crack growth was about 74 kc.

The Effect of Countersunk Hole for an Glare Patch with Araldite

Table 6-8 and Figure 6-36 show the fatigue test result. The fatigue life of the Glare patch on the countersunk skin is reduced about 50 percent as compared to the non countersunk case. The failure was type 1. The crack initiated by a single crack from a rivet hole at the mating surface, and then it was followed by multiple site cracking. The crack growth period was about 45 kc.

6.5.1.6 Conclusions on the Effect of Using of the Former Countersunk Hole Underneath the External Patch Repair

1. Using the former countersunk holes underneath an external patch repair at the run-out row of rivets, reduced the fatigue life significantly. This is due to the increase of the bearing stress as a result of the countersunk hole. Consequently, a fatigue crack can be initiated earlier as compared to the non countersunk case, as suggested by Swift [8]. Table 6-4 shows a summary of the results.

2. The fatigue life reduction due to using of the former countersunk holes in the skin is about 40 to 50 percent for both types of faying (sealant and adhesive).

3. Crack initiation always started with a single crack from a rivet hole at the mating surface, followed by multiple site cracking within a very short time.

Table 6-4, The influence of the use the former countersunk hole at run out row on patch repair.

<table>
<thead>
<tr>
<th>Patch materials</th>
<th>Non countersunk (kc)</th>
<th>Countersunk (kc)</th>
<th>Reduction, in percent</th>
</tr>
</thead>
<tbody>
<tr>
<td>Al 2024 T3 1.6 mm, sealant PR 1422-B2</td>
<td>381</td>
<td>185</td>
<td>50</td>
</tr>
<tr>
<td>Al 2024 T3 1.6, AW 106 (adhesive)</td>
<td>424</td>
<td>227</td>
<td>45</td>
</tr>
<tr>
<td>Glare 3 3/2 1.45 mm, sealant PR 1422-B2</td>
<td>628</td>
<td>313</td>
<td>50</td>
</tr>
<tr>
<td>Glare 3 3/2 1.45 mm, AW 106 (adhesive)</td>
<td>832</td>
<td>391</td>
<td>55</td>
</tr>
</tbody>
</table>

6.5.1.7 The Effect of Patch Material on the Fatigue Life

The Effect of the Patch Material on a Non Countersunk Skin with PR 1422 B2

Table 6-8 and Figure 6-36 show the test results of the fatigue testing. Glare as a patch
material, increased the fatigue life with about 60 percent as compared to the aluminum patch. The type of failure was type 1. The Glare patch had a favorable effect on the crack growth; the life between the first visible crack (visual inspection) and failure was about 120 kc. This means that a more damage tolerant repair can be achieved. The crack was initiated by a single crack for quite a long time, and it was then followed by multiple site cracking.

**The Effect of the Patch Material on the Countersunk Skin with PR 1422 B2**

Table 6-8 and Figure 6-37 show the results of the fatigue testing. The effect of the Glare patch on the countersunk skin was increasing the fatigue with about 70 percent as compared to the aluminum one, again a significant life improvement. The type of failure was type 1. The crack growth period was about 50 kc. The crack was initiated by a single crack from a rivet hole at the mating surface, and then it was followed by multiple site cracking.

**The Effect of Patch Material on a Non Countersunk Skin with Araldite**

Table 6-8 and Figure 6-38 show the results of the fatigue testing. The Glare patch gives a favorable effect in increasing the fatigue life with about 90 percent as compared to the aluminum patch. This repair configuration has the highest fatigue life. The failure was type 1 and the crack growth period was about 30 kc. The crack was initiated by a single crack, and then it was followed by a multiple site cracking within a very short time.

**The Effect of Patch Material on the Countersunk Skin with Araldite**

Table 6-8 and Figure 6-39 show the results of the fatigue testing. A Glare patch on the countersunk skin increased the fatigue life with about 70 percent as compared to the aluminum one. The failure was type 1 and the crack growth period was about 50 kc. The crack was initiated by a single crack from the rivet hole at the mating surface, and then followed by a multiple site crack within a short time.

**6.5.1.8 Conclusions of the Effect of Patch Materials**

1. Glare as a patch repair material with both types of faying surfaces, has a favorable effect on the fatigue life as compared to the aluminum. Table 6-5 shows a summary of the influence of patch materials on patch repairs.

2. The low stiffness and the fatigue insensitivity of Glare in a patch repair with 90 percent thickness of the skin being repaired (Glare $\approx 1.45$ mm and aluminum $\approx 1.6$ mm) was not critical from the fatigue standpoint. This repair causes less load attraction to the repair area, and therefore a higher fatigue life of the repaired skin can be obtained. The test results confirm the concept of soft patching [24]. The thinner Glare patch, 90 percent of the repaired skin thickness is also satisfying the static strength requirement, as mentioned in chapter 5 of part 1.

3. No crack initiation or failure of the patches (Glare and aluminum) was found
during the experiments.

4. The crack growth life after Glare patching is considerably longer if the former rivet holes are non countersunk.

5. A combination of Glare patching with cold bonding of the bonded-riveted repair results in the highest fatigue life in our experiment.

Table 6-5, The influence of the patch materials in fatigue life of patch repair.

<table>
<thead>
<tr>
<th>Type of faying</th>
<th>Al 2024 T3 1.6 mm (kc)</th>
<th>Glare 3 3/2 1.45 mm (kc)</th>
<th>Increase, in percent</th>
</tr>
</thead>
<tbody>
<tr>
<td>Sealant, non countersunk</td>
<td>381</td>
<td>628</td>
<td>65</td>
</tr>
<tr>
<td>Sealant, countersunked skin</td>
<td>185</td>
<td>313</td>
<td>70</td>
</tr>
<tr>
<td>AW 106 (adhesive), non countersunk</td>
<td>424</td>
<td>832</td>
<td>95</td>
</tr>
<tr>
<td>AW 106 (adhesive), countersunked skin</td>
<td>227</td>
<td>391</td>
<td>70</td>
</tr>
</tbody>
</table>

6.5.1.9 Conclusions of the Patch Repair

1. Most cracks were initiated at a rivet hole of the first rivet row in the skin. These cracks initiated randomly along the first rivet row, because of the imperfections during the manufacturing processes, such as the drilling, riveting, etc., and the small differences in the load transfer between the rivets at the first row (see Figure 4-7 or figure 4-1, in part 1). Rivet holes #1 or #8 of the first rivet row are not always the most critical area.

2. Glare as a patch material results in a higher fatigue life as compared to the aluminum.

3. The adhesive (AW 106) as a faying surface in patch repair results in a higher fatigue life as compared to the sealant.

4. The use of the former countersunk holes at the run-out row underneath the doubler reduced the fatigue life significantly, about 40-50 percent for both types of faying surfaces.

6.5.2 Fatigue Experiments on the Lap Joint Repair

The lap joint specimens with 3 rivet rows were used in our experiments, to simulate a repair having a cut-out larger than half frame distance in flight direction, or a repair over a lap joint in the fuselage skin (see chapter 4 of part 1). In a repair with a large cut-out, the joints will act purely as a lap joint, therefore a lap joint specimen can be used to determine its fatigue life.

Different possible doubler materials and faying surfaces were again variable in these
experiments. The clad Al 2024 T3 1.6 mm is used as the skin (lower part of the joint). Then the upper part or the so called doubler is made of clad Al 2024 T3 1.6 mm, Glare 3 3/2 1.4 mm or Glare 3 4/3 1.95 mm. The faying materials are again sealant PR 1422 B2 or cold bonding with AW 106, EC 2216 or EU 9323.

The thinner Glare doubler has again a thickness of 90 percent of the skin. However as mentioned in chapter 5 of part 1, it satisfies the static strength requirement.

The tests were carried out in a 10 ton MTS fatigue machine. The frequency was chosen in range from 10 Hz to 14 Hz. The tests were done at room temperature in an air conditioned laboratory. The constant amplitude fatigue test program involves four different maximum stress levels, which are of 80 MPa, 100 MPa, 120 MPa and 140 MPa with the stress ratio R of 0.05. Specimens were tested to failure unless the specimens were still intact after more than 1.6 million fatigue cycles. These specimens were then tested under static loading to determine their residual strength. The purpose of the fatigue tests was to generate S-N curve data for each joint configuration. Furthermore these results can be used in determining the fatigue life of a repair, at it is affected by the different doubler materials or faying surfaces.

Figure 6-15 and Figure 6-16 show the test set-up of the lap joint and the dimensions of the specimens.

![Figure 6-15, Test set-up of lap joint fatigue testing.](image)

6.5.2.1 Manufacturing of Lap Joint Specimens

The manufacturing process of the fatigue test specimens was the same as for the static specimens. Details of the manufacturing process can be seen in chapter 5 of part 1.
The countersunk holes were prepared with a conventional countersunk tool and the countersunk depth was of 1/3 of the skin thickness.

For each repair configuration two identical specimens were made for fatigue testing. Table 6-6 shows the repair configurations of the lap joints.

Table 6-6, The configurations of the lap joint repair specimens.

<table>
<thead>
<tr>
<th>Series</th>
<th>Doubler</th>
<th>Faying</th>
<th>Countersunk holes in skin</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Material</td>
<td>Thickness (mm)</td>
<td></td>
</tr>
<tr>
<td>A</td>
<td>2024 T3</td>
<td>1.6</td>
<td>PR 1422 B2 Sealant</td>
</tr>
<tr>
<td>B</td>
<td>Glare 3 3/2</td>
<td>1.45</td>
<td>no</td>
</tr>
<tr>
<td>C</td>
<td>Glare 3 4/3</td>
<td>1.95</td>
<td>AW 106, adhesive</td>
</tr>
<tr>
<td>D</td>
<td>Glare 3 3/2</td>
<td>1.45</td>
<td>EC 2216 B/A adhesive</td>
</tr>
<tr>
<td>E</td>
<td>Glare 3 3/2</td>
<td>1.45</td>
<td>EC 9323 B/A adhesive</td>
</tr>
<tr>
<td>F</td>
<td>Glare 3 3/2</td>
<td>1.45</td>
<td>PR 1422 sealant</td>
</tr>
<tr>
<td>G</td>
<td>2024 T3</td>
<td>1.6</td>
<td>yes</td>
</tr>
<tr>
<td>H</td>
<td>2024 T3</td>
<td>1.6</td>
<td>EU 9323 B/A adhesive</td>
</tr>
</tbody>
</table>

Figure 6-16, The dimensions of the lap joint specimen.
6.5.2.2 Test Results and Discussion of The Lap Joint Repair

Alclad 2024 T3 1.6 mm doubler with PR 1422 B2 as a faying surface is used as the basis for comparison of the fatigue life in this experiment. The reason for this is, that the repair procedures and the materials employed on this specimen are in accordance with the Structure Repair Manual that is commonly used in the current repair. Therefore, this specimen can be used to represent the fatigue live of the current repair method.

The classification of the fracture mode is required in order to comprehend the test results. There were three types of fracture modes which occurred during the test:

Fracture type 1

Type 1 fracture is identified by a fracture of the specimen at the edge of the overlap or at the end of the overlap (see Figure 6-16). This type of fracture occurred on the bonded-riveted specimens only.

Fracture type 2

Type 2 fracture occurred along a critical rivet row either the upper row or the bottom row, see Figure 6-16. This type of fracture occurred on both types of faying (sealant or adhesive).

Fracture type 3

Type 3 fracture occurred at the clamping of the specimens. This type of fracture is undesirable because it is a premature failure of the specimen.

The fatigue performance (S-N curve) of the individual test series of the lap joint repairs is summarized in appendix C. These test results can be used to determine the fatigue life of a repair by using (e.g.) the severity factor method. The fatigue test results found in these experiments can also be used for a qualitative comparison of the fatigue quality of the proposed repair methods, as explained in sub-chapter 6.3.1 of part 1.

6.5.2.3 The Effect of an Adhesive on Fatigue Strength

Table 6-10 to Table 6-17 show the results of the fatigue testing of the lap joint repairs. As shown in the load transfer calculations in chapter 4, the adhesive has a significant effect in the load transfer of a bonded-riveted joint. Therefore, the use of an adhesive as a faying surface increases the fatigue life considerably. The fracture of the specimens were type 1, except for the specimens with the applied gross stress of 140 MPa, the fractures were type 2. These fractures were due to delamination of the skin and the doubler between the first rivet row and the end of the over-lap. Then the adhesive locally was no longer transferring the load. As a result, the bearing stress in the rivet hole at the first rivet row increases significantly, which lead to type 1 fracture.
The high gross stress and the eccentricity of the loading in the single lap joint specimen, create two major critical conditions at the area just near the end of the overlapping, i.e. the relative displacement between the skin and the doubler, due to the applied load, and the moment caused by the eccentricity of the loading. The relative displacement causes a high shear force on the adhesive, and the moment creates peeling forces, see in Figure 6-17. The combination of these two critical conditions caused the delamination of the bonding after a few thousand cycles with the applied gross load of 140 MPa. This delamination can easily be observed visually by a magnifying glass at low loading frequencies (0.1-0.5 Hz).

![Image of peel forces due to eccentricity in loading]

Figure 6-17, Peel effect due to the eccentricity loading

In the test series, two types of the adhesive AW 106 and EC 9323, resulted in a better fatigue life as compared to the third adhesive EC 2216. Specimens with EC 2216 adhesive showed some type 2 fractures for the applied gross stress of 120 MPa. The adhesive then fails in taking part of the loads during fatigue. Delamination occurred between the primer and the metal surface. Therefore, as mentioned in the specimen manufacturing in chapter 5 of part 1, the surface preparation will have a significant effect on the bonding quality. From Figure 6-18 it can be seen that the adhesive as a faying material has a better fatigue life as compared with the sealant (as the basis specimen).

Again, these results showed that the adhesive contributed significantly in the load transfer of the bonded-riveted joint as presented in chapter 4 of part 1. Furthermore the durability of the bonding is also an important parameter to be considered, when choosing the type of adhesive for a faying surface in the design.

The spew fillet or the over-fill of the adhesive at the edges of the overlap, has a significant effect on the fatigue life. Figure 6-14 shows the spew fillet. Specimens without a spew fillet, due to unintentionally cleaning during the manufacturing, had a lower the fatigue life (about 40 percent) as compared to the specimen with a good spew fillet, as shown in Table 6-12. The spew fillet reduces the effect of the abrupt change at the cross section and the secondary bending at the edge of the overlap [28].

The crack growth period of the specimens with type 1 fracture, was about 2 kc to 10 kc, therefore it sometimes was hard to have an opportunity for visual crack detection. Type 2 fractures had a little longer crack growth period and some crack growths could be recorded.
6.5.2.4 Conclusions on the Effects of the Adhesive

1. Cold bonding adhesive as a faying surface of a lap joint repair increased the fatigue life considerably.

2. EU 9323 cold bonding adhesive as a faying surface resulted in the highest fatigue life.

3. All specimens with the applied gross stresses of 140 MPa failed by fractured of type 2, which was due to the delamination of the bonding between the first rivet row and the end of the overlapping.

4. Specimens with a gross stress below 140 MPa failed with type 1, except for specimens with the EC 2216 adhesive, tested at 120 MPa which failed according to type 2.

5. The type 1 fractures indicate that the bonding adhesive has a favorable effect on fatigue of the bonded-riveted joint, which confirms the calculation of the load transfer in chapter 4. The rivets at the first rivet row carried very little load and it was no longer the fatigue critical area.

![Figure 6-18, Effect of adhesive on fatigue life of Glare lap joint.](image)

6.5.2.5 The Effect of Doubler Materials on Fatigue Strength

Table 6-10 to Table 6-17 show the results of the fatigue testing on the lap joint repairs with various configurations of the patch materials and thickness. A thinner Glare doubler of 90 percent of skin thickness, repaired with a sealant as a faying, resulted in just slightly better fatigue lives as compared to the aluminum doubler (basis...
specimen), as shown in Figure 6-19. At the high stress level of 140 MPa, the thinner Glare doubler gave the same fatigue life as for an aluminum doubler. The thinner Glare doubler gives a little higher bearing stress for the load transfer through the rivet (see load transfer calculation in chapter 4 of part 1) and it also causes a higher gross stress. Thus, the combination of these two conditions results in a significant increase of the bearing stress on the rivet holes of the Glare doubler. As a consequence, it reduced its fatigue life. The type of fracture was type 2 at the doubler side (Glare side). This type of fracture is more damage tolerant because it can be observed by the technicians during inspection. The crack growth period between the first visual crack and failure of the Glare doubler is comparable to the result for the aluminum doubler. Most of the failures were multiple site cracking. Therefore the crack growth period is very short and sometime it is hard to detect it visually.

The thicker Glare 3 4/3 1.95 mm doubler with a sealant as a faying surface has a better fatigue life as compared to the aluminum doubler (basis specimen), as shown in Figure 6-19. The thicker Glare doubler with a somewhat lower elastic modulus, as compared to aluminum, resulted in an equal load transfer at the first and the third rivet rows. The load transfer calculation was presented in chapter 4. The thicker Glare doubler has a lower bearing stress due to its lower gross stress and equal load transfer by the rivets, therefore the type of the fracture was type 2 in the aluminum skin. For this reason the old repair concept with a thicker doubler of one gage or more of the skin being repaired, can be applied for Glare without any consequences in lowering the fatigue life. But, if this concept is applied to an aluminum doubler, then it will reduce the fatigue life considerably as mentioned by Swift [8]. The crack growth period is comparable to the thinner Glare. The longer fatigue life of the thicker Glare sheet, can give a longer inspection threshold, which means fewer man-hours for inspection.

Figure 6-19, Effect of different materials on the fatigue live of a lap joint repair with sealant as a faying surface.
6.5.2.6 Conclusions on the Effect of Doubler Materials

1. A thinner Glare doubler of 90 percent of the skin thickness, repaired with sealant, showed a comparable fatigue life as an aluminum doubler repair, while the failure occurred in the Glare doubler (type 2). This type of failure has more damage tolerance from the inspection standpoint, because this failure can be detected by the technician during inspection.

2. The use of a Glare doubler with an adhesive as a faying surface leads to a considerable improvement of the fatigue life. It is due to the adhesive which reduced the load transfer through the rivet, and thus reduced the bearing stress in the rivet holes.

3. A thicker Glare doubler of two gages more than the skin thickness, repaired with a sealant as a faying surface, shows a favorable fatigue life as compared to the aluminum doubler due to a little lower elastic modulus resulting in equal load transfer at the first and the third rivet rows. Consequently, it reduces the bearing strength in the rivet holes of the Glare doubler.

4. The test results above confirm that the current concept of repair for a doubler which should be one gage thicker than the skin being repaired, can be applied to Glare as a repair material without any consequences for the fatigue life, as well as for the static strength.

6.5.2.7 Effect of the Use of the Former Countersunk Hole on Fatigue Strength

The former countersunk holes in the skin underneath the doubler at the run-out row of a riveted repair with sealant as a faying surface reduced the fatigue life significantly, as shown in Figure 6-20. The Countersunk holes underneath the external doubler causes an increase of the bearing stress in the rivet holes, and thus reduce the fatigue life. By employing the cold bonding adhesive as a faying surface, the fatigue lives was significantly increased because the adhesive as shown in the load transfer calculation in chapter 4 of part 1, reduces the load transfer through the rivets significantly. Thus the bearing stress in the rivet hole will also be reduced.

Table 6-18 and Table 6-19 show the test results of the lap joint repair with the effect of the use of the countersunk hole underneath the doubler at the run-out row of the rivet.

6.5.2.8 Conclusions on the Effect of the Countersunk

1. The former countersunk holes underneath the doubler at the run-out row of the rivet, reduced the fatigue life considerably, about 30 percent (at the gross stress of 100 MPa). Therefore using these holes as end rivets it in repair could be avoided.
2. The use of a cold bonding adhesive on the countersunk skin increased the fatigue life significantly.

![Graph showing the influence of the use of the former countersunk hole underneath the doubler at the run-out row of rivet on fatigue life.]

Figure 6-20, The influence of the use the former countersunk hole underneath the doubler at the run-out row of rivet on fatigue life.

**6.5.2.9 Conclusions of the Lap Joint Repair Tests**

1. The type 1 fracture of the specimens with an adhesive as a faying surface is a consequence of the abrupt change in the cross section. This failure indicates that the adhesive is significant in transferring the load in the bonded-riveted joint. This type of failure is confirmed by the result of the load transfer calculations in chapter 4 of part I. The type 1 fracture results in a significant by higher fatigue life as compared to type 2 fracture. This type of fracture is the goal of using the adhesive as a faying surface.

2. The type 2 fracture of the bonded-riveted specimens is always initiated after a delamination of the adhesive between the first rivet row and the end of the overlap. The adhesive then is no longer transferring the load. In this case, the joint should be considered as a merely riveted joint. Therefore the bonding quality is important in order to achieve a good quality of repair.

3. The use of Glare doubler thicker than the skin being repaired combined with a sealant as a faying surface on the aluminum skin increased the fatigue life significantly. The use of Glare as a repair material while still adopting the current repair for an aluminum aircraft, can increase the repair threshold as well as the damage tolerance of the repair. The test results show that Glare as a repair material for the aluminum fuselage can be accepted.
6.5.3 Fatigue Experiments of Curved Specimens Under Uniaxial Pressurization Loading

The purpose of testing curved specimens under cyclic internal pressurization, is to determine the reduction factor $CF$ of the fatigue life due to the curvature effect in the lap joint specimen. This reduction factor will be used to correct the fatigue life of the flat specimens in determining the fatigue life of the actual repair. The bulging effect of the crack edges due to the internal air pressure (see Figure 6-21) and the bending moment due to the eccentricities on the single lap joint specimen, can have a significant effect on the fatigue life as well as on the crack growth [34]. The bulging effect will increase the crack growth rate. Our specimens were installed on the curved test set-up, while it was initially flat, but in practice it should be rolled first before the installation. Therefore, the specimen will experience some slight bending (see Figure 6-23), so the outer skin, called doubler in this experiment will experience more bending stress. Figure 6-22 shows the schematic drawing of the curved test set-up.

![Figure 6-21, The bulging effect on crack due to the pressurization](image)

Specimens width 500 mm and length 1080 mm, are installed in the curved test set-up, as shown in Figure 6-22. The ends in the longitudinal directions were clamped. The two ends in circumferential direction were free to move in a radial direction. Therefore, the specimen was loaded uniaxially in a circumferential direction. The radius of this test set-up is about 2000 mm. The pressurization ($\pm 0.8$ Bar) induced a maximum circumferential stress in the specimens of 110 - 115 MPa. The stress ratio $R$ was about 0.05 - 0.10. The frequency was 0.005 - 0.008 Hz. The specimens were manufactured in the same manner as for flat specimens. The number of the specimen configurations is two, which are Clad Al 2024 T3 1.6 mm with Clad Al 2024 T3 1.6 mm joint, and Glare 3 3/2 1.45 mm with Clad Al 2024 T3 1.6 mm joint, while a PR 1422 B2 sealant was used as a faying surface for both types of the specimen configurations. The number of each configuration to be tested was two (see Table 6-7). The test set-up is shown in Figure 6-24.
6.5.3.1 Test Result and Discussion of Curved Specimen With Pressurization Load

The test results of the fatigue life of the curved specimens with the pressurization load on Clad Al 2024 T3 lap joints were 102 kc and 81 kc respectively (see Table 6-21). The flat specimens with a similar nominal gross stress, failed at 262 kc, so the life reduction due to the curvature was about 2.9 times. This reduction is in agreement with the results found by Müller [11]. The crack initiation in the two specimens detected by NDT (Eddy current) occurred after 36 kc and 35 kc, respectively. The cracks in both specimens were hidden for visual inspection. As soon as the cracks could be seen from the outside, the crack growth period was very short as a result of the bulging effect. The failures in both specimens occurred at the first rivet row, which is the top row of the rivets (doubler), see Figure 6-23.

![Figure 6-22, The schematic drawing of the curved test with internal pressure.](image)

![Figure 6-23, Deformation induced by the bending.](image)

The use of the rubbers to seal the free edges of the specimen to prevent the air from leaking, resulted in a favorable effect on the stress at the edges of the specimen. This effect is like the frame effect which reduces the hoop stress at the edges of the specimen. This reduction appears when the part moving out-ward in the radial direction due to the pressurization, then the rubber restrained this radial movement. Therefore, the elongation in circumferential direction is smaller as compared to the
middle of the specimen. As a consequence, the hoop stress at that area are also smaller, Figure 6-25 shows the rubber's installation. These reductions were measured along the longitudinal direction by using strain gages, the results are shown in Figure 6-26. The stress became partially uniform after 8 cm away from the edges. Observations of the cross section on the fracture surface confirmed that cracks were never initiated from the edges of the specimen due to its lower stress.

Table 6-7, The configurations of lap joint repair, skin clad AL 2024 T3 1.6 mm

<table>
<thead>
<tr>
<th>Doubler materials and fayings</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Type I</strong></td>
</tr>
<tr>
<td>Alclad 2024 T3 1.6 mm with PR 1422 B2 (sealant)</td>
</tr>
<tr>
<td><strong>Type II</strong></td>
</tr>
<tr>
<td>Glare 3 3/2 1.45 mm with PR 1422 B2 (sealant)</td>
</tr>
</tbody>
</table>

The fatigue lives of the two specimens with the configuration of Glare 3 as doubler material and Clad Al 2024 T3 as skin material were 191 kc and 184 kc, respectively, where the specimen with higher fatigue life failed at the clamping of the aluminum skin due to fretting. The other specimen which has a lower fatigue life failed at the first rivet row on the Glare side. Flat specimens with a similar nominal gross stress failed at about 268 kc, so the reduction of the fatigue life from the flat specimens to curved specimen was about 1.4 times. This result is a little lower than the result of the Glare 3 lap joints tested by Müller [11]. The reason is that the thinner Glare doubler sheet in combination with the aluminum skin caused a little higher load transfer on the first rivet row at the Glare side, as shown in the load transfer calculation in chapter 4 of part 1. The visual cracks appeared after 135 kc, and the crack growth was very slow, as shown in Table 6-21 (about 50 kc). This was due to the effect of the intact fibers in the cracked area which are still bridging the cracks. From the damage tolerance point of view, this slow crack growth can give a longer inspection threshold and increases the safety significantly, as compared to aluminum.

Figure 6-24, Curve test set-up
6.5.3.2 Conclusions of the Curved Test

1. Specimens tested in the curved test set up show a significant fatigue life reduction as compared to the flat specimens, as shown in Figure 6-27.

2. The reduction of the fatigue life due to the curvature effect of Al 2024 T3 to Al 2024 T3 lap joint specimens is about 2.9 times, while for the combination of Glare 3/2.3 to Al 2024 T3 it is about 1.4 times. Again, the fatigue tests demonstrate that Glare as a doubler material are a superior in fatigue life as compared to the aluminum.

3. The crack edge bulging in the aluminum specimens has a significant effect on the crack growth, but this effect on Glare is much less critical.

4. Glare repair on the aluminum skin have a favorable effect from the damage tolerance point of view, i.e. a longer fatigue life and a longer crack growth.

Figure 6-25, The rubbers sealing installation.
"FATIGUE STRENGTH OF A REPAIRED FUSELAGE STRUCTURE"

Figure 6-26, Strain-gages measurements of the hoop stresses in the longitudinal direction of the curved test set-up with internal pressure.

Figure 6-27, Fatigue life comparison of uniaxial loaded flat and curved specimens.

6.6 The Fatigue Basic and Interval basic Determination

In order to obtain the actual fatigue life of a repair, the fatigue test result found in the flat specimen with uniaxial loading shall be corrected with several factors(e. g. curved effect, scatter factor, etc.) which has an effect on the fatigue life of the actual structure. The corrected fatigue lives will be needed in order to determine the
inspection threshold and the repeat inspection interval. The inspection threshold is the period after which the first inspection of the repair must be carried out. The repeat inspection interval is the period when the following inspection must be carried out to ensure the continuation of its airworthiness.

The corrected inspection threshold is called fatigue basic and the corrected repeat inspection interval is called interval basic. The correction factors are an effect in the fatigue basic \( N_{basic} \) and the interval basic \( I_{basic} \) which are 1) the curved effect, 2) the applied gross stress of the specimen (according to Müller [11] is 1.2 higher than nominal skin stress) and 3) the scatter factor \( SF \). The use of these three factors will result in a conservative fatigue life than it is found in the coupon testing done in the laboratory. Therefore in this section, there are many assumptions which are to be made in order to correct those results as close as possible to the actual fatigue life of a repair. These assumptions are also taken from the MSG 3, aging program reports and also taken from references [3, 9, 16, 17, and 35] which are used as guidelines. In accordance with these assumptions, the fatigue basic and the interval basic of a repair can be derived from the equations 6-3, 6-4 and 6-5 as follows:

Fatigue basic \( N_{basic} \):

\[
N_{basic} \leq T \cdot CF
\]

\[
N_{basic} \leq \frac{calculated\_life}{5}
\]

\[
N_{basic} \leq \frac{ERL}{2}
\]

Repeat inspection interval \( I_{basic} \):

\[
I_{basic\_Ndt} = R_{Ndt} \cdot CF
\]

\[
I_{basic\_visual} = R_{visual} \cdot CF
\]

Where:

- \( T \) is inspection threshold from the fatigue test result (using equation 6-3),
- \( R_{NDT} \) and \( R_{visual} \) are the repeat interval for NDT and visual inspection (see also equations 6-4 and 6-5),
- \( CF \) is curvature correction factor,
- \( calculated\_life \) is the design fatigue life (in flight cycle) of the aircraft (e.g. Airbus
A320 the calculated life ~ 90,000 flight cycles), and
• ERL is the Economic Repair Life (e.g. Airbus A 320, ERL ~ 48,000 flight cycles).
• From the three equations 6-8, 6-9, and 6-10, the lowest value should be chosen as
the fatigue basic.

The correction factor for the panel curvature effect for Clad Al 2024 T3 1.6 mm joint
as concluded by Müller [11] should be about 1/5 for flat specimens without any faying
surface employed on the mating surfaces. From the present experiments, this factor
was found to be about 1/2.9. The difference is mainly due to the application of the
sealant as a faying surface on both flat and curved specimens of our experiments. In
order to have a conservative result, the curvature reduction factor CF of aluminum
repair with both faying surface adhesive and sealant on the countersunk skin or on
the non countersunk skin is chosen to be 1/2.9. While the correction factor for the
combination joint of Glare and aluminum taken from this experiment is 1/1.4. This
factor is used for all Glare repairs with both types of faying surfaces, adhesives or
sealant.

The scatter factor SF for the calculation of the repair threshold for aluminum or Glare
repairs with a sealant or an adhesive as the faying surface is chosen to be 3 [9]. The
scatter factor for the repeat inspection for aluminum with both (countersunk and non
countersunk skins) is chosen as 2 and for Glare is 3 [9], regardless it uses a sealant or
an adhesive as a faying surface. These factors for Glare can be corrected to increase or
to reduce which depends on the data available concerning for this issue.

The fatigue basic and the interval basic found in this experiment are not yet corrected
for the influences of possible Environmental Damage and the Accidental Damage.
Furthermore in determining the actual repair for aircraft, both factors must be
considered, then the repair threshold must be chosen from the three ratings (Fatigue
Damage, Environmental Damage and Accidental Damage) with a rule of Whichever
 Comes First. It means that the lowest value of those three ratings be chosen as the
inspection threshold and must be done in the first opportunity of the maintenance
program. In our experiments only the fatigue damage has been considered, while the
determination of the Environmental Damage rating and the Accidental Damage rating
shall refer to MSG 3 logic diagram, provided by the aircraft manufacturer for a
specific type of aircraft.

The nominal gross stress of the flat specimen used to determine the fatigue basic and
the interval basic is 120 MPa or 1.2 higher than the nominal skin stress of MD 11.
While the nominal stress of MD 11 is according to Niu [2] is 103 MPa. Table 6-20
shows the fatigue basic and the interval basic of aluminum skin with a thickness of
1.6 mm. In this table shows that the aluminum specimen with countersunk in the skin
(G-series) results a little higher inspection threshold at stress level above 120 MPa.
Therefore, these results need for further investigation.
6.7 References


[34] Chen, D., Bulging of Fatigue Cracks in a Pressurized Aircraft Fuselage, Proefschrift, January 1991, Structure and Material Laboratory of Faculty of Aerospace and Engineering of Delft University of Technology.

Table 6-8, Fatigue test result of external patch repair.

<table>
<thead>
<tr>
<th>Patch</th>
<th>Last visual inspc, (k c)</th>
<th>Visual crack, $N_{visual}$ (k c)</th>
<th>Fatigue, $N_{critical}$ (k c)</th>
<th>Failure type</th>
<th>Crack growth, (k c)</th>
<th>Crack initiation &amp; position</th>
<th>Remark</th>
</tr>
</thead>
<tbody>
<tr>
<td>spec # 1</td>
<td>I</td>
<td>245</td>
<td>248</td>
<td>1</td>
<td>3</td>
<td>Multiple 1, 3 &amp; 7</td>
<td></td>
</tr>
<tr>
<td>Al 2024, sealant</td>
<td>II</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>spec # 2</td>
<td>I</td>
<td>484</td>
<td>513</td>
<td>1</td>
<td>30</td>
<td>Single 1</td>
<td></td>
</tr>
<tr>
<td>Al 2024, AW 106</td>
<td>II</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>spec # 3</td>
<td>I</td>
<td>343</td>
<td>426</td>
<td>1</td>
<td>80</td>
<td>Single 1</td>
<td></td>
</tr>
<tr>
<td>Glare 3, sealant</td>
<td>II</td>
<td>410</td>
<td>424</td>
<td>1</td>
<td>13</td>
<td>Multiple 1, 2 &amp; 3</td>
<td></td>
</tr>
<tr>
<td>spec # 4</td>
<td>I</td>
<td>493</td>
<td>628</td>
<td>1</td>
<td>135</td>
<td>Multiple 2 &amp; 3</td>
<td></td>
</tr>
<tr>
<td>Glare 3, AW 106</td>
<td>I</td>
<td>385</td>
<td>390</td>
<td>1</td>
<td></td>
<td>Multiple 1, 2, 3 &amp; 7</td>
<td></td>
</tr>
<tr>
<td>spec # 5</td>
<td>I</td>
<td>845</td>
<td>1.847</td>
<td>2</td>
<td></td>
<td>Single 1</td>
<td>Residual test</td>
</tr>
<tr>
<td>Al 2024, sealant, Ctr</td>
<td>II</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>spec # 6</td>
<td>I</td>
<td>825</td>
<td>832</td>
<td>3</td>
<td>7</td>
<td>Multiple 1 &amp; 2</td>
<td>Delamination</td>
</tr>
<tr>
<td>Al 2024, AW 106, Ctr</td>
<td>II</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>spec # 7</td>
<td>I</td>
<td>148</td>
<td>185</td>
<td>1</td>
<td>37</td>
<td>Single 2</td>
<td></td>
</tr>
<tr>
<td>Glare 3, sealant, Ctr</td>
<td>II</td>
<td>149</td>
<td>185</td>
<td>1</td>
<td>36</td>
<td>Single 2</td>
<td></td>
</tr>
<tr>
<td>spec # 8</td>
<td>I</td>
<td>183</td>
<td>220</td>
<td>1</td>
<td>37</td>
<td>Single 1</td>
<td>Delamination</td>
</tr>
<tr>
<td>Glare 3, AW 106, Ctr</td>
<td>II</td>
<td>206</td>
<td>234</td>
<td>1</td>
<td>28</td>
<td>Single 1</td>
<td></td>
</tr>
<tr>
<td>spec # 9</td>
<td>I</td>
<td>217</td>
<td>313</td>
<td>1</td>
<td>96</td>
<td>Single 2</td>
<td></td>
</tr>
<tr>
<td>None faying</td>
<td>II</td>
<td>180</td>
<td>232</td>
<td>1</td>
<td>52</td>
<td>Multiple 1 &amp; 8</td>
<td></td>
</tr>
<tr>
<td>spec # 10</td>
<td>I</td>
<td>384</td>
<td>434</td>
<td>1</td>
<td>50</td>
<td>Single 1</td>
<td></td>
</tr>
<tr>
<td>None faying</td>
<td>II</td>
<td>307</td>
<td>347</td>
<td>1</td>
<td>40</td>
<td>Single 3</td>
<td></td>
</tr>
<tr>
<td>None faying</td>
<td>I</td>
<td>137</td>
<td>170</td>
<td>1</td>
<td>32</td>
<td>Multiple 1, 6 &amp; 8</td>
<td></td>
</tr>
<tr>
<td>None faying</td>
<td>II</td>
<td>206</td>
<td>232</td>
<td>1</td>
<td>26</td>
<td>Multiple 2 &amp; 5</td>
<td></td>
</tr>
<tr>
<td>None faying</td>
<td>I</td>
<td>243</td>
<td>253</td>
<td>1</td>
<td></td>
<td>Multiple 2, 3 &amp; 8</td>
<td></td>
</tr>
<tr>
<td>None faying</td>
<td>II</td>
<td>433</td>
<td>460</td>
<td>1</td>
<td>270</td>
<td>Multiple 7 &amp; 8</td>
<td></td>
</tr>
</tbody>
</table>

Note: See the rivet numbering in figure 4-3 of part 1, and the fatigue in K c is kilo cycles.
Table 6-9, Inspection interval of patch repair.

<table>
<thead>
<tr>
<th>Patch</th>
<th>Fatigue, $N_{critical}$ (kc)</th>
<th>Visible crack growth, (kc)</th>
<th>Visual inspe threshold $T_{visual}$, (kc) with Scf = 2</th>
<th>Repeat inspection $R_{visual}$, (kc) with Scf = 2</th>
</tr>
</thead>
<tbody>
<tr>
<td>spc # 1, Al 2024, sealant</td>
<td>381</td>
<td>16</td>
<td>182</td>
<td>8</td>
</tr>
<tr>
<td>spc # 2, Al 2024, AW 106</td>
<td>424</td>
<td>47</td>
<td>188</td>
<td>23</td>
</tr>
<tr>
<td>spc # 3, Glare 3, sealant</td>
<td>509</td>
<td>135</td>
<td>247</td>
<td>67</td>
</tr>
<tr>
<td>spc # 4, Glare 3, AW 106</td>
<td>1,339</td>
<td>8</td>
<td>417</td>
<td>4</td>
</tr>
<tr>
<td>spc # 5, Al 2024, sealant, Ctr</td>
<td>185</td>
<td>36</td>
<td>74</td>
<td>18</td>
</tr>
<tr>
<td>spc # 6, Al 2024, AW 106, Ctr</td>
<td>227</td>
<td>32</td>
<td>97</td>
<td>16</td>
</tr>
<tr>
<td>spc # 7, Glare 3, sealant, Ctr</td>
<td>273</td>
<td>74</td>
<td>99</td>
<td>37</td>
</tr>
<tr>
<td>spc # 8, Glare 3, AW 106, Ctr</td>
<td>391</td>
<td>45</td>
<td>173</td>
<td>23</td>
</tr>
<tr>
<td>spc # 9, None faying</td>
<td>201</td>
<td>29</td>
<td>81</td>
<td>15</td>
</tr>
<tr>
<td>spc # 10, None faying</td>
<td>356</td>
<td>27</td>
<td>216</td>
<td>13</td>
</tr>
</tbody>
</table>
Table 6-10, Fatigue performance of lap joint repair, Maximum gross stress = 140 MPa, R = 0.05

<table>
<thead>
<tr>
<th>Doubler material</th>
<th>Sealant PR-1422B-2</th>
<th>Sealant PR-1422B-2</th>
<th>Sealant PR-1422B-2</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Acclad 2024-T3 (t = 1.60 mm)</td>
<td>GLARE 3-3/2-0.3 (t = 1.40 mm)</td>
<td>GLARE 3-4/3-0.3 (t = 1.95 mm)</td>
</tr>
<tr>
<td>Fay material</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Specimen No.</td>
<td>A-1</td>
<td>A-2</td>
<td>B-1</td>
</tr>
<tr>
<td>Fail due to fatigue?</td>
<td>yes</td>
<td>yes</td>
<td>yes</td>
</tr>
<tr>
<td>Fatigue failure type</td>
<td>doubler fracture (Acclad)</td>
<td>doubler fracture (GLARE)</td>
<td>doubler fracture (GLARE)</td>
</tr>
<tr>
<td>Fatigue failure location</td>
<td>along 1st rivet row (upper)</td>
<td>along 1st rivet row (upper)</td>
<td>along 1st rivet row (upper)</td>
</tr>
<tr>
<td>Crack initiation by NDT (cycles)</td>
<td>40,000</td>
<td>40,000</td>
<td>40,000</td>
</tr>
<tr>
<td>Visible cracks (cycles)</td>
<td>50,000</td>
<td>45,000</td>
<td>45,000</td>
</tr>
<tr>
<td>Fatigue life (cycles)</td>
<td>77,680</td>
<td>68,880</td>
<td>66,409</td>
</tr>
<tr>
<td>Residual strength (KN)</td>
<td>-</td>
<td>-</td>
<td>-</td>
</tr>
<tr>
<td>Static failure type</td>
<td>-</td>
<td>-</td>
<td>-</td>
</tr>
<tr>
<td>Static failure location</td>
<td>-</td>
<td>-</td>
<td>-</td>
</tr>
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</table>
Table 6-11, Fatigue performance of lap joint repair, Maximum gross stress = 140 MPa, R = 0.05

<table>
<thead>
<tr>
<th>Doubler material</th>
<th>GLARE 3-3/2-0.3 (t = 1.40 mm)</th>
<th>GLARE 3-3/2-0.3 (t = 1.40 mm)</th>
<th>GLARE 3-3/2-0.3 (t = 1.40 mm)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Fay material</td>
<td>Adhesive AW-106</td>
<td>Adhesive EC-2216 B/A</td>
<td>Adhesive EU-9323 B/A</td>
</tr>
<tr>
<td>Specimen No.</td>
<td>D-1</td>
<td>E-1</td>
<td>F-1</td>
</tr>
<tr>
<td></td>
<td>D-2</td>
<td>E-2</td>
<td>F-2</td>
</tr>
<tr>
<td>Fail due to fatigue?</td>
<td>yes</td>
<td>yes</td>
<td>yes</td>
</tr>
<tr>
<td>Fatigue failure type</td>
<td>skin fracture</td>
<td>doubler fracture</td>
<td>skin fracture</td>
</tr>
<tr>
<td>Fatigue failure location</td>
<td>edge of lap joint</td>
<td>along 1st rivet row</td>
<td>edge of joint</td>
</tr>
<tr>
<td>Crack initiation by NDT (cycles)</td>
<td>100,000</td>
<td>100,000</td>
<td>160,000</td>
</tr>
<tr>
<td>Visible cracks (cycles)</td>
<td>240,000</td>
<td>220,000</td>
<td>350,000</td>
</tr>
<tr>
<td>Fatigue life (cycles)</td>
<td>254,142</td>
<td>250,000</td>
<td>467,927</td>
</tr>
<tr>
<td>Residual strength (KN)</td>
<td>-</td>
<td>-</td>
<td>-</td>
</tr>
<tr>
<td>Static failure type</td>
<td>-</td>
<td>-</td>
<td>-</td>
</tr>
<tr>
<td>Static failure location</td>
<td>-</td>
<td>-</td>
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</tr>
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</table>
Table 6-12, Fatigue performance of lap joint repair, Maximum gross stress = 120 MPa, R = 0.05

<table>
<thead>
<tr>
<th>Doubler material</th>
<th>Alclad 2024-T3 (t = 1.60 mm)</th>
<th>GLARE 3-3/2-0.3 (t = 1.40 mm)</th>
<th>GLARE 3-4/3-0.3 (t = 1.95 mm)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Fay material</td>
<td>Sealant PR-1422B-2</td>
<td>Sealant PR-1422B-2</td>
<td>Sealant PR-1422B-2</td>
</tr>
<tr>
<td>Specimen No.</td>
<td>A-3</td>
<td>B-3</td>
<td>B-4</td>
</tr>
<tr>
<td>Fail due to fatigue?</td>
<td>yes</td>
<td>yes</td>
<td>yes</td>
</tr>
<tr>
<td>Fatigue failure type</td>
<td>doubler fracture (Alclad)</td>
<td>doubler fracture (GLARE)</td>
<td>skin fracture</td>
</tr>
<tr>
<td>Fatigue failure location</td>
<td>along 1st rivet row (upper)</td>
<td>along 1st rivet row (upper)</td>
<td>along 1st rivet row (lower)</td>
</tr>
<tr>
<td>Crack initiation by NDT (cycles)</td>
<td>500,000</td>
<td>200,000</td>
<td>190,000</td>
</tr>
<tr>
<td>Visible cracks (cycles)</td>
<td>700,000</td>
<td>240,000</td>
<td>230,000</td>
</tr>
<tr>
<td>Fatigue life (cycles)</td>
<td>731,571</td>
<td>261,925</td>
<td>243,131</td>
</tr>
<tr>
<td>Residual strength (KN)</td>
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<td>-</td>
<td>-</td>
</tr>
<tr>
<td>Static failure type</td>
<td>-</td>
<td>-</td>
<td>-</td>
</tr>
<tr>
<td>Static failure location</td>
<td>-</td>
<td>-</td>
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</table>

* The specimen with no spew fillet.
Table 6-13, Fatigue performance of lap joint repair, Maximum gross stress = 120 MPa, R = 0.05

<table>
<thead>
<tr>
<th>Doubler material</th>
<th>GLARE 3-3/2-0.3 (t = 1.40 mm)</th>
<th>GLARE 3-3/2-0.3 (t = 1.40 mm)</th>
<th>GLARE 3-3/2-0.3 (t = 1.40 mm)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Fay material</td>
<td>Adhesive AW-106</td>
<td>Adhesive EC-2216 B/A</td>
<td>Adhesive EU-9323 B/A</td>
</tr>
<tr>
<td>Specimen No.</td>
<td>D-3</td>
<td>E-3</td>
<td>F-3</td>
</tr>
<tr>
<td>Fail due to fatigue?</td>
<td>yes</td>
<td>yes</td>
<td>yes</td>
</tr>
<tr>
<td>Fatigue failure type</td>
<td>skin fracture</td>
<td>doubler fracture (GLARE)</td>
<td>skin fracture</td>
</tr>
<tr>
<td>Fatigue failure location</td>
<td>edge of joint + 1st rivet row (lower) tension</td>
<td>along the upper row of rivet holes</td>
<td>edge of joint</td>
</tr>
<tr>
<td>Crack initiation by NDT (cycles)</td>
<td>300,000</td>
<td>340,000</td>
<td>350,000</td>
</tr>
<tr>
<td>Visible cracks (cycles)</td>
<td>500,000</td>
<td>430,000</td>
<td>420,000</td>
</tr>
<tr>
<td>Fatigue life (cycles)</td>
<td>576,730</td>
<td>472,531</td>
<td>460,181</td>
</tr>
<tr>
<td>Residual strength (KN)</td>
<td>-</td>
<td>-</td>
<td>-</td>
</tr>
<tr>
<td>Static failure type</td>
<td>-</td>
<td>-</td>
<td>-</td>
</tr>
<tr>
<td>Static failure location</td>
<td>-</td>
<td>-</td>
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</tr>
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</table>
Table 6-14, Fatigue performance of lap joint repair, Maximum gross stress = 100 MPa, R = 0.05

<table>
<thead>
<tr>
<th>Doubler material</th>
<th>Alc clad 2024-T3 (t = 1.60 mm)</th>
<th>GLARE 3-3/2-0.3 (t = 1.40 mm)</th>
<th>GLARE 3-4/3-0.3 (t = 1.95 mm)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Fay material</td>
<td>Sealant PR-1422B-2</td>
<td>Sealant PR-1422B-2</td>
<td>Sealant PR-1422B-2</td>
</tr>
<tr>
<td>Specimen No.</td>
<td>A-5</td>
<td>B-5</td>
<td>C-5</td>
</tr>
<tr>
<td>Fail due to fatigue ?</td>
<td>no</td>
<td>partial failure</td>
<td>yes</td>
</tr>
<tr>
<td>Fatigue failure type</td>
<td>doubler fracture (data is useless)</td>
<td>MSD on 2 rivet locations (doubler)</td>
<td>doubler fracture (GLARE)</td>
</tr>
<tr>
<td>Fatigue failure location</td>
<td>along 1st rivet row (lower)</td>
<td>rivet no. 1 and 2 left (upper)</td>
<td>along upper row of rivet holes</td>
</tr>
<tr>
<td>Crack initiation by NDT (cycles)</td>
<td>1,400,000</td>
<td>--</td>
<td>600,000</td>
</tr>
<tr>
<td>Visible cracks (cycles)</td>
<td>--</td>
<td>--</td>
<td>1,112,000</td>
</tr>
<tr>
<td>Fatigue life (cycles)</td>
<td>1,640,000</td>
<td>607,827</td>
<td>1,600,000</td>
</tr>
<tr>
<td>Residual strength (KN)</td>
<td>71.483</td>
<td>--</td>
<td>35.482</td>
</tr>
<tr>
<td>Static failure type</td>
<td>rivet shear</td>
<td>--</td>
<td>doubler fracture (GLARE)</td>
</tr>
<tr>
<td>Static failure location</td>
<td>--</td>
<td>--</td>
<td>along 1st rivet row (upper)</td>
</tr>
</tbody>
</table>
Table 6-15, Fatigue performance of lap joint repair, Maximum gross stress = 100 MPa, R = 0.05

<table>
<thead>
<tr>
<th>Doubler material</th>
<th>GLARE 3-3/2-0.3 (t = 1.40 mm)</th>
<th>GLARE 3-3/2-0.3 (t = 1.40 mm)</th>
<th>GLARE 3-3/2-0.3 (t = 1.40 mm)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Fay material</td>
<td>Adhesive AW-106</td>
<td>Adhesive EC-2216 B/A</td>
<td>Adhesive EU-9323 B/A</td>
</tr>
<tr>
<td>Specimen No.</td>
<td>D-5</td>
<td>E-5</td>
<td>F-5</td>
</tr>
<tr>
<td>Fail due to fatigue?</td>
<td>yes</td>
<td>yes</td>
<td>no</td>
</tr>
<tr>
<td>Fatigue failure type</td>
<td>skin fracture</td>
<td>doubler fracture (GLARE)</td>
<td>skin fracture</td>
</tr>
<tr>
<td>Fatigue failure location</td>
<td>edge of clamp</td>
<td>along 1st rivet row (upper)</td>
<td>edge of joint</td>
</tr>
<tr>
<td>Crack initiation by NDT (cycles)</td>
<td>1,060,000</td>
<td>1,200,000</td>
<td>1,200,000</td>
</tr>
<tr>
<td>Visible cracks (cycles)</td>
<td>1,458,700</td>
<td>1,350,000</td>
<td>1,200,000</td>
</tr>
<tr>
<td>Fatigue life (cycles)</td>
<td>1,469,300</td>
<td>1,700,000</td>
<td>1,600,000</td>
</tr>
<tr>
<td>Residual strength (KN)</td>
<td>-</td>
<td>74.735</td>
<td>59.595</td>
</tr>
<tr>
<td>Static failure type</td>
<td>-</td>
<td>doubler fracture (GLARE)</td>
<td>skin fracture</td>
</tr>
<tr>
<td>Static failure location</td>
<td>-</td>
<td>along 1st rivet row (upper)</td>
<td>at the clamp</td>
</tr>
</tbody>
</table>

Part 1
Table 6-16. Fatigue performance of lap joint repair, Maximum gross stress = 80 MPa, R = 0.05

<table>
<thead>
<tr>
<th>Doubler material</th>
<th>Aleclad 2024-T3 (t = 1.60 mm)</th>
<th>GLARE 3-3/2-0.3 (t = 1.40 mm)</th>
<th>GLARE 3-4/3-0.3 (t = 1.95 mm)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Fay material</td>
<td>Sealant PR-1422B-2</td>
<td>Sealant PR-1422B-2</td>
<td>Sealant PR-1422B-2</td>
</tr>
<tr>
<td>Specimen No.</td>
<td>A-7</td>
<td>B-7</td>
<td>C-7</td>
</tr>
<tr>
<td>Fail due to fatigue?</td>
<td>no</td>
<td>no</td>
<td>no</td>
</tr>
<tr>
<td>Fatigue failure type</td>
<td>-</td>
<td>-</td>
<td>-</td>
</tr>
<tr>
<td>Fatigue failure location</td>
<td>-</td>
<td>-</td>
<td>-</td>
</tr>
<tr>
<td>Crack initiation by NDT (cycles)</td>
<td>--</td>
<td>1,200,000</td>
<td>1,400,000</td>
</tr>
<tr>
<td>Visible cracks (cycles)</td>
<td>--</td>
<td>--</td>
<td>--</td>
</tr>
<tr>
<td>Fatigue life (cycles)</td>
<td>1,905,378</td>
<td>1,760,000</td>
<td>1,600,000</td>
</tr>
<tr>
<td>Residual strength (KN)</td>
<td>74,508</td>
<td>71,538</td>
<td>67,572</td>
</tr>
<tr>
<td>Static failure type</td>
<td>rivet shear</td>
<td>rivet shear</td>
<td>skin fracture</td>
</tr>
<tr>
<td>Static failure location</td>
<td>-</td>
<td>-</td>
<td>along 1st rivet row (lower)</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>rivet shear</td>
</tr>
</tbody>
</table>
Table 6-17, Fatigue performance of lap joint repair, Maximum gross stress = 80 MPa, R = 0.05

<table>
<thead>
<tr>
<th>Doubler material</th>
<th>GLARE 3-3/2-0.3 (t = 1.40 mm)</th>
<th>GLARE 3-3/2-0.3 (t = 1.40 mm)</th>
<th>GLARE 3-3/2-0.3 (t = 1.40 mm)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Fay material</td>
<td>Adhesive AW-106</td>
<td>Adhesive EC-2216 B/A</td>
<td>Adhesive EU-9323 B/A</td>
</tr>
<tr>
<td>Specimen No.</td>
<td>D-7</td>
<td>E-7</td>
<td>F-7</td>
</tr>
<tr>
<td></td>
<td>D-8</td>
<td>E-8</td>
<td>F-8</td>
</tr>
<tr>
<td>Fail due to fatigue?</td>
<td>no</td>
<td>no</td>
<td>no</td>
</tr>
<tr>
<td>Fatigue failure type</td>
<td>-</td>
<td>-</td>
<td>-</td>
</tr>
<tr>
<td>Fatigue failure location</td>
<td>-</td>
<td>-</td>
<td>-</td>
</tr>
<tr>
<td>Crack initiation by NDT (cycles)</td>
<td>300,000</td>
<td>1,000,000</td>
<td>1,500,000</td>
</tr>
<tr>
<td>Visible cracks (cycles)</td>
<td>577,000</td>
<td>--</td>
<td>--</td>
</tr>
<tr>
<td>Fatigue life (cycles)</td>
<td>2,270,016</td>
<td>1,612,912</td>
<td>1,860,000</td>
</tr>
<tr>
<td>Residual strength (KN)</td>
<td>58.016</td>
<td>59.816</td>
<td>73.578</td>
</tr>
<tr>
<td>Static failure type</td>
<td>doubler failure (GLARE)</td>
<td>doubler failure (GLARE)</td>
<td>doubler failure (GLARE)</td>
</tr>
<tr>
<td>Static failure location</td>
<td>at the clamp</td>
<td>at the clamp</td>
<td>along 1st rivet row (upper)</td>
</tr>
</tbody>
</table>

Part I
Table 6-18, Fatigue performance of lap joint repair with countersunk flaws, R = 0.05

<table>
<thead>
<tr>
<th>Maximum Gross tens (R = 0.05)</th>
<th>140 MPa</th>
<th>120 MPa</th>
</tr>
</thead>
<tbody>
<tr>
<td>Doubler material</td>
<td>Alclad 2024-T3</td>
<td>Alclad 2024-T3</td>
</tr>
<tr>
<td>(t = 1.60 mm)</td>
<td>(t = 1.60 mm)</td>
<td>(t = 1.60 mm)</td>
</tr>
<tr>
<td>Sealant</td>
<td>PR-1422B-2</td>
<td>Adhesive</td>
</tr>
<tr>
<td>PR-1422B-2</td>
<td>EU-9323 B/A</td>
<td>EU-9323 B/A</td>
</tr>
<tr>
<td>Specimen No.</td>
<td>G-1</td>
<td>G-2</td>
</tr>
<tr>
<td>Fail due to fatigue</td>
<td>yes</td>
<td>--</td>
</tr>
<tr>
<td>Fatigue failure type</td>
<td>doubler fract. (Alclad)</td>
<td>--</td>
</tr>
<tr>
<td>Fatigue failure location</td>
<td>along 1st rivet row (upper)</td>
<td>--</td>
</tr>
<tr>
<td>Crack initiation by NDT (cycles)</td>
<td>120,000</td>
<td>--</td>
</tr>
<tr>
<td>Visible cracks (cycles)</td>
<td>140,000</td>
<td>--</td>
</tr>
<tr>
<td>Fatigue life (cycles)</td>
<td>169,091</td>
<td>--</td>
</tr>
<tr>
<td>Residual strength (KN)</td>
<td>--</td>
<td>--</td>
</tr>
<tr>
<td>Static failure type</td>
<td>accidental compression</td>
<td>--</td>
</tr>
<tr>
<td>Static failure location</td>
<td>--</td>
<td>--</td>
</tr>
</tbody>
</table>
Table 6-19, Fatigue performance of lap joint repair with countersunk flaws, $R = 0.05$

<table>
<thead>
<tr>
<th>Maximum Gross Stress ($R = 0.05$)</th>
<th>100 MPa</th>
<th>80 MPa</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Doubler material</strong></td>
<td>Alclad 2024-T3 (t = 1.60 mm)</td>
<td>Alclad 2024-T3 (t = 1.60 mm)</td>
</tr>
<tr>
<td><strong>Fay material</strong></td>
<td>Sealant PR-1422B-2</td>
<td>Adhesive EU-9323 B/A</td>
</tr>
<tr>
<td><strong>Specimen No.</strong></td>
<td>G-5</td>
<td>G-6</td>
</tr>
<tr>
<td><strong>Fail due to fatigue</strong></td>
<td>yes</td>
<td>yes</td>
</tr>
<tr>
<td><strong>Fatigue failure type</strong></td>
<td>doubler fract. (Alclad)</td>
<td>doubler fract. (Alclad)</td>
</tr>
<tr>
<td><strong>Fatigue failure location</strong></td>
<td>along 1st rivet row (upper)</td>
<td>at the clamp (upper)</td>
</tr>
<tr>
<td><strong>Crack initiation by NDT (cycles)</strong></td>
<td>290,000</td>
<td>610,000</td>
</tr>
<tr>
<td><strong>Visible cracks (cycles)</strong></td>
<td>300,000</td>
<td>710,000</td>
</tr>
<tr>
<td><strong>Fatigue life (cycles)</strong></td>
<td>331,584</td>
<td>718,805</td>
</tr>
<tr>
<td><strong>Residual strength (KN)</strong></td>
<td>--</td>
<td>--</td>
</tr>
<tr>
<td><strong>Static failure type</strong></td>
<td>--</td>
<td>--</td>
</tr>
<tr>
<td>Series</td>
<td>Type of doubler and faying surface</td>
<td>$N_{\text{basic}}$ (k c)</td>
</tr>
<tr>
<td>--------</td>
<td>-----------------------------------</td>
<td>--------------------------</td>
</tr>
<tr>
<td>A-Series</td>
<td>Alclad 2024 T3 1.6 mm with PR 1422 B2 sealant</td>
<td>23</td>
</tr>
<tr>
<td>B-Series</td>
<td>Glare 3 3/2 1.4 mm with PR 1422 B2 sealant</td>
<td>49</td>
</tr>
<tr>
<td>C-Series</td>
<td>Glare 3 4/3 1.95 mm with PR 1422 B2 sealant</td>
<td>125</td>
</tr>
<tr>
<td>D-Series</td>
<td>Glare 3 3/2 1.4 mm with AW 106 adhesive</td>
<td>71</td>
</tr>
<tr>
<td>E-Series</td>
<td>Glare 3 3/2 1.4 mm with EC 2216 B/ A adhesive</td>
<td>94</td>
</tr>
<tr>
<td>F-Series</td>
<td>Glare 3 3/2 with EU 9323 B/ A adhesive</td>
<td>86</td>
</tr>
<tr>
<td>G-Series</td>
<td>Alclad 2024 T3 1.6 mm with PR 1422 B2 sealant and countersunk flaws on skin</td>
<td>28</td>
</tr>
<tr>
<td>H-Series</td>
<td>Alclad 2024 T3 1.6 mm with EU 9323 B/ A adhesive and countersunk flaws on skin</td>
<td>24</td>
</tr>
</tbody>
</table>
Table 6-21, Fatigue performance of Barrel test with gross stresses of 115 - 120 MPa and R = ± 0.05.

<table>
<thead>
<tr>
<th>Type of joint</th>
<th>Crack by NDT, (k c)</th>
<th>Last Visual Insp, (k c)</th>
<th>Visible Crack, (k c)</th>
<th>Fatigue, (k c)</th>
<th>Failure Type</th>
<th>Failure Location</th>
</tr>
</thead>
<tbody>
<tr>
<td>Al 2024-Al 2024, Type I, 1st</td>
<td>36</td>
<td>101</td>
<td>-</td>
<td>102</td>
<td>MSC at First row</td>
<td>Upper sheet</td>
</tr>
<tr>
<td>Al 2024-Al 2024, Type I, 2nd</td>
<td>36</td>
<td>79</td>
<td>-</td>
<td>81</td>
<td>MSC at First row</td>
<td>Upper sheet</td>
</tr>
<tr>
<td>Glare 3-Al 2024, Type II, 1st</td>
<td>40</td>
<td>189</td>
<td>No, visible</td>
<td>191</td>
<td>-</td>
<td>At clamping on Al</td>
</tr>
<tr>
<td>Glare 3-Al 2024, Type II, 2nd</td>
<td>30</td>
<td>-</td>
<td>135</td>
<td>184</td>
<td>MSC at First row</td>
<td>Upper sheet (Glare)</td>
</tr>
</tbody>
</table>

Note: MSC is multiple site crack
Figure 6-28, Effect of faying material on aluminum patch in the non countersunk skin.

Figure 6-29, Effect of faying material on aluminum patch on countersunk skin.
Figure 6-30, Effect of faying material on Glare patch at non countersunk skin.

Figure 6-31, Effect of faying material on Glare patch at countersunk skin.
Figure 6-32, Effect of countersunk hole on Aluminum patch with sealant as a faying surface.

Figure 6-33, Effect of countersunk hole on aluminum patch with Araldite as a faying surface.
Figure 6-34, Effect of countersunk hole on Glare patch with sealant as a faying surface.

Figure 6-35, Effect of countersunk hole on Glare patch with Araldite as a faying surface.
Figure 6-36, Effect of patch material on non-countersunk skin with sealant as a faying surface.

Figure 6-37, Effect of patch material on countersunk skin with sealant as a faying surface.
Figure 6-38, Effect of patch material on non countersunk skin with Araldite as a faying.

Figure 6-39, Effect of patch material on countersunk skin with Araldite as a faying surface.
7. REPAIR ASSESSMENT PROGRAM

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In June 1988, Federal Aviation Administration (FAA) sponsored an International Conference on Aging Aircraft, the Air Transport Association of America (ATA) and the Aerospace Industries Association (AIA) initiated a program to define and recommend actions required to ensure continuing structural airworthiness of repairs in aging aircraft of the transport category [1]. The ATA and AIA established an Airworthiness Assurance Working Group (AAWG). The objective of this working group was to identify the maintenance requirements which might influence the safety of the aircraft as follows:

- Selection of the existing Service Bulletins (SB) to be recommended for aging aircraft,
- develop a corrosion prevention and control program,
- review the structural integrity program (SIP),
- review the basic maintenance program, and
- develop a repair assessment program which can determine the classification of a repair that include the damage tolerance analysis.

This chapter discusses the repair classification process using a simple computer program concerning the fatigue, static and damage tolerance of a repair in an aluminum fuselage skin. The program has the objective to considered repairs outside the door area, pressure bulkheads, and cockpit area. This computer program provides an additional guideline in determining the class of a repair; it should not be used for a primary evaluation in the certification process of a repair. The use of Glare as a repair material has not been included yet.

7.1 Introduction

Concern has been raised about the quality of repairs on existing aircraft. Most repairs
currently in service, were designed to static strength and fail safe criteria for aircraft certified prior to FAR Amendment 25-45 (1978). In general, these repairs have been kept safe by the existing maintenance programs and a sufficient durability resulting from pre-FAR 25-45. As the fleet ages, some repairs may be approaching a need for supplemental inspections to maintain the damage tolerance. Therefore, the new repair methods and repair materials have been investigated in the Material and Construction laboratory of the Delft University of Technology, Faculty of Aerospace Engineering for assessment and to increase the quality of a repair on the current aircraft.

The program includes the following processes:

- calculation of the static strength of a repair,
- calculation of the severity factor, and
- calculation of the inspection threshold and repeat inspection interval.

The basis of the Repair Calculation and Classification Program (RCCP) are the static test results and the fatigue experiments done in the laboratory (see chapter 6 of part 1) and from references [9] and [10]. To calculate the inspection threshold and repeat inspection interval for class B and C repairs, the fatigue basic and the interval basic found in chapter 6 should be multiplied by a series of factors (i.e. the influence of the pitch distance, rivet diameter, doubler thickness, etc.), where each factor accounts for the influence on inspection intervals. This program is written in Turbo Pascal version 7.0 and it is available on disk, at the Material and Construction Laboratory of the Delft University of Technology, Faculty of Aerospace Engineering.

The purpose of this RCCP is to reduce the man-hours to evaluate a repair. This computer program may reduce the man-hours with about 40 to 70 percent (depending on the complexities of a repair) as compared with the manual evaluation using a Repair Assessment Program as provided by the manufacturer.

The program provided in this chapter is not an approved guideline; therefore the use of this program in maintenance is not the responsibility of the author.

### 7.2 Repair Assessment Compliance, Logic and Procedure

The RCCP compliance of a repair on a fuselage skin with an aluminum doubler is mainly based on fatigue damage consideration, due to the pressurization load, while for covering the effects of the Environmental Damage and Accidental Damage, should be referred to the aircraft specification provided by the manufacturer or the Maintenance Steering Group 3 (MSG 3) document.

The Repair Assessment Task Group (RATG) have been coordinated with the industries, that reached a general approach for the identification of repair. Furthermore an industry agreement was reached on the repair classification which is divided into four categories as follows:

- **Category A:**
A permanent repair for which the existing zonal inspection program or baseline inspection program is adequate to ensure the continued airworthiness of the aircraft and this repair should be equal or better to un-repaired surrounding structure.

- **Category B:**
  A permanent repair which requires supplemental inspections to ensure the continued airworthiness of the aircraft.

- **Category C:**
  A time limited repair or temporary repair which satisfies the aircraft airworthiness requirement, but requires frequent supplemental inspections. The threshold for the rework or replacement (retirement) must be provided in addition to the supplemental inspections.

- **Category D:**
  Substandard repair which does not meet the aircraft’s airworthiness requirements and must be replaced or upgraded to an acceptable level before further flights.

This RCCP provides a tool for maintenance to determine the classification of a repair or recalculating an existing repair, to ensure the continued airworthiness of the aircraft. Again, this RCCP should not be used as an approved document supplemental to the existing guidance material (such as the Structure Repair Manual, Service Bulletin, or other accepted data).

The procedure and the logic diagram for determining the classification of a repair involves several stages, which are presented in the flow chart, shown in Figure 7-1. These stages are as follows:

- **Stage 1:**
  Repair identification and data collection, in this stage all the repair data should be collected from the aircraft (see sub-chapter 6.3 of part 1 for the detail procedures). The repair data must be stored and documented on a repair data sheet.

- **Stage 2:**
  In this stage, the repair should be calculated and checked for static strength requirements, as shown in chapter 5 of part 1. If a repair does not satisfy the requirements, this repair must be categorized as category D repair. The severity factor of a repair joint is also calculated using equation 6-7 in chapter 6 of part 1.

- **Stage 3:**
  The stage 3 is the categorization process which determines the category of a repair for class A, B, C and D, as shown in references [9] and [10]. This process includes the effects of the use of countersunk hole, pitch distance, etc., which may effect the fatigue strength.

- **Stage 4:**
  In this stage, the calculation of the inspection threshold and the repeat inspection interval of the class B and C repair is carried out. For the class C repair, the retirement
Figure 7-2. External doubler layout.

Figure 7-3. Fastener layout on a repair.

7.4 Repair Calculation and Classification Program (RCCP)

The RCCP is intended for determining a repair on aluminum fuselage skin only. This
program shall be run under the Turbo Pascal software. This computer program allows the maintenance engineers to compare their calculation for a repair with this result. To run this RCCP, the engineer should know about stress analysis, fatigue and be familiar with the field repair data. Otherwise, the combination of this view input may result in numerous combinations of out-put; therefore careful identification of input is necessary.

RCCP provides the following results:

- Static strength calculation,
- inspection threshold and repeat interval for a repair in category B or C, and
- severity factor.

### 7.4.1 Sample Calculation of Using RCCP

This section provides a sample of the typical input-output calculation of the RCCP. The repair data used of this calculation is taken from the F-28, the location of this repair is at the lap joint above the window belt on the middle section of the fuselage, Figure 7-4 shows the repair skim of this repair. This lap joint was a bonded-riveted joint, but in this calculation, the lap joint is assumed as a riveted joint with a sealant as a faying surface. Furthermore, the data inputs ask by the RCCP are as follows:

- **Fuselage radius** : 1.75 mm,
- **skin** : Al 2024 T3
- **thickness** : 1.4 mm,
- **skin E modulus** : 72,000 MPa,
- **skin ultimate strength** : 420 MPa,
- **running load (K_{max})** : 350 N/ mm
- **doubler** : Al 2024 T3
- **doubler thickness** : 1.45 mm,
- **doubler E modulus** : 72,000 MPa,
- **doubler ultimate strength** : 430 MPa,

**dimensions of the repair**

- **pitch distance** : 20 mm,
- **row distance** : 20 mm,
- **edge margin** : 10 mm,
- **frame distance** : 500 mm,
- **cut-out in flight direction** : 400 mm
- **Over the lap joint** : yes/ no
- **type of rivet** : NAS 1097 AD 5
- **rivet diameter** : 4.0 mm,
- **rivet ultimate strength** : 400 MPa,
- **rivet shear strength** : 150 MPa,
- **countersunk depth** : 0.9 mm
- **rivet E modulus** : 65, 000 MPa,
- **number of rivet rows** : 3
• aircraft data
• D-check (major check) interval : 5000 cycles
• Time of repair embodiment : 18,000 cycles
• fatigue design life : 90,000 cycles

The typical RCCP output is as follows:

• Repair category : B,
• inspection threshold : 38,000 cycle,
• repeat inspection interval : 5,000 cycle, and
• severity factor (SF) : 1.8

This result presented here as a demonstration, that a wide range of path materials and thicknesses can be managed by the RCCP. For example, we can change the thickness of the doubler and then the RCCP will immediately provide the result of the effect of increasing the thickness on inspection threshold, bearing stress, severity factor, etc. When the severity factor is increased, it indicates, that the fatigue strength of this repair is reduced.

7.4.2 Discussion of RCCP Solution

The result of this RCCP is in agreement with the results of the Repair Assessment Program issued by Fokker [9]. The man-hours needed to determine this repair using the Repair Assessment Program [9] is 2 and with the RCCP it is 0.8 or 65 percent less than the manual one. So this program will be very useful for the maintenance engineer, to use it as an additional aid in aircraft repair.

The use of the Glare material and the bonded-riveted joint in the RCCP will be included in the next version. During the use of this program, when found an unreasonably result or information, please reports it to the address mentioned in front of this book.
7.5 References


[5] Aircraft Inspection and Repair, Acceptable Methods, Techniques, and Practices, Department of Transportation Federal Aviation Administration, United States of America, AC 43.13 - 1A.


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Figure 7-4, The repair skim of a lap joint repair on the fuselage.
8. CONCLUSIONS

Approval Process

1. The application of Glare for fuselage skin repair in the existing aircraft could be classified as a major alteration due to the changes of the original design of the primary structure.

2. The approval of Glare as a repair material for the existing aircraft could be obtained from the aircraft manufacture by issuing a Service Bulletin (SB). Where in the SB is mentioned the reason of the alteration, effectivity of the aircraft, limitations, accomplishment instructions, etc.

Riveted and Bonded-riveted Joint

1. The load transfers of Glare 3 4/3 (2 gages thicker) and aluminum 2024 T3 1.6 mm in lap joint results a comparable load distribution in the first and the third rivet rows. This is due mainly to the Glare which has moderate elastic modulus.

2. Cold bonding adhesive in riveted lap joint has been reduced the load transfer in the first rivet row significantly. The load transfer in this rivet row is only transferred with about 0.5 to 2 percent of total load.

3. The differences of the load transfer for a joint with two rivets and three rivets rows at the first rivet row are very small or can be said the same. But for a good repair and from the damage tolerance point of view, the repair should always be desired to use a minimum of three rivet rows.

4. In patch repair, Glare patches result less load attraction into the repair area as compared with the aluminum patch repair due to a little lower elastic modulus relatively to the aluminum.

5. For the small cut-out or called patch repair, the differences of the load transfer of the rivets at first row are small for both types of patches (Glare and aluminum).

Static Test

1. The static test results of the external patch repair for both types of patch materials (Glare and aluminum) satisfy the minimum static strength requirement caused by the hoop load.
CONCLUSIONS

2. A thinner Glare of 90 percent of the skin, did not degrade the static strength as compared with aluminum patch repair. This is due to the failure occurred at the first rivet row because of the net stress.

3. Cold bonding adhesive (AW 106) as a faying material in patch repair did not increase the static strength.

4. In the lap joint repair, cold bonding adhesives as a faying surface increases the static strength significantly. It also changes the failure mode from the rivet shear to the skin fracture. It indicates that the adhesive has a significant effect in reducing the load transfer through the rivet.

5. The combination of a thinner Glare (90 percent of the skin thickness) to an aluminum reduces the static strength with about 15 percent as compared with the merely aluminum joint, but it passed the ultimate load for this specific skin thickness.

**Fatigue Test**

1. The use of the former countersink holes underneath the doubler in patch repair reduces the fatigue life with about 40 to 50 percent as compared with the non countersunked one for both types of fayings (sealant and adhesive).

2. The application of Glare sheet with cold bonding adhesive as a faying surface in patch repair results the highest fatigue life. One of the two specimens failed at the edges of the patch due to the abrupt changes with higher fatigue life as compared with the other one.

3. The Glare patch with sealant as a faying surface in patch repair results a higher fatigue life as compare with the aluminum patch repair. This result confirms that Glare patch resulted less load attraction into the repair area as shown in load transfer calculation.

4. Cold bonding adhesive as a faying surface in lap joint repair increases the fatigue life significantly. These results confirms that adhesive may reduce significantly the load transfer trough the rivets as shown in the load transfer calculation.

5. The combination of a thicker Glare 3 4/3 (2 gauges) to an aluminum in lap joint repair results in favorable fatigue life. Furthermore, the application of the old repair concept to the Glare sheet as a repair material can be adopted without any consequences in fatigue life as well as for the static strength.

**Curved Test**

1. The effect of the curvature reduces the fatigue life significantly as compared to flat specimens. The reduction of the combination Glare 3 3/2 (1.45 mm) to aluminum 2024 T3 (1.6 mm) is about 1.4 times lower and for the merely aluminum joint is
CONCLUSIONS

about 2.9 lower. Again, these test results demonstrate that Glare as a repair material is superior in fatigue as compared to the aluminum.

2. The crack growths of the aluminum specimens were incredibly fast, but for the combination Glare sheet to an aluminum was incredibly longer than the aluminum. It means that Glare repair is much more damage tolerance as compared to the aluminum.
Part 2: GLARE CARGO FLOOR

1. INTRODUCTION

As mentioned in part 1, Glare has many advantages as compared to the aluminum alloy sheet materials, i.e. superior properties with respect to fatigue strength, impact resistance, corrosion behavior, etc. The application of Glare as a repair material shows a significant improvement in the fatigue strength of the patch repair or the lap joint repair. Concerning the impact properties and the corrosion behavior, Glare can be used for aircraft floor panel application. Impact damages and corrosion are the most serious problems of aircraft cargo floor panels.

In this part investigates the optimum floor panel with respect to impact damage, corrosion, bending and compression. To achieve these requirements, several floor panel configurations with various of the top face and core materials are tested.

1.1 Reasons for Modifications

Aircraft bulk cargo floor panels consist of a low density core with a stiff skin on the top and the bottom. They offer considerable potentials for weight saving. The main loads of the floor structure are transverse shear, bending and impact. The original cargo floors used by the airlines today have a low resistance to impact during cargo handling and are prone to corrosion for metallic floors. Therefore a repair or a replacement of the floor panels as a result of puncture and/or corrosion, frequently occurs. A short life of floor panels will cause a heavy maintenance burden for the airliners. It will lead to an increasing aircraft down time, due to inspection, repair or replacement. It then becomes progressively expensive in view of the maintenance cost.
The experience of the airlines shows that most cases of floor repairs or replacements were due to corrosion, delamination, puncture or dents, followed by a surface crack. If such types of damage occurred on the floor, then the strength of the floor will be reduced significantly, and consequently will affect the airworthiness of the aircraft. Current floor structures for aircraft cargo are the sandwiches with aluminum sheet facing and different types of cores, viz., a nomex core (non metallic honeycomb core), an aluminum honeycomb core, or a balsa wood core. Corrosion of a metallic floor may start and grow progressively after a through crack occurred at the facing, due to impacts or scratches. Therefore the impact properties of cargo floor are an important criterion in floor design. Figure 1-1 shows typical damages of a cargo floor.

Figure 1-1, Typical damage of the cargo floor due to impact and scratch

The aim of the present research is to develop a new floor panel which can improve the corrosion behavior and the impact properties. Hence, the new material Glare® 5 has been chosen as one of the suitable materials for the top facing of a floor panel, developed at the Delft University of Technology. Glare® 5 is a special type of Glare for floor application. It has good properties with respect to impact, corrosion and burn resistance. It passed FAR 25.855 for burn requirements. Furthermore, to optimise the balance between cost and reliability of the floor, many types of floor configurations with a variety of top faces and core materials were tested. The strength to weight ratio of the new floor should also be considered in depth.

In our research, a drop weight tower test was used to determine the impact property of the floor. Afterwards, some specimens which experienced in impacts were exposed in a salt spray chamber for a period of minimum 1000 hours. The purpose of this test is
to obtain information about the corrosion behaviour of the floor panels after an impact.

1.2 Glare Floor

There are two different types of cargo floors used in aircraft. That are the bulk type cargo floor and the container type cargo floor. Both types of floor have different functions and requirements. In the bulk cargo floor, the loading and unloading of the cargo, will directly be placed on this floor. Therefore, this bulk type cargo floor will carry the loads directly during this process and also during flight. The container cargo floor usually is used only for maintenance and inspection purposes, therefore this floor is only used by a technician for support. Figure 1-2 and Figure 1-3 show the bulk cargo floor and the container cargo floor, respectively.

Figure 1-2, Bulk cargo floor.

There are many advantages of Glare which lead the Airbus company to inquire into the feasibility of manufacturing and service evaluation tests of bulk cargo panels with Glare as the top face in A 330 aircraft of Garuda Indonesia airlines. The purpose of this study is to obtain information concerning different solutions for the cargo floor. The concepts include Glare panels for usage in areas with a high potential for impact, to reduce or even prevent the floor from damage. Furthermore the aim of this study is to provide airlines with alternative solutions to the current cargo floors, in order to reduce maintenance cost and aircraft down time.
1.3 Current Maintenance Practice for Inspection, Repair, or Replacement

A general inspection of the bulk cargo floor is required before loading the bulk cargo to ensure the airworthiness of the aircraft, because the aircraft bulk cargo floors are structural parts of the aircraft structures. Hence, the floor panel should be capable in carrying the load during the flight. Especially in the event of a heavy landing or crash, the floor has to carry a high shear load.

The inspection of the floor is done by a general visual inspection. When the inspectors find a damage on the floor panel, the extent of the damage should be assessed in accordance with the Structure Repair Manual provided by the manufacturer. When the damage is beyond allowable limits, then a repair or a replacement must be carried out before a further flight.
1.4 References


INTRODUCTION
2. FLOOR PANEL DESIGN

2.1 Floor Design

The most commonly used aircraft bulk cargo floors consist of aluminum facing with a nomex or a balsa wood core. The purpose of using aluminum facing is due to its impact behavior, which is better than for fiber glass, but it is prone to corrosion.

In order to achieve an optimum floor panel in weight, reliability, impact, bending, etc., the following items should be considered during the design process:

1. Impact resistance of the top face,
2. impact properties of the core,
3. fire resistance of the floor panel,
4. core density,
5. corrosion behaviour of the top face and core, and
6. type of the adhesive.

During impact, the type of support of the panel has a significant effect on the impact damage of the panel. Figure 2-1 shows schematically the failure of a floor panel after impact on a rigid support. It can be seen that the top face and core just underneath the damaged area absorb the impact energy, while the bottom face does not. During the impact, the top face carries the tension loads as a membrane and the core will receive compression loads. Therefore, the top face should have good tension and impact properties while the compression properties of the core are also significant. In the open support, the floor panel will deflect. This deflection results into bending on the panel during impact. This bending may improve the impact behavior of the panel. Figure 2-2 shows a schematic diagram of a panel after the impact on the open support. The bending stiffness of the floor panel will depend on the compression and the ultimate strength of the facing material. In this case, the bottom facing will experience severe tension loading, therefore the ultimate strength of this facing has a significant effect. A high strength material like Al 7075 for the bottom facing is suitable from the strength point of view, but it is less corrosion resistant as compared to clad Al 2024.
Compromises between advantages and disadvantages must always be considered during the design.

Figure 2-1, Schematic failure mechanism of sandwich structure on rigid support.

Figure 2-2, Schematic failure mechanism of sandwich structure on open support.

In bending, the compression properties of the core, the ultimate strength of the bottom face and the bonding quality between core and facings have a significant effect. During the bending process, the core will receive the compression loads, and the adhesive will receive the shear loads. Therefore the compression strength of the core and the ultimate strength of the bottom face are important. In short, beam bending or three points bending, the core shear and the bondline failure are critical. Figure 2-3 shows a schematic drawing during the bending process of a panel as short beam bending.

Bending and impact are two important floor design issues for the size of the panel in view of optimising the strength and the weight of the panel during the material selection. To achieve comparable impact and bending properties of a floor panel, the rule of thumb in floor design is, that a floor with a comparable compression strength of the core material will result in comparable impact and bending properties.
The design requirement of the bulk cargo floor must be in accordance with FAR/JAR 25.561. The following is the general requirement provided by Daimler-Benz Aerospace Airbus for the bulk cargo floor of A330:

- The floor must be able for loading and unloading of bulk cargo,
- they should be provided with an edge filling by a putting compound, which prevents the ingress of moisture and to support the edges of the panels during loading.
- the design of hole reinforcements or inserts for floor attachment to the aircraft structure should be such: a) they must be able to introduce the loads into the structure throughout the floor panels, b) they do not appreciably protrude, and c) the protrusions in form of insert flanges are not allowed to happen on the top face.
- the floor panels shall have good impact properties when an object is dropped.
- the floor panel intended for bulk cargo transport must be covered with an anti slide surface. Then it must be ensured that it is safe for people walking on this floor even when wet. Damage to the bulk cargo due to an anti slide is not permissible.

The properties of such a bulk cargo floor should be determined by tests to meet the requirements in an approval process. The test programs required by the Airbus Industries for the bulk cargo floor are as follows:

1. Long beam bending (ASTM C 393),
2. bending under static load (DA requirement),
3. insert shear (DA requirement),
4. impact loads (ASTM D 3029),
5. climbing drum peel test (ASTM 1781),
6. stabilized core compression (ASTM C 365), and
7. warpage, where this test is to check the flatness of a panel (DA requirement).

2.2 Material Selection for Facing and Core
As mentioned in the above section, the facing material should have good impact properties and corrosion behaviour. To achieve this requirement, a sheet material like Glare® 5 can be used as the top face for this application, where this material has good impact properties for low impact velocity, corrosion behavior and it meets the burn-through requirement (FAR 121.314). For a new aircraft design, these requirements become mandatory. Therefore the material selection process is an important step in floor design.

The ultimate strength and the impact properties of the facing materials are important in floor design, as mentioned above. When a crack occurs in the top facing, the impact energy of the panel will reduce tremendously. The use of a high strength material like Al 7075 will result in surface cracks due to its low yield capacity. A suitable material for the top face should have a high yield capacity, should not be sensitive to a surface crack, and should have good corrosion resistance. In this case, Glare 5 is a suitable material for the top face. For the bottom face, the material should also have a good ultimate strength and good corrosion behavior. Therefore the use of Al 2024 T3 in the present research is the proper selection for the bottom facing.

Compression of the core has a significant effect on impact and especially on bending, as mentioned in section 2.1. In bending, the reduction of the total thickness of the panel due to the compression load, will considerably reduce the bending stiffness of the floor panel. To select the core material, the compression strength to weight ratio can be used as the parameter [11], the higher the ratio, the better the impact and the bending properties will be. For example, to achieve a comparable compression strength, a Nomex core has a density of 128 kg/ m³, the density of a foam core (Rohacell) is 210 kg/ m³, and for a balsa core it is 140-150 kg/ m³.

Water absorption in a core like balsa, is another important parameter to be considered during the material selection, it will have a significant effect on the corrosion and delamination of the bonding. When using balsa for the core, the edges of the panel should be protected with a sealing or putting compound.

Figure 2-4 shows the configuration and the type of materials used for a bulk cargo floor of Airbus 330 Garuda Indonesia Airline.

2.3 Adhesive Selection

The adhesive has a significant effect on the properties of the floor panel. The criteria to select the adhesive are as follows:

1. The strength of the adhesive should have a comparable strength with the core or the facing materials,
2. the adhesive should give a good bonding quality between the adhesive and the facings and between the adhesive and the core.
3. the adhesive should not be too fragile, and
4. the curing temperature for hot bonding should not be higher than the limits as for the core and the facing materials.
If the strength of the adhesive is much higher than the strength of the core or the facing materials, it will not increase the properties of the panel. The higher the strength of the adhesive, it usually is more expensive. As a result, the price of the floor will be considerably higher.

The curing temperature has a significant effect on the quality of the floor panel. A higher curing temperature will result in a higher deformation of the panel after the curing due to the different thickness and thermal expansion coefficient of the facing materials. Thus, the low curing temperature is desirable for floor panel production. Preferably a cold bonding adhesive should be used if possible.

2.4 Approval Procedure of a Newly Cargo Floor

The application of a new floor panel in an existing aircraft requires approval before it can be used. The approval of this floor can be carried out either by the airline with their local authorities or by the aircraft manufacturer by issuing a Service Bulletin (SB).

The application of a new cargo floor can be categorized as a major or minor alteration, depending on the extent of these changes to the original design of the structure. As explained in chapter 3 of part 1, that even small changes should be approved by the airworthiness authorities. In the present chapter guidance is given to assess whether the alteration shall be classified as a minor or a major alteration.

For example, the modification of the container floor of an MD-11, can be classified as a minor alteration because this floor is a secondary structure and this modification does not effect the structural integrity of the aircraft. So the approval of this modification can follow the process as written in chapter 3 of part 1.
Current floor panel

- 1
- 2
- 3
- 4
- 5
- 6
- 7
- 8

Glare floor

- 1
- 2
- 3
- 4
- 5
- 6
- 7

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<tr>
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</tr>
<tr>
<td>6</td>
<td>Prepreg</td>
<td>0.250</td>
</tr>
<tr>
<td>7</td>
<td>Prepreg</td>
<td>0.175</td>
</tr>
<tr>
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<tr>
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<tr>
<td>7</td>
<td>Al 7075 T3 .4 mm</td>
<td>1.120</td>
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<td>Total</td>
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Figure 2-4, Bulk cargo floor configurations of Airbus 330 of Garuda Indonesia Airline.
2.5 References


3. FLOOR PANEL PROCESSING AND MANUFACTURING

3. FLOOR PANEL PROCESSING AND MANUFACTURING  163
3.1 Material Preparation and Surface Treatment  163
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This chapter discusses the manufacturing process of the floor panel. The process described is used in Structures and Materials Laboratory, of the Delft University of Technology Faculty of Aerospace Engineering.

3.1 Material Preparation and Surface Treatment

The material preparation for floor panel manufacturing is an important step to achieve a good quality of the panel. Cleaning process and surface treatment are the two processes with large effect on the bonding quality of the core and facings. Failures in preparing these processes may lead to reduce the strength and the durability of the panel. All surfaces to be bonded must receive a good cleaning process and surface treatment.

Glare 5 used in our experiments were already treated by chromic acid anodizing and the primer BR 127 as surface treatments. These treatments are an option for Glare sheet while other surface treatments may also be requested when it is necessary. So the preparation process for the Glare sheet was only a cleaning process, using acetone just before bonding.

The aluminum facing used in our experiments is Clad Al 2024 T3 0.8mm and Clad Al2024 T3 0.5 mm as the top face and Clad 2024 T3 0.3 mm as the bottom face (for all panel configurations). The surface treatment of the aluminum sheet was:

- Remove fat, dirt and contaminants using a solvent (i.e. acetone),
- oil degreasing using alkaline,
- remove aluminum oxidise using a pickling process with the chromic sulphuric acid,
- drying, and
- apply the chromic acid anodizing just after drying (not longer than 30 minutes).

The cores used were foam core Rohacell 110, balsa wood and the nomex core. The surface treatments applied on the balsa wood and the nomex cores are a cleaning
using a high air pressure and acetone, while for the foam core it is only cleaning, using high air pressure.

Tabel ??? shows the floor panel configurations.

Table 3-1, Floor panel configurations.

<table>
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<tr>
<th>Spc #</th>
<th>Top skin</th>
<th>Core</th>
<th>Bottom skin</th>
<th>Remark</th>
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<td>Nomex</td>
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<td>Clad Al 2024 T3, 0.3</td>
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<td>Nomex</td>
<td>Clad Al 2024 T3, 0.3</td>
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<td>Clad Al 2024 T3, 0.3</td>
<td></td>
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<td>Spc #7</td>
<td>Bare Al 2024 T3, 0.5</td>
<td>Balsa</td>
<td>Bare Al 2024 T3, 0.3</td>
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<td>Spc #9</td>
<td>Glass fiber</td>
<td>Nomex</td>
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</table>

3.2 Curing process

In order to achieve the maximum quality of the floor panels, the curing process was carried out in an autoclave. The vacuum pressure, outside pressure and temperature were controlled in accordance with the specification of the adhesive as given by the manufacturer. A vacuum bag is used to seal the panel in order to apply the vacuum pressure to obtain an adequate bonding quality. When the pressure is applied, it must be sure that the vacuum bag is sealed properly over the entire panel. An initial pressure of 5 to 8 psi shall be established for 5 to 10 minutes to evacuate the entrapped air. Then full pressure of about 20 to 30 psi shall be applied prior to the application of heat for the adhesive cure. The vacuum bag installation is shown in Figure 3-1.

The cure temperature is measured by a thermocouple placed at the top of the panel. After all above preparations have been made, the heating process for curing the
adhesive is applied. The adhesive was AF 163, manufactured by 3 M. The curing cycle is shown in Figure 3-2. The temperature is raised at a maximum rate of 3⁰C per minute to 120⁰C ± 5⁰C, followed by a cure at this temperature for a minimum of 60 minutes. The temperature on and around the panel must be monitored and controlled carefully.

After the curing process, the panel inspection is as follows:

- general visual inspection for the surface condition, i. e. damage, wrinkling, or any other form of damage,
- determine the warpage of the edges of the panel in accordance with Daimler Benz Aerospace Airbus ECB-10-01-96 [11], Figure 3-3 shows the measurement of the warpage on a panel, and
- check the delamination or bad bonding quality using a tapping test with a coin.
Figure 3-1, Vacuum bag application for sandwich curing process.
Hold for 60 minutes at 120 ± 6°C

Heat-up 1°C to 3°C per-minute
Cool-down 3°C to 5°C per-minute

Bellow 40°C
Release pressure

Room Temp ——— Time ———

Figure 3-2, Cure process of adhesive AF 163.

Figure 3-3, Warpage measurement of a panel after curing process.
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4.5 Service Evaluation Test of a Glare Panel for the MD-11 Aircraft and Glare stiffeners for the Fokker F-28 Mk-4000 .......... 184

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To achieve the optimum floor panel configuration, there are four test programs which shall be considered to determine the properties of the floor panel before it continues for a further test program. The four test programs are the impact, bending, compression and corrosion. These test programs are suitable to evaluate the floor panel configuration which gives the optimum performance.

This part described the results of the floor panel tested in the Structures and Materials Laboratory, of the Delft University of Technology Faculty of Aerospace Engineering.

4.1 Impact Test

As mention before, impact is an important criterion for a bulk cargo floor. Therefore, impact test were carried out as a first step of the test program. After these test, promising panel configurations (see Table 3-1 in chapter 3 of part 2) could be used to continue a further test programs.

This section describes the results of the impact tests with various top faces and core materials. The purposes is to determine the influences of the facing surface and the core material on the impact properties of the panel.

There are several variables that can affect the impact resistance of a panel, such as the nose shape of the impactor, mass, material hardness, impact velocity, obliquity
(impact angle), dimensions of the panel and specimen support (boundary condition). Because of the complexities of the impact process and very limited theoretical possibility to predict the impact properties of a panel, impact tests are the only acceptable method for the airworthiness authority for approval of a new floor panel in civil aviation.

The impact tests were carried out with the B-2 tower impact tester [2] developed by Structures and Materials Laboratory of Delft University of Technology of the Faculty of Aerospace Engineering. The test set up and the schematic diagram of the B-2 impact tester are shown in Figure 4-1 and Figure 4-2. The Airbus standard for the impact test is the tower drop test as mentioned in ASTM D 3029. The difference of the two methods is associated with the accuracy measured for the impact energy. The impact energy of the B-2 tower impact test is measured as the kinetic energy of the impactor just a few millimetres before the impactor touches the panel. The speed of the impactor is measured by a pair of photo cells, which together with the mass of the impactor is used to calculate the impact energy. In the ASTM D 3029 tower drop test, the energy is derived from the height of the impactor before it is released throughout the tubing. During the impactor travelling inside the tubing, there are some energy losses due to friction between the impactor and the tubing, and due to the air inside the tube, which will be compressed by the impactor. The compressed air inside the tube will act as a damping effect to the impactor, and it thus reduces the impact energy. Due to these losses the nominal impact energy of this impact test will be somewhat higher as compared with the result of the B 2 impact tester. The B-2 impact tester is more accurate than the tower drop test (ASTM D 3029). The impact energy of the B-2 can be varied by changing the height or the weight of the impactor. The impact energy of the B-2 impact tester can be calculated as follows:

\[ E_a = 0.5 M V_i^2 - M g h_m \]  

4-1

Where:

\( E_a \) is the impact energy,
\( M \) is the mass of the impactor,
\( V_i \) is the velocity of the impactor just before it reaches the panel,
\( g \) is the acceleration of the gravity, and
\( h_m \) is the height of the rebound of the impactor.

However, during the experiment, the height of the rebound of the impactor was very small, especially when significant damage occurred on the panel, then almost no rebound could be seen. Therefore the potential energy of the impactor due to the rebound can be neglected. The impact energy of the panel is equal to the kinetic energy of the impactor just before hitting the panel:

\[ E_a = 0.5 M V_i^2 \]  

4-2
From reference [2], the average velocity \( V_{av} \) of the impactor can be derived from the relationship between \( V_{av} \) and \( V_c \) as follows:

\[
V_{av} = g \left[ \frac{AB}{\left[ V_c^2 - 2gBC \right]^{1/2} - \left[ V_c^2 - 2gAC \right]^{1/2}} \right] \tag{4-3}
\]

where \( AB \) is the distance between the photo cells of about 30 mm and \( BC \) is the distance between the lower photo cell and the panel surface. Vlot [2] mentioned that the correction for the speed of the impactor is not necessary. This error is less than 1%, therefore the speed of the impactor is:

\[
V_{av} = V_{AB} \tag{4-4}
\]

where \( V_{AB} \) is the average velocity between the photo cells, the impact energy of the panel can be written as follows:

\[
E_u = 0.5M \left[ \frac{AB}{t_{AB}} \right]^2 \tag{4-5}
\]

where \( t_{AB} \) is the time between the photo cells and measured by a digital counter with a frequency of 1 MHz.

To determine the effects of the support on impact properties, two types of support have been used in the experiments: an open circular support, and a rigid support. The open circular support represents the impact on the middle of a floor panel. The rigid support represents the impact at the edges of a floor panel, where the edges of the panel are normally supported by the frames. In the circular support, the panel is allowed to deflect; at that will influence the impact behavior significantly. Figure 4-3 shows the schematic diagram of the supports.

The criterion to decide the panel failure or not, must be defined in order to measure the impact property of a panel. The panel is considered to fail when a crack occurred on the top face as visible by the naked eye under normal laboratory lightning conditions. This failure criterion is slightly different from the impact criteria used by the Airbus Industries [20]. According to the Airbus Industries, the crack is measured by using a needle, when the needle can go through the crack then the panel has failed. However, according to experience, differences between the two methods are not significant. Figure 4-4 shows surface cracks of damaged panel specimens.

The statistical method used to determine the impact energy is the Up and Down method as recommended by the ASTM D 3029. This method is appropriate to determine the means impact energy of the floor panel. The method can be described as follows: If the panel does not fail, then the impact energy in the next test shall be
to 12). From the test results can be determined in which range of energy the panel will fail, and when the panel does not fail. Furthermore, this range indicates the boundary of the impact energy of the panel, and this boundary is taken as the impact property of the panel. In order to meet the requirements in the statistical calculation, the numbers of the panel to be tested are among 8 to 12 specimens for each panel configuration.

Figure 4-1, B-2 test set-up.

Figure 4-2, Schematic diagram of B-2 Impact tester.
Figure 4-3, The support types used in this experiment.

Figure 4-4, Surface crack of damaged sandwiches

4.1.1 Test Results and Discussion of The Impact Test

In the rigid impact support, the panel with balsa or nomex cores failed due to core crushing or core collapse. The size of the dent or the damage area of the nomex core was limited, but for the balsa core or the foam core the damage areas are larger. The impact energy of the nomex core is lower than the balsa or the foam cores event with the same facing material. In this rigid support, the bottom facing has not influenced the impact property of the panel and there was no damage found on this facing. Figure 4-5 shows a cross section of the floor panel after the impact on the rigid support. It can be seen that the damages of all three types of cores occurred only underneath the top face. The bottom face is not affected by this impact. The differences of the impact energy between the Glare and the aluminum top facing were relatively small, but Glare 5 2/1 0.3 with a thickness of 0.85 mm has a little higher impact energy. Figure 4-6 shows the result of the impact test on rigid support with various panel configurations.
Figure 4-5, The cut-out of the floor panel after impact on rigid support.

Figure 4-6, Impact energy of various sandwiches on rigid support.

In the open support, the failures of the balsa core with all types of the top facings are core shear and core crushing underneath the impacted area. It means that the lignin link of the wood is critical in impact. The shear of the core due to the impact load caused the bottom facing to transfer the impact energy through its deformation. So, this type of failure mechanism increases the impact energy of the balsa core. Figure 4-8 shows the cut out of the balsa core after an impact. From this figure it can be seen that the delamination between core and facing (at the top and the bottom) is due to the core shear. Figure 4-9 shows the schematic failure of the balsa core.

The failure of the nomex core on the open support was due to the core collapse just underneath the impacted area while the lower part of the damaged core (closed to the bottom facing) was still intact. It means that the core far below the damaged area has taken less of the impact energy. The size of the damage was rather small or just as big

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as the impactor diameter and the core crushing only occurred underneath the top face at the impacted area. The bottom face did not receive too much load during the impact process, as indicated by a small change of the bottom facing, see Figure 4-8. From this figure it can be seen that, after the core collapsed just underneath the top face at the damaged area, the impact energy is no longer transmitted to the bottom face. The energy was spread out mostly through the top face around the damaged core. In this case, the top face became critical. The impact of the nomex panel resulted in a lower impact energy as compared to the balsa core.

The failure of the foam core on the open support was due to cell crushing underneath the impacted area, followed by core failure caused by the shear load during the impact. This failure continued to grow along the bond line between the bottom face and the core until it stooped at the edge of the clamping. The failure along the bottom face still occurred in the core itself. It means that the bonding between the core and the facing is still intact due to the strong adhesive material.

The impact energy of Glare panel with the balsa core resulted in the highest impact energy. The differences of both types of Glare top faces were negligible. The impact energy results of the open circular support are presented in Figure 4-7.

![Figure 4-7, Impact energy of various sandwiches on open circular support.](image)

The impact energy results obtained with the circular support show a significant difference as compared to the rigid one, see Figure 4-10, Figure 4-11 and Figure 4-12 the various types of cores. In the open circular support, the bottom face will contribute in absorbing the energy by its yielding. This impact process creates various ways for the panel to absorb the impact energy, i.e. core crushing, yielding, etc. From the figures, it can be seen that Glare 5 2/1 .3 with the thickness of 1.1 mm, always gives
the highest impact property. Therefore, the use of Glare as the top face on the floor panel implies a significant improvement.

Figure 4-8, The cross section of the floor panel after impact on open circular support.

Figure 4-9, The schematic diagram of the balsa core after impact on open circular support.

4.2 Bending Test

The bending strength is the other important parameter in floor design. This strength needs to be tested in order to determine the floor load per square meter of the aircraft. The aircraft floor shall be capable in carrying the load of the bulk cargo. This load becomes critical when the aircraft is in positive g-force or hard landing. Therefore the strength of the bulk cargo floor must be designed to withstand these conditions.

Bending tests have been conducted in accordance with the ASTM C 393 [7]. There are two methods to conduct the bending tests mentioned in this manual, viz., four points bending (long beam bending) and three points bending (short beam bending). The differences of the two methods are the size of the specimen to be tested and the failure mode. In four points bending, the types of failure are core compression and yielding of the bottom face. In three points bending, the central load is applied by one
point of loading and the types of failure are core shear and delamination between the bottom face and the core. In our experiments, the three points bending has been chosen to determine the bending characteristics of the floor panel. The reason is because it needs less material and it is also suitable in measuring the bending properties of the floor panel.

Figure 4-10, Impact of balsa sandwiches on rigid and open circular support.

Figure 4-11, Impact of nomex sandwiches on rigid and open circular support.
Figure 4-12, Impact of foam sandwiches on rigid and open circular support.

The test set-up of the three points bending is shown in Figure 4-13. The loads were applied with a constant rate through the displacement of the moveable lower table. The rate of loading in accordance with the ASTM C 393 is 0.45 mm per-minute or the maximum load shall be reached within 3 to 6 minutes.

The minimum number of specimens to be tested is 12. The sizes of the panel are 151.6 ± 1 mm (length), 76.2 ± 1 mm (wide) and 11 mm (thickness). Each specimen has been carefully inspected for detecting imperfections caused during the manufacturing and the cutting process. As mentioned before, the impact criterion is an important parameter in floor design. Therefore only one new panel configuration with the highest impact properties was used in the bending tests. The two types of panel configuration tested are the original floor panel of MD-11 as a basis for comparison and a new type of panel made of Glare 5 2/1 .3 as the top face, balsa core and AL 2024 T3 0.3 mm as the bottom face.

4.2.1 Test Results and Discussion of The Bending Test

The test results of the bending testing are summarized in Table 4-1. The failure modes of the two types of panels were core shear and delamination between the bottom face and the core. It shows that the shear strength of the core and the bonding quality between the core and the facing are the important parameter in the three points bending test. The delamination occurred between the adhesive and the core. Increasing the strength of the adhesive in this case will not give any improvement of the bending properties of the panel.

The difference between the bending properties of the new panel and the original panel of MD-11 was mainly associated with the density of the balsa core. The core density
of the new panel is 148 kg/m³ and the core density of the original MD-11 panel is 110 kg/m³. The bending properties of the panel is strongly affected by the core density. Therefore the new panel is much stronger than the original one. Core shear occurred at the panel above the support. The schematic of the failure mechanism of the balsa panel in three points bending is shown in Figure 4-14.

<table>
<thead>
<tr>
<th>Type of Sandwich</th>
<th>Max. Load in (Newton)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Newly designed panel</td>
<td>407</td>
</tr>
<tr>
<td>Glare 5 2/1 .3, Balsa and Al 2024 T3 0.3 mm</td>
<td></td>
</tr>
<tr>
<td>MD-11 Floor Panel</td>
<td>108</td>
</tr>
<tr>
<td>Al 2024 T3 0.5 mm, Balsa and Al 2024 T3 0.3 mm</td>
<td></td>
</tr>
</tbody>
</table>

Table 4-1, The result of bending test.

![Diagram of three points bending test set-up](image)

Figure 4-13, Three points bending test set-up.

![Diagram of core shear](image)

Figure 4-14, The schematic diagram of the failure mechanism of the balsa sandwich.
4.3 Compression Test

The compression test is used to determine the compression properties of the core, which are significant in floor design. In the aircraft floor, the main loads are compression and shear. The bending stiffness of a floor panel is very much depending on the compression strength of the core. When the panel thickness reduces due to core compression, the moment of inertia of the panel will also reduce. As a result, the bending behavior of the panel will change significantly. Akay and Hanna [3] mentioned that the compression strength of a core should be used as a baseline parameter in floor design. In chapter 2 the influence of the compression strength on the bending property of the panel was explained.

The density of a balsa core, can vary from 90 to 150 kg/ m³ because of the nature of the balsa wood. In order to be more precisely sure about the compression properties, the compression test shall be carried out for each batch of the balsa core. A solid core alike balsa wood, may result in good impact properties and it has also excellent fatigue resistance, shear strength, bond strength and noise insulation. The disadvantages of the balsa wood as a core material are the water absorption and the sensitivity to high temperature. A special treatment at the edges of the panel should be made in order to avoid water absorption along the edges.

A foam core has also a closed cell structure, but a foam core does not absorb water. The compression strength to density ratio of the foam core is lower as compared to the balsa or nomex core. Therefore a foam core e.g. Rohacell will result in a heavier panel as compared to the other two cores used in our experiments. As an example, for an equal compression strength as for a nomex core of 128 kg/ m³ or a balsa core of 140 kg/ m³, the density of the Rohacell foam should be of 220 kg/ m³.

The compression tests were carried out in accordance with the ASTM C 365 [6]. There are two methods to determine the compression property of a core, type A and type B methods. The type A method provides complete load and deformation data, so it gives the possibility to calculate the compressive stress at any load in relation to the modulus of elasticity. The type B method is an alternating method to measure the compression strength at core crushing. The test method used in our experiments is type B. The cross section of the specimen according to the ASTM C 365 is 50 by 50 mm. The high is depended in the requirements of the aircraft manufacturer, it usually is about 10 to 13 mm as mention in many Structure Repair Manual (SRM). The loads were applied at the rate of 140 Newton/ sec or the maximum load shall be reached within 3 to 6 minutes as suggested by the ASTM. The test set-up can be seen in Figure 4-15.

4.3.1 Test Result and Discussion of The Compression Test

The results of the compression test are summarized in Table 4-2. The compression strength is the maximum load at crushing of the core divided by the area. The values are the average of the number of the specimens. Table 4-3 shows several properties of the core material found in the literature and information from the manufacture. The
compression strength of the nomex core and the foam core are the same as provided by the manufacturers.

Table 4-2, Compression test results of various core materials.

<table>
<thead>
<tr>
<th>Type of core material</th>
<th>Density in kg/m³</th>
<th>Compression strength MPa</th>
</tr>
</thead>
<tbody>
<tr>
<td>Glare 5 - balsa wood core</td>
<td>148</td>
<td>14.04</td>
</tr>
<tr>
<td>Original MD-11 balsa wood core</td>
<td>133</td>
<td>10.3</td>
</tr>
<tr>
<td>HRH-10-1/8-9.0 nomex honeycomb</td>
<td>128</td>
<td>15</td>
</tr>
<tr>
<td>Rohacell 110 foam core</td>
<td>110</td>
<td>3</td>
</tr>
</tbody>
</table>

Table 4-3, The property of several cores

<table>
<thead>
<tr>
<th></th>
<th>density (kg/m³)</th>
<th>tensile strength (MPa)</th>
<th>compressive strength (MPa)</th>
<th>shear strength (MPa)</th>
<th>shear modulus (MPa)</th>
</tr>
</thead>
<tbody>
<tr>
<td>end grain balsa wood</td>
<td>150</td>
<td>13.1</td>
<td>12.9</td>
<td>3</td>
<td>159</td>
</tr>
<tr>
<td>HRH-10-1/8-9.0 honeycomb</td>
<td>144</td>
<td>-</td>
<td>15.2</td>
<td>3.1 (L), 1.8 (T)</td>
<td>138 (L), 80 (T)</td>
</tr>
<tr>
<td>Rohacell 110 foam**</td>
<td>110</td>
<td>3.5</td>
<td>3.0</td>
<td>2.2</td>
<td>50</td>
</tr>
<tr>
<td>Rohacell 200 foam**</td>
<td>205</td>
<td>6.8</td>
<td>9</td>
<td>5</td>
<td>150</td>
</tr>
</tbody>
</table>

* from reference [15]

** from manufacturer, and L is longitudinal and T is transverse

Figure 4-15, Compression test set-up.
4.4 Corrosion Test

The corrosion behaviour of the metallic floor is important in the bulk cargo floor. Problems associated with corrosion of the cargo floor are coming from sea food transportation, environment, lack of corrosion protection, damaging the corrosion protection layer due to puncture or scratching during the cargo loading and unloading. Corrosion will reduce the strength of the panel considerably and frequent replacements or repairs of the panel will be necessary. Therefore, the material selection and the surface treatment for the facing surfaces are the significant steps of the panel design.

Most corrosion problems in the bulk cargo floor are surface corrosion, pitting corrosion and direct chemical attack. Therefore, a good surface treatment is recommended to be considered since the beginning of the design.

The present experiments have been conducted on specimens with impact damage. The purpose is to determine the corrosion effect of a damaged panel with various types of top faces and core materials. To avoid corrosion from the edges of the specimens, all edges were protected with paraffin.

The corrosion testing is conducted in a salt spray chamber in accordance with ASTM B 117-73 part 10 [8]. The purpose of using the salt spray chamber is to accelerate the corrosion process. The test is carried out for a minimum duration of 1500 hours that the minimum time in the ASTM standard is 100 hours. The temperature of the chamber is chosen to be 40 °C.

4.4.1 Test Results and Discussion of The Corrosion Test

The panel made of bare Al 2024 T3 with a balsa core was completely corroded. The surface corrosion and pitting corrosion occurred on the entire surface of the panel as shown in Figure 4-16. This test represents damage of the surface protection layer, if the top face is made of bare material with an anodizing layer. The anodizing layer is very thin and can easily be damaged during loading and unloading of the bulk cargo. The corrosion damage of these specimens is not repairable and must be replaced.

The panel made of the clad Al 2024 T3 0.8 mm as the top face with a balsa or a nomex core showed surface corrosion and pitting corrosion at the impacted area. The surface corrosion can be indicated by a decoloration of the corroded area as shown in Figure 4-17. The pitting corrosion is caused by damaging of the oxide layer due to the impact process. A more severe corrosion at the impacted area was found on the specimen with crack. The corrosion started from the cracked area and then continued to grow around it. The corrosion of both balsa and nomex core was very limited just underneath the damaged area where the crack was present. Corrosion or delamination of the bond-line area between the facing and the core was not discovered during the experiment. The condition of the panel with a balsa or the nomex core is repairable.
Figure 4-16, Surface condition of bare Al 2024 T3 panel specimen after exposure in the salt spray chamber.

The corrosion of the Glare panel was negligible. It only occurred in a very limited area or just around the cracked area. This good corrosion behavior of the Glare panel is mainly due to the good surface protection employed on the Glare sheet. There is no corrosion discovered underneath the facing surface for both types of core material when the crack is not present. The surface condition of a Glare panel after exposure in the salt spray chamber remains unchanged. For the uncracked specimen, repair is not necessary until the overhaul of the aircraft. Figure 4-18 shows the surface condition of a Glare panel after the exposed in the salt spray chamber.

Figure 4-17, Surface condition of clad Al 2024 T3 0.8 mm panel after exposure in the salt spray chamber.

The balsa core resulted in an excellent corrosion behaviour as long as no debonding has occurred and a good surface treatment is applied to the facing surfaces. It
confirmed that balsa as a core material has an excellent corrosion behavior for the bulk cargo floor application.

![Before](image1.png) ![After](image2.png)

Figure 4-18, Surface conditions of Glare panel after exposure in the salt spray chamber.

4.5 Service Evaluation Test of a Glare Panel for the MD-11 Aircraft and Glare stiffeners for the Fokker F-28 Mk-4000

As a result of extensive testing in the Structures and Materials Laboratory of the Faculty Aerospace Engineering of the Delft University of Technology laboratory, a new floor panel has been developed, which has good impact properties and a good corrosion behavior. This floor panel consists of Glare 5 2/1 as the top face, balsa wood as the core and clad Al 2024 T3 0.3 mm as the bottom face. The test results showed that Glare as the top face was always resulting in the best impact property. Balsa wood was the best core material. Such a floor panel has been tested for service evaluation in the container cargo floor of an MD-11 of the Garuda Indonesia Airlines.

The panel was installed in the container cargo floor at the door area. The edges of the panel were protected with the epoxy resin to avoid water absorption. After the installation, the gap between the floor and frame was filled with a silicon rubber to prevent the water trapped in this gap. Figure 4-19 shows the installation of the Glare floor panel on the MD-11.

The result was satisfactory and increased the life of the panel about twice, as compared to the original floor. The condition of this floor panel when removed from the aircraft is still good and there is no corrosion or puncture found on this floor.
Figure 4-19, Installation of Glare floor in MD-11.

Figure 4-20 shows the installation of Glare floor stiffener in the bulk cargo floor of Fokker F-28 MK 4000 at the door area. This experiment is to determine the effect of the corrosion and impact damages to the Glare structures in actual condition. These 4 Glare stiffeners are still flying for more than 3 years, where the life of the aluminum stiffener is among 1 to 2 years. Therefore the advantages of Glare in corrosion behaviour and impact resistance would significantly have a consequence in cost saving.

Figure 4-20, Glare stiffener in bulk cargo of Fokker F-28 MK4000
4.6 References


[7] ASTM C 393-62, Flexure Test of Flat Sandwich Construction


5. CONCLUSIONS

Impact Test

1. Glare sheet as a top face resulted the highest impact resistance of in floor panels compared to 2024 T3 Alclad sheet material.

2. Balsa as a core material resulted the highest impact resistance of floor panels.

3. The failure energy of rigdly support panel is much lower than for a panel with an open circular support. In the rigid support, the bottom face is not affected by the impact process, but in the open circular support, the bottom face is significantly affected.

4. A foam core with a comparable density to the nomex core, but much less compression strength, the failure energy is lower than for balsa or nomex cores.

Bending Test

1. The bonding quality between the facings and core has significant effect on the strength of a panel.

2. In short beam bending, it shows that the shear properties are important.

Corrosion Test

1. There are two types of corrosion found in the corrosion tests, that are surface corrosion and pitting corrosion.

2. A balsa core resulted in an excellent corrosion behaviour of a floor panel.

3. The chromic acid anodizing and the BR 127 primer as the surface treatment on Glare sheet gives excellent corrosion protection on the floor panel. There is no delamination between the facings and core due to the bond-line corrosion for all types of cores during the corrosion test.

Service Evaluation Test

1. The combination of Glare 5 2/1 .3 as the to face with a the balsa core and Al 2024 T3 0.3 mm as the bottom face in floor panel gives the highest impact property.
This floor configuration had been tested for in service evaluation in the container cargo floor of MD 11. It gives a significant improvement. The life of a new floor panel was improved about twice as compared to the aluminum floor (original floor). When removed from aircraft, is still good, which is only with a minor scratch. The damages are not necessary to repair until the next C check interval (+ one year period).

2. There is no corrosion found in this floor panel, after a period about 2 years. It shows that the surface treatment applied on the floor panel is satisfactory.

3. The reliability of a new floor was satisfy the airline, it means that Glare floor panel can be considered to replace the aluminum floor.

4. The Glare floor stiffeners in the bulk cargo floor of Fokker F-28 result a significant improvement as compare to the aluminum stiffener. The life of aluminum stiffeners at the door area are between 1 to 2 years and the Glare stiffeners are still flying for more than 3 years.

Table 5-1 is summary of the results of the floor panel test programs. In this table shows the performance of the top face and core material with respect to the impact, bending compression and corrosion in floor panel application.

Table 5-1, Summary of the results of the floor panel test programs for top face and core material

<table>
<thead>
<tr>
<th>Type of test</th>
<th>Impact</th>
<th>Bending</th>
<th>Compression</th>
<th>Corrosion</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Top face</strong></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Alclad 2024 T3</td>
<td>Fair</td>
<td>-</td>
<td>Not affected</td>
<td>Fair</td>
</tr>
<tr>
<td>Bare 2024 T3</td>
<td>Fair</td>
<td>-</td>
<td>Not affected</td>
<td>Bad</td>
</tr>
<tr>
<td>Glare 5 2/1</td>
<td>Good</td>
<td>-</td>
<td>Not affected</td>
<td>Good</td>
</tr>
<tr>
<td><strong>Core</strong></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Balsa</td>
<td>Good</td>
<td>Fair</td>
<td>Fair</td>
<td>Fair</td>
</tr>
<tr>
<td>Nomex</td>
<td>Fair</td>
<td>Not tested</td>
<td>Best</td>
<td>Best</td>
</tr>
<tr>
<td>Foam</td>
<td>Bad</td>
<td>Not tested</td>
<td>Bad</td>
<td>Best</td>
</tr>
</tbody>
</table>
Part 3: GLARE FOR A NEWLY DESIGNED AIRCRAFT STRUCTURE

1. INTRODUCTION

1.1. References

It is known that the step by step development in aircraft maintenance at the present, is a great support in aircraft design, with direct involvement of the operators, e.g., Boeing 777, MD-11, etc. A combination of experience in aircraft design and maintenance, will result in best aircraft, which meets the operator requirements. This part discusses about the aircraft design with respect to the maintenance issue.

As we look into the history of aircraft structures, since the Wright Brothers could make their first successful aircraft, that aircraft was made of wooden structure. The introduction of aluminum alloy as a substitution for the wooden structures, has changed the whole character and appearance of the aircraft. Since that time it has made a major jump in the aircraft design. The changes from wooden structures to aluminum structures introduced a new problem in the design and operation, which is fatigue corrosion. The next stride was the applications of the composite material, which is lighter, stronger, insensitive to the fatigue good corrosion resistance. The use of the composite material in aircraft structures, again introduced new problems, i.e., delamination, water absorption, chemical attack, impact damage, new inspection methods, environmental deterioration, etc. References [1] mentioned disadvantages of the composites as compared with the aluminum structures, such as material handling, delamination, etc.

Since maintenance cost rise up from 3 to 4 times the aircraft purchase price during the lifetime of the aircraft, the operator inputs become an important issue in relation to the aircraft design. The objective of this part is to provide information for the designer on the importance of the maintenance issue and operation aspects in aircraft design.

The effect of the maintenance issue in the material selection is discussed in the next chapter with respect to the issues arising out Maintenance steering Group 3 (MSG 3). Material selection is a critical process in aircraft design. Failures at this stage, may affect the operational safety and economy, e.g. section 41 modification of Boeing 747. This modification was taken a huge man-hours and aircraft down time because it mandated by the authorities. It means that a malfunction these structures may have an
adverse affect the safety of the aircraft. The reason of section 41 modification is a fatigue on the fuselage frames due to the use of Al 7075. It was then changed with an Al 2024 T3.

Maintenance and operation cost aspects are frequently overlooked in the design stage, which lead to inescapable problems for the operator. It is well known by the aircraft industries and airlines, that cost savings in maintenance and operation much depend on the initial design of the aircraft. Therefore the most economical way in reducing the maintenance cost, is to incorporate the maintenance issue from the beginning of the design. Extensive studies about maintenance and operation cost savings in new aircrafts have been carried out in this part, which also includes the fuel savings due to the use of Glare on the fuselage skin of Airbus A 3XX.
1.1. References


2. TECHNICAL MAINTENANCE ISSUE

2.1 An Overview of Maintenance Steering Group 3 (MSG 3)

In the past, the maintenance program was developed by the aircraft manufacturer and approved by the Regulatory Authority of the country of manufacture. All maintenance tasks were classified as the Hard Time (HT). HT is a primary maintenance process, under which an item is removed from service for reconditioning, or heavy maintenance or bench checked before a previously specified time [15]. The introduction of a new technology and new material; then the reliability of the aircraft systems and structural parts becomes excel lent. Throughout the years, the airlines learned that many types of failure could not be prevented by such maintenance activities (HT).

Subsequently, in 1968, the Air Transport Association of America (ATA) developed the Maintenance Steering Group 1 (MSG 1) for the Boeing 747 aircraft [15,16]. Afterwards it was decided that experience gained on this project, shall be applied to update the logic decision and to delete some B 747 detailed information. So a universal document could be made applicable for later new types of aircrafts. This work was done and resulted in a document, MSG 2. It was used to develop the aircraft maintenance programs since 1970. Then the next step was the upgrading from the MSG 2 to MSG 3. There were no big differences between them, whereas the MSG 2, was process oriented and the MSG 3 was task oriented and the incorporation of the damage tolerance rule.

Since the introduction of the MSG 1, the maintenance program was developed together among the airlines, manufacturers and the authorities. As a result, the new
aircraft maintenance program which is developed with this logic decision process, shows easier to maintain and cheaper than the old generation of the aircrafts.

2.2 Aircraft Maintenance Program Development

A maintenance program is necessary to be developed for each new aircraft type prior to its introduction into service. This program is mandated by government order. It is a requirement for a new aircraft in obtaining the Type Certificate (TC). The purpose of the program is to develop a Maintenance Planing Proposal (MPP) to assist the regulatory authority in establishing an initial scheduled maintenance for a new aircraft type. This program also uses to maintain the inherent safety and reliability level of the aircraft. The result is a Maintenance Review Board document (MRB document). The MRB document must be used as an initial to develop the airline maintenance program which is tailored for each airline. The program consists of the aircraft system, power-plant, avionics and electrical, and the aircraft structure. The tasks may include:

1. Lubrication or servicing (LU/SV)
2. Operational check/ Functional check (OP/FC)
3. Trough visual check/ General visual check (TVC/GVC)
4. Restoration (RS)
5. Discard (DS)

Once the aircraft enters into service, the initial maintenance program is subject to the continuous development and update in regard with the modification and experience. The environmental effects and the maintenance experiences are the significant factors within the maintenance program, but they are not in detail discussed in MRB documents for the specific operator. Therefore the procedure for updating has to be included in the maintenance program by the airline itself.

The maintenance program can be divided into two major groups. The first group consists of the aircraft system, power-plant, avionics and electrical, called Maintenance Significant Items (MSIs). The second group consists of the structural items, called Structural Significant Items (SSIs), [16]. The MSI and SSI are identified by the aircraft manufacture using its engineering judgement in a conservative way and based on the anticipated consequences of failure. While the MSIs are those items whose failures are:

1. could affect safety (on the ground or in flight), and /
2. could be undetectable or are not likely to be detected during their operation, and /
3. could have a significant operational impact, and /
4. could have a significant economical impact.

A SSI is “any detail, element or assembly, which contributes significantly to carrying flight, ground, pressure or control loads and whose failure could affect the structural integrity necessary for the safety of the aircraft” [16]. These two groups are described in the MRB document and they have been mandated by the government order to be included in the initial Airline Maintenance program as the minimum requirement.
This section discusses the relation between the structural maintenance program and the material selection process in aircraft design, especially for the fuselage structure. The aircraft structure is defined for all load carrying members including wings, fuselage, engine mounting, etc.

The aim of this structure maintenance program analysis, is to look systematically at the structural elements of the aircraft to determine its significance in terms of the structural function. Therefore the structure elements (SSI items) will be assessed for its susceptibility in detecting them for three basic forms of the structural deterioration, namely, the Accidental Damage (AD), Environmental Deterioration (ED) and Fatigue Damage (FD), [23]. The inspection programs are required to ensure that any structural deterioration can be detected before it affects the structural integrity of the aircraft.

The SSI analysis procedure can be divided in the following stages:

1. The preparation of the SSI selection list and description,
2. determination of possible sources of the structural deterioration,
3. selection of the inspection requirements per SSI (inspection threshold, repeat inspection interval, method, and equipment),
4. overall consolidation of inspection requirements, and
5. preparation of the maintenance task description, where required, to initiate task incorporation with the aircraft structure repair manual or with non-destructive testing manual.

2.3 Process of Material Selection in Aircraft Design

As mentioned above, material selection is an important steps in aircraft design. The process described in this section is a result of compromised of various considerations, which is obtained from the discussions among the aircraft manufactures (IPTN, Fokker, and Airbus Industries), maintenance issues derived from MSG 3, reference [19] and from experiences. The process of the material selection could be consisted of a group of questions with respect to aircraft design, production, maintenance, operation, etc. These questions can be divided into five groups as follows:

1. Commercial aspect,
2. material properties,
3. material processing and production,
4. operational related,
5. maintenance related.

The last two group of question numbers 4 (operational related) and 5 (maintenance related) are most derived from the MSG 3 and the Structure Repair Manual (SRM) of the existing aircraft.

Furthermore these five groups can be explained as follows:
ad 1. The commercial aspects; The aspects are not just being able to obtain the material on a commercial basis, but it shall also be available in many forms and types (e.g. Al clad 2024 T3, bare 2024 T3, etc.). The availability of the materials will also depend on the production capacity and industries-wide demand. In this case, the investment cost should also be included in the calculation, when the new material is selected. The factors may affect in the material selection are:

a) The availability and producibility,
b) the material cost,
c) the fabrication characteristics, and
d) the materials handling cost.

ad 2. Material properties; For the calculation purposes, material properties are required to determine static strength, fatigue performance, residual strength, etc. To support the calculation in aircraft design, material properties should be provided and carefully analysed before it is decided to be used. The minimum properties should be provided with respect to the material selection are:

a) The static strength efficiency,
b) fatigue damage behaviour,
c) fracture toughness and crack growth,
d) corrosion and embrittlement,
e) environmental stability (ultraviolet, aging, etc.), and
f) impact damage.

ad 3. Material processing and production in aircraft manufacture; Material processing and production are to be considered, when a decision involving the use of a new material has been taken. The method used to analyze the material processing and production is based on the complexity factor. This factor expresses the relative effort required to perform such process that is needed to produce this part. The questions related with this issue are as follows:

a) Are the existing material processes applicable (i.e. cleaning process, surface treatment, etc.)?,
   b) are the existing machinery applicable (i.e. drilling, forming, cutting, etc.)?,
   c) are the existing production procedures applicable?,
   d) do the existing technician need to be trained?,
   e) are the existing inspection methods applicable?,
   f) are the level of the rejected parts low?, and
   g) are the existing quality control procedures applicable?.

ad 4. Operations related; Problems of the operational related could be derived from the MSG 3 logic decision process which frequently occurred in service. In the new aircraft, the inspection threshold and repeat inspection intervals must be defined to the extent of the source of damage. There are three basic sources of damage which could initiate structural problems in operations that should be taken into account during the material selection, which are:
a) Accidental damage; The inspection threshold for accidental damage which normally corresponds to the period when the accidental damage can be inspected before it is extended to the critical situation. This inspection threshold is a result of a discrete event which may occur at any time during the life of the aircraft. Therefore the system to determine the inspection interval has to be developed for each SSI which is considered for susceptibility to the accidental damage. There are two characteristics that can be used to determine the inspection interval:

- The frequency of exposure to the possible source of the accidental damage. This rating is basically as a function of the location in the aircraft (e.g. door area is highly susceptible for the accidental damage),
- the effect of such damage on the SSI, i.e. it could lead to subsequent fatigue, delamination or corrosion damage.

These two above ratings should be combined into a single rating and translates into an inspection interval. The problems which are related to the operational are:

- Resistance to Foreign Object Damage (FOD),
- in service FOD limit,
- how to detect or inspect the FOD and what is the dent limitation?,
- does lightning strike occur,
- how to inspect for lightning strike and what is the limitation?,
- can delamination occur,
- when and where do delamination occur,
- how to inspect delamination and what is the limitation?,
- does erosion has significant effects,
- what is limitation of the erosion?,
- do cold and hot soak have significant effects,
- how to inspect the damage of cold and hot soak,
- etc.

b) Environmental Damage (ED); This inspection threshold which mostly depends on the corrosion behaviour of the material and the corrosion protection level applied on the parts, e.g., Al 7079 has a lower inspection threshold as compared to the Al 2024 due to its corrosion behaviour. The environmental deterioration should be evaluated in determining its susceptibility to and timely detection of the corrosion and stress corrosion.

The occurrence and spread of the corrosion is not something that can be calculated, therefore there is no quantitative background data as a basis for determining the inspection interval. In this case, the choice of interval becomes subjective and leads to a wide variety. The susceptibility to corrosion should be assessed on the basis of probable exposure to an adverse environment and an adequacy of the protection system. Material characteristics, coupled with the likelihood of sustained tensile stress, are used as a basis to determine the susceptibility to the stress corrosion. The
time of detection is determined by using the sensitivity to a relative size of damage with the visibility of the SSI for inspection, or called a rating system. This rating system is needed in order to ensure the degree of consistency in determining the inspection interval. Again, the three rating above of material type, protection system and area (e.g. lavatory, galley, etc.) should be combined into a single rating which determined the inspection interval ED.

c) Fatigue damage; The inspections related to fatigue damage, where fatigue damage will occur after a threshold. This threshold should be provided by the aircraft manufacture and approved by the regulatory authority. This inspection threshold and repeat inspection interval, must lead to an inspection program which provides a high probability of detecting fatigue damage for a world wide fleet, before it reduces the residual strength below the allowable levels. To achieve the high level of safety, the items should be considered as follows:

- the residual strength which includes the effects of the multiple site fatigue damage, and
- the crack growth rate which includes the effects of the multiple site fatigue damage.

The damage detection period which corresponds with the fatigue damage may be detected by the NDI. This period varied in accordance with the method to be used and may also be influenced by such accessibility, viewing distance, etc.

The fatigue inspection threshold and the repeat inspection interval, should be checked with the maintenance schedule, to ensure the damage tolerant requirement. If the structural part is considered as the critical item, this item shall be classified as Airworthiness Limitation Items (ALI's). ALI, is the item when failure has a directly adverse effect on the safety of the aircraft. These items are usually mentioned in chapter 5 of the maintenance manual of each aircraft type.

ad 5. Maintenance related; Since introducing the MSG 1, the maintenance aspect becomes important in the material selection during the design of the aircraft. The capability of the material in maintenance must be considered in this process, otherwise this aircraft could lead to difficulties in maintaining its airworthiness. From the data of aging aircraft programs and maintenance experiences, the problems which frequently occurred in maintenance, can be summarised as follows:

a) Reparability, i.e. worked out of the skin for corrosion or accidental damage,
b) susceptibility to chemicals, such as paint stripping, skydroll, fuel, oil, etc.,
c) possibility of polishing for fuselage skin,
d) delamination if effective; how to inspect and repair,
e) do the existing equipments applicable,
f) do the maintenance personnel need to be trained, and
g) do the existing inspection procedures applicable.

The process of the material selection above can be presented in the flow chart, as shown in Figure 2-1 with regard to the level of importance on each step.

A study of the effect of the application Glare sheet for aircraft structure in the rating system of MSG 3 is discussed in appendix D. The impact of Glare in the rating system is significant in view of the inspection threshold and repeat inspection, especially in relation with other structural parts which have different in life, e.g., aluminum parts, composite parts, etc.
2.4 References


3. GLARE AND ECONOMY

3.1 Maintenance Cost Savings due to the Use of Glare

Since the mid 1980’s, materials, production engineers, and design engineers were forced to design for cost reduction instead of reducing weight. The competition within aircraft manufacturing was forced by their engineers, to produce excellence of their design in terms of competitiveness, performance (weight saving), low maintenance cost and reliability.

The major obstacle of introducing a new material in aircraft structure for the transport category aircraft is the cost. During the early days of flights of the Wright Brothers, the need to reduce the weight, was driven by the limit of the capabilities of the power plant. Since 1970, the increase of the fuel price pushed much forward the performance of the aircraft (e.g. fuel consumption, reliability, etc.) as the main criterion. Politic, economic, social environment and the airline business strategy, changed in the 1990’s, which had a significant impact on the civil aircraft industries. The market for passenger and freight increased, and the fares dramatically reduced. It caused world competition for producing the aircraft, since this time, the pricing policy given by International Air Transport Association (IATA) was no longer a guideline for the airlines to determine their air-fare; where the actual air-fare was far below the IATA standard. This situation forced the aircraft manufacturers to produce the aircraft cheaper which will be less maintenance cost, that means less inspection, less repair, less modification, and simple to operate.

The early B 747-100, the letter check intervals were an A check of 50 hours, B check of 200 hours, C check of 800 hours and the structural inspection/ D check of 9,000 hours. During the life of the aircraft, escalation of the programs of the letter check intervals were made. Today, the average operators are flying with an A check of 330 hours, C check 4,000 hours and a structural inspection/ D check of 22,000 hours, while this extension is mainly due to experiences and some improvements through modifications. Furthermore, with a mature design and maintenance, many operators of B 747-400 are now finding it more economical to start their initial structural inspection interval in 30,000 hours, and subsequently followed by 3,000 - 5,000 hours inspection intervals. This example implies, that the improvement in structural inspection intervals has a considerable effect in reducing the maintenance cost.
Airframe maintenance is the major task in aircraft overhaul and contribute to about 8 percent in direct operating cost. Therefore, the improvement, however, will then be significant in cost saving. Figure 3-1 shows the MD 80 maintenance cost breakdown, reference [14].

![Pie chart showing maintenance cost breakdown of MD 80](image)

Figure 3-1, Direct operating cost of MD 80

Fatigue inspection intervals very much depend on the crack initiation, detectable by NDT (inspection threshold) and the critical crack length. As demonstrated by the test results in chapter 6 of part 1. Crack initiation of Glare repair is significantly longer as compared to aluminum repair. Figure 3-2 shows the influence of the repair materials on aluminum skin repair. Müller [20] found that Glare lap joint has better fatigue life than aluminum. Glare sheet for aircraft structure may result in a longer inspection interval, and consequently it reduces the maintenance cost significantly. The inspection of Glare skin can be performed the same manner as for aluminum structure, e.g. Eddy Current, Ultrasonic, etc.

An extensive incorporation of the new state of the art of Glare sheet for fuselage skin on new aircraft, will be accompanied by an improvement in operational benefit and maintenance opportunity trough:

1. The operational characteristics of this new material has been improved substantially over the monolithic aluminum by better fatigue behaviour, impact resistance and corrosion properties. These three types of damages are the major obstacle in the operation and maintenance as shown in material selection of chapter 2 in this part. The benefits of Glare sheet over aluminum alloy, could significantly extend the inspection threshold and the repeat inspection intervals.
2. Maintenance cost reduction could be reduced by less man-hours for repair and inspection due to the improvement of the material properties. An investigation in maintenance of Glare structure with regard to the questions mentioned in chapter 2 of the part 3 [4], has shown that the methods, equipment, inspection procedures, skills, etc., are similar to the aluminum structure [4,7]. Therefore, the use of Glare sheet for fuselage skin in new aircraft design does not require new investments, e.g. training, tooling, etc.

Aircraft corrosion at increasing aircraft age becomes more widespread and it is more likely to occur concurrently with other form of damages, such as fatigue cracking. One of the important issues about corrosion, is pitting corrosion. This type of corrosion when it occurred in aluminum can penetrate through the thickness, but it will not be presented in Glare sheet. If a pitting corrosion presented on one layer of the aluminum, it will not penetrate the aluminum layer below, because the glass layer in the Glare sheet will prevent the corrosion to continue. As a result, Glare fuselage will have a longer inspection threshold, longer repeat inspection, and less damage caused by pitting corrosion (Environmental Damage/ED). Figure 3-3 shows the cross section of fiber metal laminate and aluminum after exposed in salt spray chamber for corrosion test. In reference [11] and the experiences of Garuda Indonesia, show that corrosion causes about 70 to 80 percent of repairs in overhaul.

Figure 3-2, Influences of different doubler materials in fatigue life, see chapter 6 of part 1.
To give an overview of the benefits of the usage of Glare sheet in aircraft structures, this section shows the calculation of the man-hours of structural parts. The calculations are based on the man-hours needed for inspection in the fuselage skin of MD-11 aircraft during their lifetimes. The reason of using MD-11 is due to the tasks in the Maintenance Review Board (MRB) document, which have been developed considering of the three damage sources (fatigue, environmental and accidental damages) and it becomes more simple. The calculations of the man-hours are base on the MRB document issued by the FAA [2], while the predictions of the man-hours for Glare fuselage skin of MD-11 are based on the test data done in the laboratory and from some references [3, 4, 7, 19]. Therefore some assumptions should be made in order to simplify the calculation of the maintenance man-hours for Glare fuselage skin, as follows:

1. The fatigue life of the Glare fuselage is 1.5 times longer than aluminum (Table 6-9 chapter 6 of part 1),
2. The man-hours for corrosion inspection of Glare fuselage are about 10 percent less than the aluminum due to its good corrosion behavior, and
3. The accidental damage threshold of Glare fuselage is 1.4 longer than the aluminum due to its good impact property.

The calculation considering the items which are only related to the fuselage skin inspection taken from the structural maintenance program (MRB document) and the zonal inspection program. Table 3-1 shows the sample table used to calculate the inspection-hours of the fuselage skin. The results of the man-hours calculations for aluminum and Glare fuselage are shown in Figure 3-4. From this calculation can be understood that the major check interval have significant effect in the man-hours. For example, the inspection threshold of Glare fuselage for corrosion is 7.5 years and the repeat inspection is 7.5 years, but the major check interval of this aircraft is 5 years. In this case, the inspection for the corrosion of Glare fuselage must be also carried out in five years instead of 7.5 years, thus these advantages will not be significant. Therefore, the major check of Glare structures should be accomplished in 7.5 years inspection interval to maximise the benefit of Glare structures.
Table 3-1, A sample table for inspection-hours calculation for aluminum fuselage and Glare fuselage.

| SSI items or Zonal ins-prog | Reason | Type of insp | Zonal | man-hours | | --- | --- | --- | --- | --- | --- | --- |
| --- | --- | --- | --- | --- | --- | --- |
| 53.10.04/01 | AD | GVI | 550, ... | 2 | . | . | . |
| /02 | ED | DI | 261, ... | 60 | . | . | . |
| /03 | FD | DI | 221 | 10 | . | . | . |
| Total man-hours | . | . | . | . | 2300 | . | . | . |

Note, - AD is Accidental Damage, ED is Fatigue Damage and FD is Fatigue Damage.
- GVI is general visual check and DI is detail inspection.

Figure 3-4, Expected benefit of Glare fuselage in direct man-hours of fuselage skin inspection.

The average saving per major check visit of the inspection-hours of Glare structures will be US $ 8750.
3.2 Fuel Savings Due to Weight Reduction

Since the airframe design should be traded off for weight and cost to meet the overall objective, then the relationship between them might be known. In large aircraft, weight is the important issue. Increasing the size of aircraft will also increase the weight of aircraft significantly, as a consequence, the aircraft payload and range will reduce. The use of Glare for fuselage skin is able to reduce the fuselage weight about 15-30 percent as mentioned in reference [4] due to its lower density. This weight reduction can be translated into two advantages, which is more payload or less fuel burn.

This section discusses the advantage of the fuel burn reduction due to weight reduction of Glare structure. This fuel burn reduction can be determined by using the Berguet formula. The calculation is based on data as given by the existing aircraft, e.g., Boeing 747-100. To simplify the calculation of the fuel burn reduction, hence some assumptions should be made as follows:

1. The flight profile of both aircraft (aluminum fuselage and Glare fuselage) remain the same,
2. the aircraft has the same range as before,
3. the center gravity remains the same,
4. the specific fuel consumption of both aircrafts remains constant,
5. \( W_m \) is the average weight of Maximum Take-Off Weight (MTOW) and Maximum Zero Fuel Weight (MZFW), and
6. the lift coefficient can be assumed:

\[
C_L = \frac{W_{av} \cdot g}{q \cdot S} \tag{3-1}
\]

Where:

\( g \) is dynamic pressure at cruise altitude \( \sim 0.5 \rho V^2 \)
\( \rho \) is air density
\( S \) is wing area, and
\( q \) is acceleration of gravity.

Then the fuel burn reduction from reference [5] can be calculated as follows:

\[
\ln \left( \frac{W_{1c}}{W_{2c}} \right) = \left( \frac{C_L}{C_D} \right)_{c} \ln \left[ \frac{\% W_{1c}}{\% W_{1c} - \% W_{Fx}} \right] \tag{3-2}
\]
Where:

index 1 and 2 indicate the MTOW and MZFW respectively,
index c and x indicate the current design and the new design respectively, and
$C_D$ can be determined as a function of $C_L$:

$$C_D = 0.0176 + \frac{(C_L - 0.1430)^2}{11.0}$$

3-3

Assuming that fuel burn of the B 747-200 due to the use of Glare in fuselage skin is reduced the MZFW about 0.7 percent (± 1690 kg), [5]. Table 3-2 shows the parameter of B 747-100 used in this calculation. Considering this assumption, the reduction of the fuel burn that can be obtained about 0.511 percent or equal to 750 kg per flight (12 hours flying time). For this large aircraft, the flight cycles is 400 per year with the flight duration average 12 hours per day, then the saving for the fuel burn will be $1.8$ million (25 years) or $75,000$ per year (fuel price $0.25/\text{kg}$).

Table 3-2, Data of B 747-100 (type certificate data sheets)

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Magnitude</th>
</tr>
</thead>
<tbody>
<tr>
<td>MTOW</td>
<td>362,875 kg</td>
</tr>
<tr>
<td>MZFW</td>
<td>242,670 kg</td>
</tr>
<tr>
<td>Wing span</td>
<td>64.44 m</td>
</tr>
<tr>
<td>Cruise altitude</td>
<td>10,000 m ($\rho = 0.4127 \text{ kg/ m}^3$)</td>
</tr>
<tr>
<td>Cruise speed</td>
<td>0.86 M = 927 km/ hr</td>
</tr>
</tbody>
</table>
3.3 References


4. CONCLUSIONS

1. A Glare sheet for fuselage skin in new aircraft (designed) results in a significant maintenance cost reduction offer the aluminum alloy.

2. The production technique and inspection method of Glare can be carried out in the same manner as for monolithic aluminum.

3. The study of the application of Glare sheet for fuselage skin of MD-11 shows significant saving in inspection-hours. This benefit in inspection is mainly due to a good corrosion behavior Glare sheet. Most inspection tasks in the structure maintenance program and zonal inspection of MD-11 are the corrosion.

4. The low density and fatigue insensitive of Glare can result in a lighter aircraft, meaning a lower fuel consumption as compared to aluminum fuselage. A study of Glare fuselage for B 747-100 can save the fuel burn to about US $ 1.8 million (per 25 years) or US $ 75,000 (per year) with the fuel price of US $ 0.25 per kg.
Part 4: THE INFLUENCE OF GLARE SHEET IN REPAIR, ON THE LIFE CYCLE COST

1. INTRODUCTION

1.1. References

In order to choose the best capital investment in aircraft repairs, the field engineer requires a number of available repair alternatives that can be evaluated using an appropriate method for comparison. Since the alternatives are different in terms of costs, benefits, and timing, the basis of comparisons shall take into account these differences as the time value of money. There are several methods that may be used for comparisons that are the Net Present Value (NPV), Internal Rate of Return (IRR), Capital Recovery with Return, Life Cycle Cost, Break-even Analysis, etc. The choice of the comparison method depends on the company policy, the complexity of the problem and the required accuracy. In general, literature surveys indicate that the preferred method for repair cost comparison is the Life Cycle Cost (LCC). This method is suitable in comparing alternatives for the airline engineer and does not need too much data which some times is difficult.

Most of the repairs designed today are based on mechanical properties consideration, without too much concern about the economical aspect. In view of the fact that the economical aspects and mechanical aspects are important issues to be considered in aircraft repair. The recent combination of economic trends, rising inflation, cost growth experience, safety, the continuing reduction of buying power, budget limitations, increased competition, and so on, has created an awareness and interest in analyzing a method for total system and production cost. Not only are the acquisition costs associated with the accomplishment of the repair rising, but the costs for inspection and maintenance are also increasing at alarming rates. This is due primarily to the safety and economy.

While obtaining more knowledge about the advantages of using Glare as a repair material over the monolithic aluminum as a repair material, the economical aspect should also be taken into account to determine its advantage. The superiority of a Glare repair over the monolithic aluminum should also be translated into its economical value.

The aims of this part are threefold, i. e.:
INTRODUCTION

- Introducing economical considerations of repair in maintenance,
- to describe a procedure that may be used to compare several repair proposals through their economical value,
- to compare the cost over the life span of a repair, e.g. Glare repair versus aluminum repair, and

This economical evaluation will show that the application of Glare in aircraft repair will give a significant benefit. The standard repair procedure on existing aircraft with an aluminum repair is used as a basis for comparison. Furthermore the LCC of all repair proposals will be compared to the LCC of the basic repairs, then the most economical repair proposal can be selected.
1.1. References


2. LIFE CYCLE COST

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2.1 Cost Breakdown Structure ...................................................................................... 221
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This chapter introduces a procedure for the airline engineer to analyse repair proposals involving economical benefits, to embrace all phases of the system life cycle, i.e. repair design, production, operational support, inspection and repair (maintenance). The use of life cycle cost approach as a tool for the airline engineer, in comparing several repair alternatives that can enhance the economic competitiveness in aircraft maintenance.

2.1 Cost Breakdown Structure

Airline maintenance cost breakdowns, are depending on the purpose for which they will be used. In planning of the aircraft maintenance, the following aspects of costs are required:

1. A general breakdown of costs; to show the cost trends over a period of time that measure the cost effectiveness of particular areas such as direct operating costs, maintenance efficiency, etc.,
2. assessment of costs; is essential in any evaluation of investments, whether for new aircrafts or for aircraft repairs, and
3. cost identification; is a crucial task within the maintenance planing to support the policies over the individual cost driver in aircraft repairs.

Cost classifications of repair have come into existence to be used as a basis to determine the life cycle cost and economic analysis. These classifications are useful to collect all the sources and effects of costs that may have a bearing in repairs.

A Cost Breakdown Structure (CBS) is an other way to classify the cost elements of repairs. CBS can be used as a basis for assessing the life cycle cost of each alternative being considered. The procedure should be simple for the airline engineer to do this analysis due to the limited information and time available in field.

The CBS should be linked with the objectives and resources. It should be constituted the cost as function of activity, major element of task and more discrete classes for common or like items. This CBS should be tailored for specific type of systems being evaluated, therefore it should exhibit the following characteristics:
1. Life cycle cost should be considered and identified in the cost breakdown structure, in order to determine all cost during the repair (acquisition) and the operation phase,
2. the cost should be broken down into a level which for the airline engineer could be easier to find the data, and
3. the man-hours, materials, inspections and maintenance costs should be broken down for each repair alternative.

Figure 2-1 shows the CBS of a repair as developed by the author.

2.2 Definition of Life Cycle Cost

The Life Cycle Cost (LCC) is a method of calculating the total cost of ownership over the life span of asset. LCC has widely been used by governments to minimize the cost of projects over their useful life. In aircraft repair or alterations, LCC is an aid used to justify the production cost over its lifetime. When operators are faced with the choice between two or more proposals with a specific time limit, can be known from the LCC analysis for the cost for each proposal over its useful lifetime. LCC analysis can be established by using the present worth method (present value) and the Equivalent Uniform Annual Worth Method (EUAW) to analyse of all the expenses required to operate and to maintain a repair or a modification over its lifetime.

The method used to calculate LCC in this experiment is the Equivalent Uniform Annual Worth (EUAW) method. The advantage of using this method is, when the alternatives have different lives. The present dollar and the future dollar's value in EUAW are being converted to a uniform annual cost, while taking into account that time value of money is assumed at a particular interest rate. It means that all incomes and disbursements have to be converted into an equivalent uniform annual amount, that will be the same for each period.

To calculate the EUAW only requires the knowledge of the following values:

1. First cost (investment cost)
2. retirement (removal cost, if any),
3. annual cost (inspection, repair, etc.),
4. life of the alternative (n), and
5. interest rate, i.
Figure 2-1, A cost breakdown structure of repair
LIFE CYCLE COST

Then the EUAW can be determined as follows:

\[ \text{EUAW} = \text{EUAW of First cost} + \text{EUAW of Retirement cost} + \text{EUAW of Inspection cost} \]

Where:

- The EUAW of First cost is:

\[ \text{First Cost} * (AP; i \%, n) \]

In which \((AP; i \%, n)\) is the Capital-recovery factor. The Capital-recovery factor is when this factor is multiplied by the given acquisition cost \(P\) (first cost) it will yield the equivalent uniform annual worth \((CP)\), over \(n\) years at interest rate \(i\), reference [13]. The formula is as follows:

\[ CP = P * (AP; i \%, n) = P * \left( \frac{(1+i)^n - i}{(1+i)^n - 1} \right) \]

- The EUAW of the Retirement cost is:

\[ \text{Retirement cost} * (AF; i \%, n) \]

In which \((AF; i \%, n)\) is the Sinking-Fund factor. The Sinking-Fund factor is when this factor is multiplied by the given future worth \(F\) it will yield the uniform annual worth \((A)\), over \(n\) years at interest rate \(i\). The formula is as follows:

\[ A = F * (AF; i \%, n) = F * \left( \frac{i}{(1+i)^n - 1} \right) \]

In repair of aircraft, the annual cost represents the operating cost, in this case it is the inspections cost. The inspection of the aircraft consists of a threshold inspection and the repeat inspections, where these two types of inspections usually shall be carried out in a period of more than one year or during the major check visits. Therefore, the calculation of the EUAW of the operating cost should be determined for each inspection schedule. Whereas the future value of the inspection cost should be converted into the Present Value (PV) for each inspection cost [5]. The calculation of the \(PV\) is:
\[ PV = \frac{IC}{(1+i)^t} \]  

Where,

- \( IC \) is the cost per inspection,
- \( i \) is the interest rate in \( \% \), and
- \( t \) is the time when the inspection should be carried out.

Then, this \( PV \) of inspection cost should be added together for each inspection to establish the total present value (\( PVT \)) of a repair during its lifetime. From reference [13], when this value is multiplied by the Capital recovery factor it will yield the equivalent uniform annual worth of inspection (\( AI \)) over \( n \) years at interest rate \( i \) as follows:

\[ AI = PVT \times (AP; i \%, n) \]

\[ AI = PVT \times \left( \frac{(1+i)^n \cdot i}{(1+i)^n - 1} \right) \]

Where,

- \( i \) is the interest rate,
- \( n \) is the life of a repair in year, and
- \( PVT \) is the total present value of inspections cost.
2.3 References


3. THE LIFE CYCLE COST COMPARISONS OF SEVERAL REPAIR PROPOSALS

3.1 Conditions for LCC Calculations
3.2 Life Cycle Cost Comparisons
3.3 References

The maintenance costs are the elements of aircraft operating cost where the biggest improvements can be made. Reduction in heavy check down-times is one of the more recent advances in airframe maintenance. This can also be achieved through the extension of the airframe life or repair life by the use of advanced techniques or new materials alike Glare. The longer life of airframe can often result in considerable savings of maintenance hours, as a consequence, it reduces the aircraft operating cost and increases the aircraft availability.

This chapter introduces an example of cost comparisons of several repair proposals. The use of life cycle cost approach as an additional tool in maintenance engineering is relatively new for the airline engineer; therefore this example can give an expression about the application of LCC.

3.1 Conditions for LCC Calculations

The cost comparison of repair proposals is based only on the fatigue life of repair, other factors such as Environment Damages (ED) and Accidental Damages (AD) are not taken into account. This limitation shall be made in order to simplify the calculation.

The cost break-down structure of repairs used in this calculation has been explained in chapter 2 of part 4. It is based on the information of a repair performed in Garuda Indonesian on an MD 11 aircraft [7]. The repair was carried out in accordance with the Engineering order issued by the Engineering department. So the material and the man-hours calculation are based on this Engineering order. The damage of the aircraft is caused by a tail strike during take-off and the damage areas were between frames STA 1931 and STA 1964, and between longerons 51 L and 50 R. Figure 3-1 shows the drawing sketch taken from the Engineering order. From this drawing it can be seen that an aluminum repair consists of two aluminum 2024 T3 with both the thicknesses of 1.2 mm. Where for Glare repair as demonstrated by the test, in chapter 6 of part 1, it can be applied with only one skin layer and does not need to consist of
doublers as for aluminum, because of the little lower elastic modulus and the good fatigue resistance of Glare. Therefore, a Glare doubler can be installed with the thickness as the original skin or one or more gages thicker than the original skin thickness without any consequences in reducing the fatigue life. In this case, the thickness of the Glare doubler is 1.95 mm (Glare 3 4/3 0.3), or one gage thicker than the original skin.

For the simplification in calculation, the repair method employed for Glare remain the same as for aluminum repair. The faying surface used for those repairs are the sealant PR 1422-B2 and the rivet is NAS 1097 AD 6.

Figure 3-1, Repair skin of MD-11
3.2 Life Cycle Cost Comparisons

The aircraft maintenance activity must constantly consider to reduce the cost and introduce to new technologies. This requires a procedure to evaluate and then to be used as a basis for comparison of the various proposals.

When the airline engineer select the new repair proposal, he must compare the available proposals using an appropriate method. In this case, the method is Life Cycle Cost (LCC). This section illustrates how two or more repair proposals can be ranked using the LCC analysis.

The calculation charts shown in Table 3-1 and Table 3-3 (typical for airline), are designed to present the details of the acquisition cost (first cost) in aluminum and Glare repairs, respectively. Table 3-2 and Table 3-4 show the calculation chart of the operating cost for aluminum and Glare repairs, respectively.

As mentioned in chapter 2 of part 4, that the operating cost should be calculated in equivalent uniform annual worth. Therefore, operating cost is calculated for the individual year when the inspection must be carried out, then the cost of each inspection will be calculated in the present value ($PV$). From the total present value ($PVT$) of the operating cost, it then will find the $EUAW$ of the operating cost for each repair proposal.

The retirement cost of both repairs does not exist, because those repairs are classified as permanent repairs or class A for the Glare repair and class B (see repair classifications in chapter 7 of part 1) for the aluminum. It means that repairs have the same life as the aircraft itself.

The repair data used in this calculation are as follows:

- Aircraft type: MD - 11,
- Age: 5 years,
- Flight cycle: 900 cycles/ year,
- Aircraft overhaul (D check): 5 years,
- Aircraft life for calculation: 30 years, and
- The interest rate $i$: 5 % per year

**Aluminum repair**

- The repair threshold form Table 6-19 (chapter 6 of part 1) is 25.5 years, the inspection threshold for NDT must be carried out after 25 years (during the overhaul) or at 30 year (aircraft life), and
- The repeat inspection from Table 6-16 (chapter 6 of part 1) is 11.8 years, the repeat inspection for NDT must be carried out over every 10 years after the
threshold inspection. Thus, there is no repeat inspection because it is beyond the life of the aircraft.

**Glare repair**

- For the Glare repair, inspection is not necessary, because the inspection threshold found in Table 6-16 (chapter 6 of part 1) is 61.8 years or beyond the lifetime of the aircraft. Therefore there is no operation cost (inspection cost) for Glare repair.

From the Table 3-1 and Table 3-3 for aluminum repair and Table 3-2 and Table 3-4 for Glare repair, can be calculated the total EUAW (see chapter 2 of part 4) of each repair proposal as follows:

EUAW of Aluminum and Glare repair are:

\[
\text{EUAW} = \text{EUAW of First cost} + \text{EUAW of Inspection cost}
\]

For aluminum repair  
\[
\text{EUAW} = 438.343 + 1.337 \\
= 439.680 \text{ } US
\]

For Glare repair  
\[
\text{EUAW} = 393.301 + 0 \\
= 393.301 \text{ } US
\]

It shows that the LCC of Glare repair is lower compared with the aluminum repair, because the Glare repair consists of only one skin doubler instead of two as for the aluminum and the primered Glare sheet causes less man-hours for surface treatment. So, in this case, the high cost of purchasing Glare sheet is not to much effect in LCC, where the material cost of aluminum repair is only 3.4 percent of the total man-hour cost, and 11.2 percent for the Glare repair. It shall be remembered that the high ratio of the material over the man-hour cost of Glare repair, is due to the lower man-hour cost as shown in Table 3-3. If the total down time of the aircraft should be considered in a major check visit, then Glare repair will give a lower aircraft down time due to a longer inspection threshold and repeat inspection intervals.
3.3 References


231
Table 3-1, Repair calculation chart of aluminum repair, in US $.

<table>
<thead>
<tr>
<th>Material cost</th>
<th>Cost</th>
</tr>
</thead>
<tbody>
<tr>
<td>Outer external doubler Al 2024 T3 1.2 mm, area 0.65 m²</td>
<td>56.27</td>
</tr>
<tr>
<td>Inner external doubler Al 2024 T3 1.2 mm, area 0.79 m²</td>
<td>68.39</td>
</tr>
<tr>
<td>Filler Al 2024 T3 1.8 mm, area 0.21 m²</td>
<td>18.18</td>
</tr>
<tr>
<td>Stringer, 4 ea, Al 2024 T3 1.6 mm, area 0.66 m²</td>
<td>57.13</td>
</tr>
<tr>
<td>Scotch-brite, 5 pieces, a $ 1</td>
<td>5</td>
</tr>
<tr>
<td>Masking tape no 232-1, 1 roll, a $ 4 per roll</td>
<td>4</td>
</tr>
<tr>
<td>Rags, 5 kg, a $ 0.78 / kg</td>
<td>3.9</td>
</tr>
<tr>
<td>Lint-free cloth, 0.1 roll</td>
<td>10</td>
</tr>
<tr>
<td>Rivet NAS 1097 AD 6 (360 pieces ≈ 0.3 kg), a $ 16.5 / kg</td>
<td>5</td>
</tr>
<tr>
<td>MEK, 0.5 liter, a $ 1 / liter</td>
<td>0.5</td>
</tr>
<tr>
<td>Primer 0.2 liter, a $ 33/ kg</td>
<td>6.6</td>
</tr>
<tr>
<td>Paint 0.2 liter, a $ 35.3 / kg</td>
<td>7.06</td>
</tr>
<tr>
<td>Alodine 0.1 liter, a $ 2 / liter</td>
<td>.2</td>
</tr>
<tr>
<td>Drill 1 ea, a $ 2</td>
<td>2</td>
</tr>
<tr>
<td>Sealant PR 1422 B2 .83 L for 1.65 sq. m, a $ 87.50 / liter</td>
<td>43.75</td>
</tr>
</tbody>
</table>

**Man-Hours (Mech 50 S/ m-h, Eng 65 S/ m-h)**

- Assessing of damage, 6 m-h: 300
- Repair design and drawing (Engineering), 6 m-h: 390
- Cutting of damage area, 12 m-h: 700
- Stringer manufacturing and surface treatment, 14 m-h: 700
- Material cutting and matching, 6 m-h (for 2 doubler skin): 300
- Stringer installation, 16 m-h: 800
- Drilling and deburring, 26 m-h: 1300
- Surface treatment & primer, 5 m-h (for 2 doubler skin): 250
- Riveting & inspection, 20 m-h: 1000
- Painting, 3 m-h: 150

**EUAW of First cost**

<p>| | |</p>
<table>
<thead>
<tr>
<th></th>
<th></th>
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</thead>
<tbody>
<tr>
<td></td>
<td>5890</td>
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<td>6177.98</td>
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<td></td>
<td>438.343</td>
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Table 3-2, Operating cost of aluminum repair, in US $.

<table>
<thead>
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<th>Operating Cost, NDI (Eddy current, 50 S/ m-h)</th>
<th>Cost</th>
<th>P V</th>
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</thead>
<tbody>
<tr>
<td>Inspection threshold at year 25th, 1 m-h</td>
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<tr>
<td>Repeat inspection (beyond the aircraft life)</td>
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**EUAW of Inspection cost**

<p>| | |</p>
<table>
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<th></th>
<th></th>
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</thead>
<tbody>
<tr>
<td></td>
<td>1.337</td>
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Note: PV is present value
Table 3-3, Repair calculation chart of Glare repair, in US $.

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<td>External doubler Glare 3 4/3 0.3 1.95 mm, area 0.79 m²</td>
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<td>Filler Al 2024 T3 1.8 mm, area 0.21 m²</td>
<td>18.18</td>
</tr>
<tr>
<td>Stringer, 4 ea, Al 2024 T3 1.6 mm, area 0.66 m²</td>
<td>57.13</td>
</tr>
<tr>
<td>Scotch-brite, 5 pieces</td>
<td>5</td>
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<td>Masking tape no 232-1, 1 roll</td>
<td>4</td>
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<tr>
<td>Rags, 3 kg</td>
<td>2.34</td>
</tr>
<tr>
<td>Lint-free cloth, 0.1 roll</td>
<td>10</td>
</tr>
<tr>
<td>Rivet NAS 1097 AD 6 (360 pieces ≈ 0.3 kg)</td>
<td>5</td>
</tr>
<tr>
<td>MEK, 0.5 liter</td>
<td>0.5</td>
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<td>Primer, 0.2 liter</td>
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<td>Paint, 0.2 liter</td>
<td>7.06</td>
</tr>
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<td>Alodine, 0.1 liter</td>
<td>0.2</td>
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<td>Sealant PR 1422 B2, .83 L for 1.65 sq. m (.5 L/ sq. meter)</td>
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</table>


<table>
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<th>Man-Hours, (Mech 50 $/ m-h, Eng 65 $/ m-h)</th>
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</tr>
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<tbody>
<tr>
<td>Assessing of damage, 6 m-h</td>
<td>300</td>
</tr>
<tr>
<td>Repair design and drawing, 6 m-h</td>
<td>390</td>
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<td>Cutting of damage area, 12 m-h</td>
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<tr>
<td>Material cutting and matching, 4m-h (for 1 doubler skin)</td>
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<td>Stringer manufacturing and surface treatment, 14 m-h</td>
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<tr>
<td>Stringer installation, 16 m-h</td>
<td>800</td>
</tr>
<tr>
<td>Drilling and deburring, 20 m-h</td>
<td>1000</td>
</tr>
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<td>Surface treatment &amp; primer, 1.5 m-h (for 1 doubler skin)</td>
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<tr>
<td>Riveting &amp; inspection, 16 m-h</td>
<td>600</td>
</tr>
<tr>
<td>Painting, 3 m-h</td>
<td>150</td>
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</tbody>
</table>

<table>
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<th>EUAW of First cost</th>
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<td></td>
<td>4915</td>
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<td>5547.16</td>
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**EUAW of First cost**

<table>
<thead>
<tr>
<th>Operating Cost, NDI (Eddy current, 50 $/ m-h)</th>
<th>Cost</th>
<th>P V</th>
</tr>
</thead>
<tbody>
<tr>
<td>Inspection threshold (beyond the aircraft life)</td>
<td>0</td>
<td>0</td>
</tr>
</tbody>
</table>

**EUAW of Inspection cost**

0

Note; PV is present value
4. CONCLUSIONS

1. Glare repair in MD-11 fuselage skin resulted more economical as compared to the aluminum repair.

2. Glare repair resulted less man-hours, because it consist of one sheet. It could be adopted due to good fatigue behavior. Glare repair also does not required additional inspection during the life of the aircraft.
CONCLUSIONS and RECOMMENDATIONS

Overview

Since 1990, the market for passengers and freight increased, but the air-fares were dramatically reduced. It caused a world wide exhausting competition in the airline business. As a result there was a lot of pressure on the airlines and aircraft manufacturers to reduce maintenance costs.

In service, lack of time and insufficient space in aircraft structures have frequently led to inadequate repairs and to fatigue problems afterwards. Small changes of the current repair procedures are necessary to improve the airworthiness of the aircraft without increasing the cost of the repair. The application of Glare sheet and bonded-riveted repair as covered in part 1 is a way to achieve this goal. The use of Glare as a repair material as well as for new aircraft (designed) has been studied in depth. It has been confirmed that Glare repairs can reduce the Life Cycle Cost (LCC) to about 10 percent as compared to aluminum repair.

Impact damage and corrosion are critical issues for the aircraft cargo floors. The application of Glare sheet as a top facing material in cargo floor sandwich panels has been proposed in our experiments to improve the impact properties of floor panels. The results has been given in part 2.

Before compiling conclusions of the various chapters in more detail, the major findings in relation to the goals of the present research can be summarized as follows:

- Glare as a repair material for fuselage skin structures can give significant improvements as shown by extensive experimental programs and theoretical analysis.
• The improvements included longer repair fatigue lives, as well as reduced maintenance costs due to increased inspection threshold and repeat inspection periods.

• A bonded-riveted repair is highly superior to a riveted repair with a sealant on the faying surface.

• A study on repair options as made in the present thesis should be a combination of experimental efforts and theoretical analysis. Such studies are relevant to the operator only when it includes a cost-effectiveness analysis as well.

Conclusions of Part 1

1. The approval for application of Glare sheet material in fuselage repair of existing aircraft is classified as a major alteration, because it changes the original concepts of designing repairs for the primary structures. The approval of Glare as a repair material should be obtained from the aircraft manufacture by issuing a Service Bulletin (SB).

2. The load transfer of Glare 3 4/3 (2 gages thicker than the skin thickness) and an aluminum 2024 T3 skin of 1.6 mm in a lap joint resulted in a comparable load distributions in the top row and the bottom row of the rivets.

3. In a bonded-riveted lap joint, the load transfer by the first rivet row is only about a few percent, as shown in load transfer calculations. It indicates that the adhesive has a significant effect in load transfer in bonded-riveted joints. This trend has been confirmed in fatigue tests, where failures occurred at end of the overlapping at much higher fatigue life as compared with sealant as a faying surface.

4. In patch repair, Glare patch attracts less load into the repair area (soft patching) as compared to an aluminum alloy patch. Differences of the load transfer by the rivets of the first rivet row were very small. The combination of these small differences and the imperfection during manufacturing (e.g. drilling, riveting, etc.) caused the cracks initiate randomly from the rivet holes along the first row.

5. For a patch repair over a small cut-out, a thinner Glare sheet of 90 percent of the skin thickness, did not reduce the static strength as compared to aluminum 2024-T3 patch repair. The skin failures were at the first rivet row. It means that increasing the patch thickness will not increase the static strength.

6. Glare sheet patch repairs resulted in a higher fatigue life as compared to the aluminum 2024 T3 patch repair. The use of cold bonding adhesive (AW 106) as a faying surface increases the fatigue life significantly as compared to the sealant.
7. The use of the former countersunk holes in the skin underneath the doubler reduced the fatigue life of the patch repair with about 40 to 50 percent for both types of faying surfaces (sealant PR 1422 b2 and adhesive AW 106).

8. A thinner Glare sheet of 90 percent of the skin thickness in a lap joint repair reduced the static strength with about 15 percent as compared to a full aluminum 2024-T3 lap joint, but it still passed the ultimate load for the specific skin thickness of aluminum 2024 T3 of 1.6 mm.

9. Static failures of all specimens with sealant as a faying surface in the riveted lap joint repairs occurred by rivet shear. The application of cold bonding adhesive as a faying surface changes the failure mode from rivet shear to skin or patch fracture. It means that the adhesive has a significant effect in reducing the bearing stress due to the load transfer through the rivet.

10. The thicker Glare sheet, two gages thicker than the skin, with a sealant as a faying surface, increased the fatigue life significantly. Thus the use of Glare as a repair material for existing aircraft with the current repair method is no doubt favorable for fatigue as well as for static loading.

11. Fatigue tests on curved specimens loaded uniaxially have shown a significant fatigue life reduction if compared to flat specimens. The reduction factor on life of a full aluminum 2024 T3 1.6 mm lap joint with a sealant as a faying surface is about 2.9 and for a Glare 3 3/2 (1.45 mm) to aluminum 2024 T3 (1.6 mm) joint it was 1.4. Again, the Glare sheet shows a significant improvement in fatigue life of single lap joint.

12. The use of a new Repair Calculation and Classification Program (RCCP) to assess existing repairs and newly designed repairs as part of the maintenance activities, may reduce the engineering calculation hours significantly. This program is intended to help the airline engineers to analyze a repair for to ensure the required safety level. The RCCP is not an approved reference, therefore the use of this program in maintenance is not the responsibility of the author.

Conclusions of Part 2

1. Glare sheet as the top face of a cargo floor panel resulted in the highest impact resistance. Balsa as a core material gave the highest impact properties. The combination of Glare sheet as a top face and balsa as a core resulted in the highest impact properties of the panels tested.

2. The impact failure energy of a panel with a rigid support is much lower as compared the open support. In the rigid support, the bottom facing does not affect the impact process, but in the open circular support, the bottom facing is significant because of its yielding.
3. The impact tests have shown that a foam core, with a comparable density to the nomex core, but much less compression strength, reduced the impact properties.

4. Failures in bending tests on short sandwich beam specimens occurred by core shear. In the balsa core occurred between the lignins of the wood.

5. Chromic acid anodizing and the BR 127 primer as a surface treatment of the Glare sheet gave excellent corrosion protection in floor panel applications. There is no delamination between the facings and core due to the bond-line corrosion for all types of cores during the corrosion test.

6. Balsa as a core material showed an excellent corrosion behaviour, which is also true for the nomex core.

7. There are two types of corrosion found in the specimens after exposure in the salt spray chamber for corrosion testing, i.e. surface corrosion and pitting corrosion.

8. A service evaluation test of a Glare floor panel with a balsa core in a container cargo floor of an MD-11 aircraft gave favorable results. The life of this floor panel is at least twice the life of an aluminum floor panel (original floor panel). When removed from the aircraft, the condition of the Glare floor panel was still in a serviceable condition with minor scratches. These minor damages did not necessitate a repair.

Conclusions of Part 3

1. The use of Glare sheet for the fuselage skin of new aircraft design results in a significant maintenance advantage as compared to using the classical aluminum alloys sheet materials. A Glare fuselage skin on an MD-11 aircraft may reduce the inspection-hours with about 10 percent as compared to the aluminum one. This reduction is mainly due to the better corrosion and impact behavior of Glare sheet. Where the inspection tasks of the SSI and the zonal inspection program are due to corrosion (75-85 percent of the total task).

2. The low density and fatigue insensitivity of Glare sheet may result in a lighter aircraft, meaning less fuel burn or more pay load as compared to using aluminum alloys. The saving of the fuel burn of an aircraft (Airbus A 3xx) may reach about US $ 1.8 million (25 years) or $ 75,000 US per year (fuel price $ 0.25/ kg).

3. The production techniques and inspection methods of a Glare structure can be the same as for aluminum alloy structures. Thus, the use of Glare in production does not need new investment in tools as well as for training.
Conclusions of Part 4

1. The study of Glare repair in an aircraft (MD-11) shows that the Glare repair results in lowering Life Cycle Cost (LCC) as compared to the aluminum. This saving is mainly due to the better fatigue behavior of Glare sheet. Therefore, aluminum repair, the doubler consists of two aluminum sheets, but for Glare repair consists only one sheet with a thickness of one gauge thicker than the skin. Thus Glare repair requires less man-hours as compared to the aluminum repair.

2. Another advantage of Glare as a repair material is a longer inspection threshold and a longer repeat inspection interval, thus reducing maintenance costs.

Recommendations for Further Research

The present investigation for Glare as a repair material has clearly shown the advantages in patch repair and lap joint repair, as confirmed by analysis and testing. The approval process of Glare as a repair material for existing aircraft remains an important step in aircraft maintenance. Furthermore, extensive fatigue testing of Glare repairs is still needed in order to support this approval process, e.g. with respect to influences of rivet pitch, different rivet diameter, different rivet material, etc.

For repair assessment, the computer program RCCP can be further improved and expanded to include Glare repair. The influence of the repair location in relation to the stress system in each section of interest should be established for different types of aircraft.
Appendix A

RIVETED LAP JOINT

This appendix reviews the loads transfer analysis in a riveted lap joint. This lap joint as previously mentioned, which for determining the load transfer in a repair, where the skin cut-out is bigger than half frame distance or in a repair which over the skin lap joint of the fuselage.

The load transfer calculation in the aircraft repair is significant to determine the bearing stress in a repair, whereas it needs to predict the fatigue quality. In the special case, where the area of the repair is limited, while the number of rivet row and rivet pitch can not be applied in such good ways. This repair needs a good justification in determining the repair quality. To determine the load transfer in the riveted joint, the following simplification should be applied:

1. The skin and the doubler are assumed as a bar strip which is loaded in tension,

2. the rivets are simplified as an elastic spring constant under shear loading and so have a spring constant,

3. the moment due to unsimetry of loading in the single lap joint is ignored, and

4. the joint is modeled as shown in figure A-1 and A-2.

For fastener spring constant $C_{riv}$ or so called rivet flexibility, the model of Vlieger/Brooke in chap 3.2 has been chosen for load transfer calculation. The reason is due to this model considering the differences of types of the material for skin and doubler in the calculation of rivet flexibility.

$$C_{riv} = \frac{1}{E_r \cdot D} \left[ 5.0 + 0.8 \left( \frac{E_r \cdot D}{E_s \cdot t_s} + \frac{E_r \cdot D}{E_{pc} \cdot t_{pc}} \right) \right] \quad A-1$$

The spring constant of skin and doubler ($C_s$ and $C_d$) can be derived by the equation as follows:
\[ \delta = P \frac{L}{A.E} \]  

By the substitution of the cross section \( A \) by \( t_s \) (skin thickness) and \( W \) (rivet pitch) to the equation above, so the spring constant of skin and doubler can be written as follows:

\[ C_s = \frac{L}{t_s.W.E_s} \]  

and

\[ C_d = \frac{L}{t_d.W.E_d} \]

Where:

- \( t_s \) or \( t_d \) is the skin or doubler thickness,
- \( W \) is the pitch length,
- \( E_s \) or \( E_d \) is the skin or doubler modulus, and
- \( L \) is a distance between the rivet rows.

Figure A-1, Model of riveted lap joint.
Appendix A

Figure A-2, Idealization of riveted lap joint.

From Figure A-2 can be drawn the equilibrium of force as follows:

\[ P_{s1} - P_{s2} = F_1, \quad P_{s2} - P_{s3} = F_2 \ldots \text{etc} \]  \hspace{1cm} A-5

and

\[ P_{d1} - P_{d2} = -F_1, \quad P_{d2} - P_{d3} = -F_2 \ldots \text{etc} \]  \hspace{1cm} A-6

Where \( s \) and \( d \) denote for the skin and doubler. From the compatibility model of figure A-3 can be written the relation of the displacement between first rivet and second rivet as follows:

\[ L + \Delta L_{d1} + \Delta R_2 = L + \Delta L_{s1} + \Delta R_1 \]  \hspace{1cm} A-7

Where:

\( L \) is a distance between the rivet rows,

\( \Delta L_{d1} \) is the displacement of the doubler,

\( \Delta L_{s1} \) is the displacement of the skin, and

\( \Delta R_1 \) is the rivet displacement.

In the equation above, the distance between the rivet row \( L \) can be erased and the substitution of equations A-3 and A-4 to the equation A-7, then it can be rewritten as follows:
Figure A-3, Compatibility model of rivet displacement.

While, $P_{dl}$ is:

$$P_{dl} = F_2 + F_3 + ... etc$$  \hspace{1cm} \text{(A-9)}$$

and

$$P_{sl} = F_1; \text{ etc}$$  \hspace{1cm} \text{(A-10)}$$

Substitution equation A-9 and A-10 to A-8, thus:

$$- F_1 (C_s + C_{riv}) + F_2 (C_d + C_{riv}) + ... + F_i (C_d) = 0$$

$$... + ... + ... + ... = 0$$

$$- F_1 C_s + ... + F_{i-1} (C_s + C_{riv}) + F_i (C_d + C_{riv}) = 0$$

$$- F_1 - ... - ... - F_i = - F$$

This equation above can be written with the matrix as follows:
\[ \Delta ff = [C][F] \]  

So by inverting the flexibility matrix \([C]^{-1}\), then it can be derived from the loads transfer of each rivet row as follows:

\[ [F] = [C]^{-1} \cdot [\Delta ff] \]
Matchad Program for Load Transfer Calculation of Lap Joint with sealant Faying

This program is also available on disk in Material Laboratory, Faculty of Aerospace, Delft University of Technology, The Netherlands.

Geometry of Input:
Appendix A

Input

MPa = 1000000 Pa

Load = 100 MPa ——> Applied gross load

i = 3 ——> Number of rivet rows

L_0 = 20 mm ——> Distance between the rivet row

t_skin = 1.6 mm ——> Skin thickness in mm

t_doubler = 1.6 mm ——> Doubler thickness in mm

E_doubler = 72000 MPa ——> Doubler modulus

E_skin = 72000 MPa ——> Skin modulus

D = 3 mm ——> Rivet diameter

E_rivet1 = 70000 MPa ——> Rivet modulus

E_rivet2 = 70000 MPa ——> Rivet modulus

P_pitch = 20 mm ——> Rivet pitch

Program

Q = 1

m = i

i = i - 1

Force = Load \cdot t_skin \cdot P_pitch

L_0 + t_skin + D + P_pitch = 0.04 m

K_doubler = \frac{L_0}{t_doubler \cdot P_pitch \cdot E_doubler}

m = 0, 2.. i

K_skin = \frac{L_0}{t_skin \cdot P_pitch \cdot E_skin}

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\[
\begin{align*}
C_m &= \frac{5 - 0.8 \left( \frac{E_{\text{rivet}}}{E_{\text{doubler}}} \right) D + E_{\text{skint}}}{E_{\text{rivet}} D} \\
C_{m+1} &= \frac{5 + 0.8 \left( \frac{E_{\text{rivet2}}} {E_{\text{doubler}}} \right) D + E_{\text{skint}}}{E_{\text{rivet2}}} \\
\end{align*}
\]

\[j = 2 \ldots i\]

\[M_{0,0} = (K_{\text{skin}} + C_0), M_{0,1} = (K_{\text{doubler}} - C_1)\]

\[n = 1 \ldots i \quad j = 0 \ldots i\]

\[M_{n,j} = -K_{\text{skin}}, \quad k = 0 \ldots i - 2, \quad l = 2 \ldots i\]

\[M_{k,l} = K_{\text{doubler}}\]

\[k = 1 \ldots i - 1\]

\[M_{k,k} = -(K_{\text{skin}} + C_k), \quad M_{k,k+1} = (K_{\text{doubler}} + C_{k+1})\]

\[P_{i,0} = \text{Load-t\_skin P\_pitch (K\_skin)}, \quad \frac{K_{\text{skin}}}{C_1} = 0.23\]

\[F = \frac{M^{-1}P}{\text{Load-t\_skin P\_pitch}} \times 100\]

\[X = \frac{D}{P\_pitch}, \quad A = \frac{t\_doubler}{D}\]

**Severity factor calculation program**

**Constanta taken from Michael C. Y. Niu**

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Fillet radii</td>
<td>1.0 - 1.5</td>
</tr>
<tr>
<td>Standard hole drilled</td>
<td>1.0</td>
</tr>
<tr>
<td>Broached or remer</td>
<td>0.9</td>
</tr>
<tr>
<td>Cold worked hole</td>
<td>0.7-0.8</td>
</tr>
<tr>
<td>Open hole</td>
<td>1.0</td>
</tr>
<tr>
<td>Lock bolt (steel)</td>
<td>0.75</td>
</tr>
<tr>
<td>Rivet</td>
<td>0.75</td>
</tr>
<tr>
<td>Threaded bolt</td>
<td>0.75 - 0.9</td>
</tr>
<tr>
<td>Taper lock</td>
<td>0.5</td>
</tr>
<tr>
<td>Hi-lock</td>
<td>0.75</td>
</tr>
</tbody>
</table>
\[ \alpha = 1 \quad \beta = 0.75 \]

\[ K_{tb} = 125 \cdot X^3 - 3.7 \cdot X^2 + 1.7 \cdot X + 1 \]

\[ \theta = -0.05 A^3 + 0.4 A^2 + 0.75 A + 1 \]

\[ \begin{align*}
\sigma_{doubler_1} &= \frac{F_0}{100} \cdot K_{tb} \cdot \frac{D \cdot t_{doubler}}{\text{Load}} \quad \sigma_{skin_1} &= \frac{F_i}{100} \cdot K_{tb} \cdot \frac{D \cdot t_{skin}}{\text{Load}} \\
\text{Ratio}_{doubler} &= \frac{\sigma_{doubler_1}}{\text{Load}} \quad \text{Ratio}_{skin} &= \frac{\sigma_{skin_1}}{\text{Load}} \\
\end{align*} \]

\[ K_{tg} = ((24.76 X^4 - 8.19 X^3) + 4.22 X^2) + 0.35 X + 3 \]

\[ \begin{align*}
\sigma_{doubler_2} &= \frac{(100 - F_0)}{100} \cdot K_{tg} \cdot \frac{P_{pitch \_doubler}}{\text{Load}} \\
\sigma_{skin_2} &= \frac{(100 - F_i)}{100} \cdot K_{tg} \cdot \frac{P_{pitch \_skin}}{\text{Load}} \\
\sigma_{max \_skin} &= \sigma_{skin_1} + \sigma_{skin_2} \\
\sigma_{max \_doubler} &= \sigma_{doubler_1} + \sigma_{doubler_2} \\
\end{align*} \]

\[ \text{Ratio}_{flex \_rivet \_skin} = \frac{K_{skin}}{C_l} \]

\[ \text{SF}_{\text{skin}} = \frac{\alpha \cdot \beta \cdot \sigma_{max \_skin}}{\text{Load}} \quad \text{SF}_{\text{doubler}} = \frac{\alpha \cdot \beta \cdot \sigma_{max \_doubler}}{\text{Load}} \]

**Out-put**

\[ \begin{align*}
\begin{bmatrix} 35.55 \end{bmatrix} \\
\end{bmatrix}
\end{align*} \]

\[ \text{Result of load transfer, beginning with 1st row from the top, in the percent of total load} \]

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σ_max_skin = 635.23 MPa  ----> Stress maximum in rivet hole

σ_max_doubler = 635.23 MPa

Ratio_skin = 4.33  ----> Bearing ratio
Ratio_doubler = 4.33

SF_skin = 4.76  ----> Severity factor of skin and doubler
SF_doubler = 4.76
Appendix B

BONDED-RIVETED LAP JOINT

This appendix reviews the loads transfer analysis in bonded-riveted lap joint. In this case, the adhesive as previously mentioned in sub-chapter 4.5 is used as a faying surface. It will act as an anti fretting and will take a part in the load transfer. As a result, the bearing stress of the rivet holes will decrease significantly, and consequently increases the fatigue life of the joint.

To determine the load transfer in bonded-riveted joint, the following assumption should be taken:

1. The skin and the doubler shall be assumed as a bar strip which is loaded in tension as discussed in appendix A,

2. the rivet is simplified as an elastic spring constant under shear loading and have a spring constant as shown in appendix A,

3. the moment due to the unsimetry of the loading in a single lap joint is ignored,

4. the joint flexibility is mainly due to the rivet and adhesive layer,

5. the adhesive layer is expressed as a rivet, in this case the adhesive will deforms due to the bending moment and the shear force,

6. the adhesive assumes as a beam under the shear load, and

7. the joint is modeled as a riveted joint, which shows in figures B-1.

Furthermore, the flexibility of the adhesive can be determined with some assumptions which are discussed bellow:

The effects of the shear deformations on the deflections of a beam are usually relatively small as compared to the flexural deformations. However, in regards to the greater accuracy is required to calculate the shear deflection of the adhesive in bonded-riveted joint for load transfer calculation.

In this case, the unit-load method for obtaining the deflections is used based upon the
virtual work. So the expression for the external work is:

\[ W_{ext} = I \cdot \Delta \]  \hspace{1cm} B-1

Let consider that the elements of dimension \( dx, dy, \) and \( dz \) from the beam interior (figure B-2) are subjected to a unit load. Where acting from the sides of this element are the normal stress \( \sigma \) and the shear stresses \( \tau \) due to the bending moment \( M_u \) and the shear force \( V_u \) produced by the unit load. The stresses can be calculated using the flexure and shear formulas as follows:

\[ \sigma = \frac{M_u \cdot Y}{I} \]  \hspace{1cm} B-2

\[ \tau = \frac{V_u \cdot Q}{I \cdot b} \]  \hspace{1cm} B-3

![Diagram](image)

Figure B- 1 ,Idealization of bonded riveted lap joint.

The corresponding of the deformation due to the bending moment \( M_L \) and the shear force \( V_L \) are the extensional strain \( \varepsilon \) and the shear strain \( \gamma \), where the deformations are:

\[ \varepsilon = \frac{M_L \cdot y}{E \cdot I} \]  \hspace{1cm} B-4

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\[ \gamma = \frac{V_L \cdot Q}{G.I.b} \quad \text{B-5} \]

Figure B-2, Beam element.

Where, the internal virtual work of the stresses \( \sigma \) and \( \tau \) acts on the differential element. So the total internal work can be obtained by integrating throughout the volume of the beam as follows:

\[ W_{\text{int}} = \int (\sigma . dy . dz)(\epsilon . dx) + \int (\tau dy . dz)(\gamma . dx) \]

\[ W_{\text{int}} = \int \frac{M_u \cdot M_L \cdot y^2}{E . I^2} . dx . dy . dz + \int \frac{V_u \cdot V_L \cdot Q^2}{G . I^2 . b^2} . dx . dy . dz \quad \text{B-6} \]

To simplify this above expression, the following quantities are constants; \( M_u, M_L, V_u, V_L, E, G, \) and \( I \). So this can separate for each of the preceding integrals into an integration of the cross-sectional area and an integration along the axis of the beam, as follows:
\[ W_{\text{int}} = \int_{L} \frac{M_u \cdot M_L}{E \cdot I^2} \left[ \int_A y^2 \cdot dy \cdot dz \right] \cdot dx + \int_{L} \frac{V_u \cdot V_L}{G \cdot I^2} \left[ \int_A \frac{Q^2}{b^2} \cdot dy \cdot dz \right] \cdot dx \quad \text{B-7} \]

Where the symbols \( L \) and \( A \) denote that the integration is throughout the length \( L \) of the beam and the cross-sectional area \( A \), respectively. While the first term of equation B-7 in the brackets is the moment inertia \( I \) (property of the cross section). Then, the second term in brackets is depend only upon to the cross-sectional dimensions of the beam, so this new cross-sectional property is called form factor for shear \( f_s \) as follows:

\[ f_s = \frac{A}{I^2} \cdot \int_A \frac{Q^2}{b^2} \cdot dA \quad \text{B-8} \]

Where \( dA = dy \cdot dz \) represents an element of the area in cross section of the beam. The form factor is a dimension-less that depend on the shape of the beam. In this case the beam is rectangle, whereas the \( f_s \) from reference [9] in chapter 3, page 694, that is found for 1.2.

By replacing the two terms in the brackets in equation B-7 with \( I \) and \( f_s \cdot F / A \), respectively, then by equating the external work (equation B-1) with the internal work can be found by the unit load equation for the deflection \( \Delta \) as follows:

\[ \Delta = \int \frac{M_u \cdot M_L}{E \cdot I} \cdot dx + \int \frac{f_s \cdot V_u \cdot V_L}{G \cdot A} \cdot dx \quad \text{B-9} \]

To determine the shear deflections in a beam by using the unit load, let assume that the deflection at the midpoint of a uniformly loaded beam with a simple support. The coordinate \( x \) is measured from the left-hand support of the beam, so the expressions of bending moment and shear force due to the actual loads are:

\[ M_L = \frac{q \cdot L \cdot x}{2} - \frac{q \cdot x^2}{2} \quad \text{B-10} \]

\[ V_L = \frac{q \cdot L}{2} - q \cdot x \quad \text{B-11} \]

Where \( q \) is the intensity of the uniform load. Then the unit load acting at the middle of the beams results in the following bending moment and shear force:
\[ M_u = \frac{1.(x)}{2} \]

Where \( 0 < x < L/2 \)

\[ V_u = \frac{1}{2} \]

By substitution the equations B-10 - B-13 to equation B-9, so the expression for the deflection \( \delta \) at the midpoint of the beam can be written as follows:

\[
\delta = \frac{2}{E.I} \int_{0}^{L/2} \frac{x}{2} \left( q \cdot \frac{L}{2} - q \cdot x^2 \right) dx + \frac{2 \cdot f_s}{G \cdot A} \int_{0}^{L/2} \left( q \cdot \frac{L}{2} - q \cdot x \right) dx
\]

\[
= \frac{5 \cdot q \cdot L^4}{384 \cdot E \cdot I} \left( 1 + \frac{48 \cdot f_s \cdot E \cdot I}{5 \cdot G \cdot A \cdot L^2} \right)
\]

For a cantilever beam with a concentrated load \( P \) at the free end, the shear deflection can be written as follows:

\[
\delta = \frac{P \cdot L^3}{3 \cdot E \cdot I} \left( 1 + \frac{3 \cdot f_s \cdot E \cdot I}{G \cdot A \cdot L^2} \right)
\]

So from this equation can be obtained the rivet flexibility of an adhesive in bonded-riveted joint as follows:

\[
C_b = \frac{I^3}{3 \cdot E_a \cdot I} \left( 1 + \frac{3 \cdot f_s \cdot E_a \cdot I}{G \cdot A \cdot I^2} \right)
\]

Where:

- \( I \) is the thickness of the adhesive layer,
- \( E_a \) is adhesive modulus,
- \( I \) is moment inertia of adhesive,
- \( A \) is cross section area of adhesive,
- \( G \) is adhesive shear modulus,
- \( f_s \) is the form factor, for rectangle is equal 1.2.
By substitution the flexibility of the adhesive from equation B-16 to A-11, the compatibility model of the bonded-riveted joint can be obtained by the expression of the compatibility displacement as follows:

\[
[\Delta ff] = [C][F]
\]

So by inverting the flexibility matrix \([C]^{-1}\), can derive the loads transfer in bonded-riveted joint by the adhesive and the rivet:

\[
[F] = [C]^{-1}[\Delta ff]
\]
Matchad Program for Load Transfer Calculation of Lap Joint with Adhesive Faying

This program is also available on disk in Material Laboratory, Faculty of Aerospace, Delft University of Technology, The Netherlands.

Geometry of Input:
Input:

\[ \text{MPa} := 1000000 \text{Pa} \quad \text{Load} := 100 \text{ MPa} \]

\[ i = 5 \quad \text{--> Number of rivet rows} \]

\[ L_0 = 20 \text{-mm} \quad \text{--> Distance between the rivet rows} \]

\[ t_{\text{skin}} = 1.6 \text{-mm} \quad \text{--> Skin thickness in mm} \]

\[ t_{\text{doubler}} = 1.6 \text{-mm} \quad \text{--> Doubler thickness in mm} \]

\[ E_{\text{doubler}} = 72000 \text{ MPa} \quad \text{--> Doubler modulus} \]

\[ E_{\text{skin}} = 72000 \text{ MPa} \quad \text{--> Skin modulus in Mpa} \]

\[ D = 3 \text{-mm} \quad \text{--> Rivet diameter} \]

\[ E_{\text{rivet}} = 70000 \text{ MPa} \quad \text{--> Rivet modulus in Mpa} \]

\[ P_{\text{pitch}} = 20 \text{-mm} \quad \text{--> Rivet pitch in mm} \]

\[ t_{\text{glue}} = 0.2 \text{-mm} \quad \text{--> Thickness of glue} \]

\[ E_{\text{glue}} = 2870 \text{ MPa} \quad \text{--> Glue modulus} \]

\[ G = 1070 \text{ MPa} \quad \text{--> Glue shear modulus} \]

Program

\[ i := i - 1 \quad m := i \quad Q := 1 \]

\[ \text{Force} := \text{Load} \cdot t_{\text{skin}} \cdot P_{\text{pitch}} \]

\[ K_{\text{doubler}} = \frac{L_0}{t_{\text{doubler}} \cdot P_{\text{pitch}} \cdot E_{\text{doubler}}} \]

\[ K_{\text{skin}} = \frac{L_0}{t_{\text{skin}} \cdot P_{\text{pitch}} \cdot E_{\text{skin}}} \quad m = 0, \ldots, i \]

\[ C_m = \left[ 1 + \frac{3 \cdot 1.2 \cdot E_{\text{glue}} \cdot P_{\text{pitch}} \cdot L_0^3}{G \cdot \left( P_{\text{pitch}} \cdot L_0 - \frac{3.14 \cdot D^3}{4} \right) \cdot t_{\text{glue}}^3} \right] \cdot \frac{t_{\text{glue}}^3}{3 \cdot E_{\text{glue}} \cdot P_{\text{pitch}} \cdot L_0^3} \]
\[ C_0 = \left[ 1 - \frac{3.12E_{\text{glue}}P_{\text{pitchL}}0^3}{G\left(P_{\text{pitchL}}0.5L_0 - 0.5\times3.14 \frac{D^2}{4}\right) \cdot t_{\text{glue}}^2} \right] \cdot \frac{(t_{\text{glue}})^3}{3E_{\text{glue}}P_{\text{pitchL}}0^3} \]

\[ C_{m+1} = \frac{5 + 0.8 \left( \frac{E_{\text{rivetD}}}{E_{\text{doublerL}} + E_{\text{skintL}}} \right)}{E_{\text{rivetD}}} \]

\[ C_i = \left[ 1 + \frac{3.12E_{\text{glue}}P_{\text{pitchL}}0^3}{G\left(P_{\text{pitchL}}0.5L_0 - 0.5\times3.14 \frac{D^2}{4}\right) \cdot t_{\text{glue}}^2} \right] \cdot \frac{t_{\text{glue}}^3}{3E_{\text{glue}}P_{\text{pitchL}}0^3} \]

\[ j = 2..i \]

\[ M_{0,0} = (K_{\text{skinL}} + C_0) \quad M_{0,i} = (K_{\text{doublerL}} + C_i) \]

\[ M_{0,j} = K_{\text{doublerL}} \]

\[ n = 1..i \quad j = 0..i \]

\[ M_{n,j} = -K_{\text{skinL}} \quad k = 0..i-2 \quad l = 2..i \]

\[ M_{k,1} = K_{\text{doublerL}} \quad k = 1..i-1 \]

\[ M_{k,k} = -(K_{\text{skinL}} + C_k) \quad M_{k,k+1} = (K_{\text{doublerL}} + C_{k+1}) \]

\[ P_{i,0} = \text{Load} \cdot t_{\text{skinL}}P_{\text{pitch}}(-K_{\text{skinL}}) \]

\[ F = \frac{\text{M}^1 \cdot P}{\text{Load} \cdot t_{\text{skinL}}P_{\text{pitch}}} \cdot 100 \]

**Output**

\[
\begin{bmatrix}
48.1535 \\
0.6079
\end{bmatrix}
\]

\[ F = 2.4772 \quad \text{--> Result of load transfer, beginning with 1st row from the top, in the percent of total load} \]

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Appendix C

FATIGUE TEST RESULT OF SINGLE LAP JOINT REPAIR

In chapter 6 of part 1 the fatigue tests of single lap joint have been carried out. This appendix are summarizing the individual S-N curve of the fatigue test of single lap joint. These test result represent of a repair with the cut-out size larger than half frame distance or a repair over the skin lap joint. The specimens consist of the skin of Al 2024 T3 1.6 mm (lower part) with eight posible combinations of doubler materials (upper part) and faying surfaces, as mention in Table C-1.

Table C-1, Lap joint repair configuration.

<table>
<thead>
<tr>
<th>Series</th>
<th>Type of doubler and faying</th>
</tr>
</thead>
<tbody>
<tr>
<td>A-Series</td>
<td>Alclad 2024 T3 1.6 mm with PR 1422 B2 sealant</td>
</tr>
<tr>
<td>B-Series</td>
<td>Glare 3 3/2 1.4 mm with PR 1422 B2 sealant</td>
</tr>
<tr>
<td>C-Series</td>
<td>Glare 3 4/3 1.95 mm with PR 1422 B2 sealant</td>
</tr>
<tr>
<td>D-Series</td>
<td>Glare 3 3/2 1.4 mm with AW 106 adhesive</td>
</tr>
<tr>
<td>E-Series</td>
<td>Glare 3 3/2 1.4 mm with EC 2216 B/ A adhesive</td>
</tr>
<tr>
<td>F-Series</td>
<td>Glare 3 3/2 with EU 9323 B/ A adhesive</td>
</tr>
<tr>
<td>G-Series</td>
<td>Alclad 2024 T3 1.6 mm with PR 1422 B2 sealant and countersunk flaw on skin</td>
</tr>
<tr>
<td>H-Series</td>
<td>Alclad 2024 T3 1.6 mm with EU 9323 B/A adhesive and countersunk flaw on skin</td>
</tr>
</tbody>
</table>

The fatigue test program consists of four different stress levels; 80 Mpa, 100 Mpa, 120 Mpa and140 Mpa, while each stress level consist of two identical specimens. All specimens were tested to failure, except for specimens which were still intact after undergoing more than 1.6 million fatigue cycles which were static tested to determine their residual strengths. The frequency was chosen in range from 10 Hz to 14 Hz. The test were done at room temperature in an labatory. These test result can be used to determine the fatigue life of a repair, by means of qualitative comparison using the severity factor, i. e. the higher severity factor has a lower fatigue life.

The dimension of the specimens are shown in Figure C-1. The manufacturing process of the fatigue test specimens were done in the same manner as for the static. The detail of this process can be seen in chapter 5 of part 1.
Figure C-1, The dimensions of the lap joint specimen.
Figure C-2, Fatigue performance of Al clad 2024 T3 1.6 mm lap joint with PR 1422 B2 sealant.

Figure C-3, Fatigue performance of Glare 3 1.4 mm and Al clad 2024 T3 1.6 mm lap joint with PR 1422 B2 sealant.

Figure C-4, Fatigue performance of Glare 3 1.95 mm and Al clad 2024 T3 1.6 mm lap joint with PR 1422 B2 sealant.
Figure C-5, Fatigue performance of Glare 3 1.4 mm and Alclad 2024 T3 1.6 mm lap joint with AW 106 adhesive.

Figure C-6, Fatigue performance of Glare 3 1.4 mm and Alclad 2024 T3 1.6 mm with EC 2216 adhesive.
Figure C-7, Fatigue performance of Glare 3 1.4 mm and Alclad 2024 T3 1.6 mm lap joint with EU 9323 adhesive.

Figure C-8, Fatigue performance of Alclad 2024 T3 1.6 mm lap joint with countersink flaws and PR 1422 B2 sealant.
Figure C-9, Fatigue performance of Alclad 2024 T3 1.6 mm lap joint with countersink flaws and EU 9323 adhesive.
Appendix D

ENVIRONMENTAL AND ACCIDENTAL DAMAGES RATING SYSTEMS OF GLARE SHEET

The purpose of this appendix is to give an example of the development of the rating system for new material, Glare. The development of these rating systems is based on the similar structures of the existing aircraft (MD-11) and pertinent test results. The Fatigue Damage (FD) is not discussed here because this evaluation is done as part of the certification requirements of the aircraft (specifically FAR and JAR 25.571).

As part of the structural inspection program development it is necessary to rate each Structural Significant Item in terms of susceptibility (likelihood of damage) and delectability (timely detection of damage). This study may give as an input for further development of the rating system for Glare structure.

Rating Environmental Deterioration

Environmental deterioration rating systems shall allow to evaluate for susceptibility to and timely detection of corrosion and delamination. The susceptibility to corrosion and delamination should assess on the basis of probable exposure to an adverse environment and adequacy of the protective system. The protection systems of Glare structures are assumed as same manner for aluminum structures. To determine an effective program for ED inspection, the following areas must be considered.

- Corrosion type,
- material susceptibility to ED,
- level of contamination, e.g., skydroll, oil, fuel, etc., and
- level of protection.

Due to the good corrosion behavior of Glare as compared to aluminum alloy, as discussed in chapter 2 of part 3, it results that a Glare structure will has a longer recommended interval as for aluminum. To calculate the inspection interval of ED,
there are three rating systems as follows:

**Susceptibility to ED**

<table>
<thead>
<tr>
<th>Description</th>
<th>Rating</th>
</tr>
</thead>
<tbody>
<tr>
<td>Highly susceptible to most forms of general corrosion and stress corrosion attack</td>
<td>5</td>
</tr>
<tr>
<td>Highly susceptible to most forms of general corrosion or stress corrosion attack</td>
<td>4</td>
</tr>
<tr>
<td>Average susceptible to many forms of general corrosion and stress corrosion attack</td>
<td>3</td>
</tr>
<tr>
<td>Average susceptible to only a few forms of corrosion attack</td>
<td>2</td>
</tr>
<tr>
<td>Highly tolerant to all of general corrosion and stress corrosion attack</td>
<td>1</td>
</tr>
</tbody>
</table>

**Level of Contamination**

<table>
<thead>
<tr>
<th>Description</th>
<th>Rating</th>
</tr>
</thead>
<tbody>
<tr>
<td>Highly to be exposed to a corrosive environment (bilge area, under lavatory, Lower fuselage skin, etc.), with a high probability of having built-in stress</td>
<td>5</td>
</tr>
<tr>
<td>Highly likely to be exposed to either a corrosive environment or high built-in stresses</td>
<td>4</td>
</tr>
<tr>
<td>Average likelihood of exposure to a corrosive environment</td>
<td>3</td>
</tr>
<tr>
<td>Average likelihood of exposure to either a corrosive environment or built-in stresses</td>
<td>2</td>
</tr>
<tr>
<td>Very unlikely to be exposed to either a corrosive environment or built-in stresses</td>
<td>1</td>
</tr>
</tbody>
</table>

**Level of Treatment**

<table>
<thead>
<tr>
<th>Description</th>
<th>Rating</th>
</tr>
</thead>
<tbody>
<tr>
<td>No treatment is utilized</td>
<td>2</td>
</tr>
<tr>
<td>Some general treatments are given with an average likelihood of treatment preventing some forms of corrosions</td>
<td>1</td>
</tr>
<tr>
<td>General treatment is applied with an average likelihood of treatment preventing most forms of corrosion attack</td>
<td>0</td>
</tr>
<tr>
<td>Special treatment is applied for preventing most forms of corrosion</td>
<td>-1</td>
</tr>
<tr>
<td>Special treatment is applied with an excellent chance of preventing all types of corrosion</td>
<td>-2</td>
</tr>
</tbody>
</table>

**Inspection interval calculation**

The totals rating values of those three ratings as discussed above are summed, and recommended inspection intervals are assigned based on the following guidelines.
<table>
<thead>
<tr>
<th>Total rating value</th>
<th>Recommended Interval</th>
</tr>
</thead>
<tbody>
<tr>
<td>0 - 5</td>
<td>60 Months</td>
</tr>
<tr>
<td>5 - 8</td>
<td>30 Months</td>
</tr>
<tr>
<td>8 - 11</td>
<td>15 Months</td>
</tr>
<tr>
<td>11 - 14</td>
<td>*</td>
</tr>
</tbody>
</table>

* Special inspection program may be necessary, this SSI should be re-evaluated to determine if special treatment and/ or redesign is warrant.

**Rating Accidental Damage**

Accidental Damage (AD) is the physical deterioration of an item caused by contact or impact with an object or influence which is not part of the aircraft. These may cause by human error during manufacturing, operation of the aircraft or maintenance practices. The AD rating systems should include of the following:

- Damage sources,
- Residual strength after accidental damage on the SSI, and
- Timely detection of damage.

The rating systems of AD are as follows:

**Susceptibility to AD**

<table>
<thead>
<tr>
<th>Description</th>
<th>Rating</th>
</tr>
</thead>
<tbody>
<tr>
<td>High probability of damage occurring due to several AD sources</td>
<td>5</td>
</tr>
<tr>
<td>High probability of damage occurring due to only a few AD sources</td>
<td>4</td>
</tr>
<tr>
<td>Average probability of damage to occur due to several sources</td>
<td>3</td>
</tr>
<tr>
<td>Average probability of damage to occur due to only few sources</td>
<td>2</td>
</tr>
<tr>
<td>Very unlikely for damage to occur</td>
<td>1</td>
</tr>
</tbody>
</table>

**SSI tolerance of AD**

<table>
<thead>
<tr>
<th>Description</th>
<th>Rating</th>
</tr>
</thead>
<tbody>
<tr>
<td>Low residual strength and/ or high crack growth when AD present and unlikely detection of AD with visual inspection</td>
<td>5</td>
</tr>
<tr>
<td>Low residual strength/ high crack growth with a high degree of visual inspection to detect AD</td>
<td>4</td>
</tr>
<tr>
<td>Average residual strength and crack growth, likelihood of detection with a visual inspection</td>
<td>3</td>
</tr>
<tr>
<td>Excellent residual strength/ crack growth with an average likelihood of detection with general visual inspections</td>
<td>2</td>
</tr>
<tr>
<td>Excellent residual strength and crack growth with excellent detection with visual inspection</td>
<td>1</td>
</tr>
</tbody>
</table>

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Inspection interval calculation

The totals rating values of those three ratings as discussed above are summed, and recommended inspection intervals are assigned based on the following guidelines.

<table>
<thead>
<tr>
<th>Total rating value</th>
<th>Recommended Interval</th>
</tr>
</thead>
<tbody>
<tr>
<td>2 - 4</td>
<td>90 Months</td>
</tr>
<tr>
<td>5 - 6</td>
<td>30 Months</td>
</tr>
<tr>
<td>7 - 8</td>
<td>15 Months</td>
</tr>
<tr>
<td>9 - 10</td>
<td>*</td>
</tr>
</tbody>
</table>

* Special inspection program may be necessary, this SSI should be re-evaluated to determine if special treatment and/ or redesign is warrant.
SAMENVATTING (Dutch Summary)


De onderzochte reparatietechnieken zijn de huidige geklonken reparaties, gelijmd-geklonken reparaties en het gebruik van Glare als reparatiemateriaal. De toepassing van Glare plaatmateriaal in reparaties aan de romp is goed voor de damage tolerance van geklonken reparaties. Lijm als continue tussenlaag in geklonken reparaties verbetert de vermoeingslevensduur aanzienlijk, evenals de statische sterkte. De combinatie van Glare plaatmateriaal met gelijmd-geklonken reparaties resulteert in de grootste vermoeingslevensduur.

Om de reparatieklasse te kunnen bepalen is het noodzakelijk het Reparatie Assessment Programma (RAP) in vliegtuigonderhoud te implementeren. Dit brengt echter een niet te verwaarlozen hoeveelheid manuren met zich mee voor evaluatie, met name voor het beoordelen van de huidige reparatietechnieken. Om de onderhoudsdienst te helpen het aantal engineering-uren te reduceren en een voldoende veiligheidsniveau te behouden is een computerprogramma ontwikkeld, het zogenaamde Reparatie Calculatie en Classificatie Programma (RCCP).

Het onderzoek aan nieuwe vloerpanelconfiguraties is uitgevoerd middels variatie van het materiaal van de toplagen en kern. Het testprogramma bestond uit impact, buiging, druk en corrosie. De keuze van een Glare sandwich voor de laadvloer is de beste oplossing gebleken. Het resulteert in een significant langere levensduur.

Uitgebreid onderzoek naar de toepassing van Glare voor huidbeplating in recent ontworpen vliegtuigen, tonen dat brandstofbesparing en reductie van
onderhoudskosten gerealiseerd kunnen worden door de gewichtsbesparing en de damage tolerance eigenschappen van de geavanceerde Glare constructie.

Een economische analyse is essentieel om de technische voordelen van Glare reparaties te bevestigen. De Life Cycle Cost analyse heeft bevestigd dat, in vergelijking tot aluminium legeringen, Glare als reparatiemateriaal het meest economisch is.
CURRICULUM VITAE

Soerjanto Tjahjono was born on May 23, 1960 in Jakarta. After graduating from high school in 1979, he entered the Institute of Technology Bandung (ITB), where he studied Mechanical and Aeronautical Engineering and graduated in 1986.

He joined the Garuda Indonesia Airline as a system analyst and computer programmer. In 1988, he was transferred to Garuda Maintenance Facility (GMF) commissioned an airframe and power-plant technician. In beginning 1990, he joined to Engineering department as junior structure engineer. In 1991, as the lead of narrow Body structure group for Fokker F-28, DC-9 and Boeing 737, and he was assigned to develop Reliability Control Program for Garuda Indonesia Airline. Since 1991 he served also as an instructor for Corrosion Control Program, Composite Repair and Airworthiness. In the end of 1993 he led of the structure department.

In January 1995 he moved to Delft, The Netherlands to begin his doctoral studies under supervision of Prof. ir. L. B. Vogelesang and Prof. ir. K. Smit. As part of the Structures and Materials group, the author investigates Glare as a repair material in existing aircraft and as a replacement for aircraft structural parts (e.g. cargo floor panel).