ISTANBUL TECHNICAL UNIVERSITY ★ GRADUATE SCHOOL OF SCIENCE ENGINEERING AND TECHNOLOGY

STRUCTURAL REPAIR DESIGN METHODOLOGY FOR AIRCRAFT FUSELAGE

M.Sc. THESIS

Zehra HASILCI

Department of Mechanical Engineering

Machine Design Programme

SEPTEMBER 2014

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İSTANBUL TEKNİK ÜNİVERSİTESİ ★ FEN BİLİMLERİ ENSTİTÜSÜ

UÇAK GÖVDESİ İÇİN YAPISAL TAMİR TASARIM METODOLOJİSİ

YÜKSEK LİSANS TEZİ

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Zehra HASILCI, a **M.Sc.** student of ITU Graduate School of Science Engineering and Technology student ID 503111233, successfully defended the thesis entitled "**STRUCTURAL REPAIR DESIGN METHODOLOGY FOR AIRCRAFT FUSELAGE**", which she prepared after fulfilling the requirements specified in the associated legislations, before the jury whose signatures are below.

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FOREWORD

In preparing this thesis, it was necessary to obtain and collect vast amounts of information and data from many sources. Sincere, appreciation is given to the THY Technic A.S. Structural Repair Shop Team for their gracious help. Thanks also to those who contributed to this thesis, Assist. Prof. Dr. Vedat TEMIZ. Special thanks to my mother and my father for their never ending moral and financial supports.

Lastly, it is my hope that this thesis, with its wide scope and information on the application of technology on aircraft structural design. It will prove not only to be a valuable reference tool for designing sound airframes with structural integrity but also as a bridge to carry over the valuable experience. I hope you will enjoy reading this thesis.

September, 2014

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ABBREVIATIONS

$\mathbf{A_i}$	**:** Cross Sectional Area of the Element [mm^2]
$\mathbf{a_i}$	**:** External Breadth of the Element [mm]
$\mathbf{b_i}$	**:** Calculate Element Breadth [mm]
$\mathbf{E_c}$	**:** Young's Modulus [Mpa]
$\mathbf{F_{bru}}$	**:** Allowable Ultimate Bearing Stress. Examples use e/d = 2.0 [Mpa]
$\mathbf{F_{su}}$	**:** Allowable Ultimate Stress in Pure Shear [Mpa]
$\mathbf{F_{tu}}$	**:** Allowable Tensile Stress [Mpa]
$\mathbf{F_{ty}}$	**:** Yield Strength at 0,2% Strain [Mpa]
L	**:** Profile Length [mm]
r	**:** Radius [mm]
$\mathbf{s_i}$	**:** Element Thickness [mm]

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STRUCTURAL REPAIR DESIGN METHODOLOGY FOR AIRCRAFT FUSELAGE

ABSTRACT

The aircraft structure is continually faced with requirements for openings at webs and panels to provide access or to let other members such as control rods or cables, hydraulic lines, electric wire bundles, etc., pass through. Other cutouts such as: windows, doors, servicing panels and hatches. Moreover, an aircraft fuselage can be damaged due to numerous ways such as a bird crash, lightning, hail squall and a baggage car crash. These structural damages have to be repaired by cutout fuselage. The cut-out repairs with doubler on aircraft fuselage are investigated in this thesis.

Engineers view any cutouts in airframe structures with disfavor because the necessary reinforcement of the cutout increase costs and adds weight to the overall design. In addition, the design and sizing of cutouts is a difficult process and a problem area for both static and fatigue strength and there is insufficient design data.

The main goal of the project is to develop a fast & temporary repair design. When an aircraft is damaged out of the Turkish Technic Inc., the damaged aircraft must be returned as soon as possible. Otherwise, the company must send qualified worker and spend a lot of money like passengers and technicians of hotel & other costs. Thus, the repair’s cost will be very high. Furthermore, the repair which is done in the out of the company sometimes gives trouble. This study is aimed to use as a general benchmark for future work in Turkish Technic Inc.

This thesis is intended to repair to airframe structures must be calculated in terms of static and fatigue. The thesis shows how to obtain the number of fasteners for fast & temporary repair and fatigue life. Step – by – step calculations are explained for repair design.

From structural standpoint, the thesis is intended to be used as a tool to help achieve structural integrity according to international regulations, specifications, criteria, etc., for designing commercial aircraft. It can also be considered as a troubleshooting guide for airline structural maintenance and repair engineers.

Key Words: Aircraft Repair Design, Rapid Repair, Structural Design, Reverse Engineering.

UÇAK GÖVDESİ İÇİN YAPISAL TAMİR TASARIMI METODOLOJİSİ

ÖZET

Uçak yapısı, webde ve panellerde erişimi sağlamak için ya da control çubukları, kablolar, hidrolik borular, elektrik kablo demetleri gibi arasından geçen diğer elemanların yerleşimi için sürekli olarak deliklerin delinmesi sorunuyla karşı karşıya kalır. Diğer kesimler; pencereler, kapılar, besleme panelleri ve kapaklardır. Bunun yanı sıra, uçak gövdesi kuş çarpması, yıldırım hasarı, şiddetli dolu yağışları ve bagaj arabası çarpması gibi çok çeşitli şekillerde zarar görebilir. Bu gibi yapısal hasarların uçak gövdesinin kesilmesiyle tamir edilmesi gerekir. Bu tezde uçak gövdesi üzerinde doubler kullanılarak yapılan kesim tamirleri incelenmiştir.

Mühendisler uçak üzerinde herhangi bir kesimin olması taraftarı değildirler; çünkü kesimler tüm tasarımın ağırlığını ve maliyetini arttırır. Bunun yanı sıra, tasarım ve kesimlerin boyutunu belirleme zor bir süreçtir. Hem statik & yorulma stresleri açısından hem de tasarım için yetersiz veri olduğundan problemler vardır.

Projenin asıl amacı, hızlı ve geçici bir tamir tasarımı geliştirmektir. Uçak Türk Havayolları Teknik A.Ş. dışında herhangi bir yerde zarar gördüğünde, uçak olabildiğince çabuk geri getirilmelidir. Aksi taktirde, şirket kalifiye işçilerini yollamak zorunda kalır. Ayrıca, yolcuların & teknik personelin otel ve diğer masraflarına bir çok para harcamak zorunda kalır. Bu yüzden, tamir maliyetleri çok yüksek olacaktır. Dahası, şirket dışında yapılan tamir bazı sorunlara yol açacaktır. Bu çalışmanın Türk Havayolları Teknik A.Ş.'de gelecek çalışmalar için genel bir örnek niteliğinde kullanılması amaçlanmıştır.

Bu tez static ve yorulma açısından uçak gövdesindeki yapısal hasarları tamir edebilmek amacıyla hazırlanmıştır. Tez, hızlı & geçici bir tamir için bağlantı sayılarının ve yorulma ömrünün nasıl belirleneceğini gösterir. Tamir tasarımı için adım adım hesaplar açıklanmıştır.

Yapısal bir bakış açısından, tez ticari uçak tasarımları için, uluslarası düzenlemelere, şartnamelere, kriterlere… arasındaki yapısal bütünlüğü kurmak için bir araç olarak kullanılması amaçlanmıştır. Aynı zamanda, havayolu yapısal bakım ve tamir mühendisleri için bir tamir rehberi olarak göz önünde bulundurulabilir.

Anahtar Kelimeler: Uçak Tamir Tasarımı, Hızlı Tamir, Yapısal Tasarım, Tersine Mühendislik

1. GENERAL INFORMATION

1.1.Introduction

In aviation sector, maintenance is of great importance. Maintenance organizations have to fix damages on aircraft in a rapid and sustained manner in order to provide aircraft continually working. These organizations have aircraft structural maintenance manuals provided by the aircraft manufacturers. In these manuals there are sections explaining how to do some basic maintenance. By this way, simple damage can be repaired on the aircraft. Although most of the damages that occurred while using aircraft are simple, damages out of the contents of the aircraft structural maintenance manual also occurred frequently. In these cases, in order to repair these damages maintenance organizations prepare a report to aircraft manufacturer that defines the damage and ask for the permanent repair solution. Manufacturers review these reports and send the final permanent maintenance to the maintenance organizations. As expected, these services cost a lot of money to the maintenance organizations. Eventually, maintenance organizations have to hire engineers that have necessary knowledge and infrastructure about technical properties of aircraft so that maintenance organizations can reach up a level that they repair each kind of damages. Only manufacturers possess of technical data of various load's effects on aircraft which they do not share with others. In these situations maintenance organizations have to use *reverse engineering methods* for these damage repairs.

The design of repairs for airframe structural damage requires a working knowledge of *material properties and applications, the nature of structural loading and deformation, and the principles of joint design*. These topics are explained in this study. The objective of this thesis is to increase the probability that repairs for damage outside the scope of the SRM will be accepted by an approved regulatory authority.

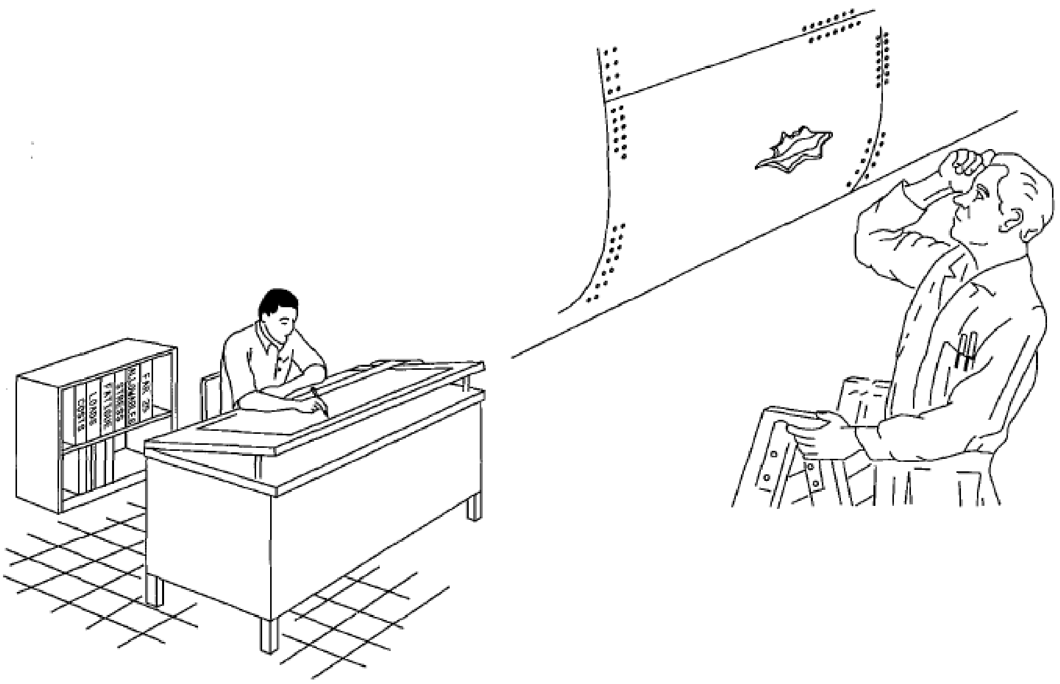


Figure 1.1 The Design of Repairs for Airframe Structural Damage

1.2.Development Progress

The modern aeronautical engineering of aircraft design has been an evolutionary process accelerated tremendously in recent times from the demanding requirements for safety and the pressures of competitive economics in structural design. For example:

1900 – 1915: In this period, the Wright Brothers' demonstration of practical mechanical flight, power requirements, stability and control were overriding considerations. A successful flight was one which permitted repair and turn around in a few weeks or days. Strength considerations were subordinate and ultimate strength of a few critical parts was the extent of structural analysis.

1915 – 1930: World War I accelerated the solution of power plants and stability and control problems. Engine reliability was improved by ground qualification (fatigue) testing.

1930 – 1940: Commercial development of metal aircraft for public transport took place in this era. Design and analysis emphasized static ultimate strength and, except for the engine, had little or no consideration for airframe fatigue.

1940 – 1955: During this period, there grew an increasing awareness of the fatigue potential in airframe safety. A large increase in performance capability resulted from WW II technology. Higher material static strengths were developed without a corresponding increase in fatigue strength. Static ultimate design alone was not sufficient; it was joined by fatigue design.

1955 – present: Safety from fatigue alone was recognized to be inadequate; fail-safe and damage tolerance, i.e., static strength of damaged requires adequate inspection intervals to discover and repair fatigue and other damage before cracks reach catastrophic proportions.

The primary objective of the structural designer is to achieve the maximum possible safety margin and achieve the maximum possible safety margin and achieve a 'reasonable' lifetime of the aircraft structure. Economic obsolescence may not come as soon as anticipated. For example, some of the old DC-3's still flying today are approaching or exceeding 100,000 hours of service. This record is achieved only by fail-safe structure, knowledge of when and where to look for cracks, and replacement of a few vital parts [11].

1.3. Loads - External and Internal

1.3.1. External Load

External loads are typically defined as those forces and loads that act on the aircraft structure. These loads determine the required strength of the structural elements of the aircraft. The external loads discussed in this course may be placed in one of the following categories:

- ✓ Air or Maneuvering
- ✓ Ground
- ✓ Power Plant
- ✓ Pressurization

1.3.2. Internal Loads

Internal loads are those loads experienced by the structural elements as a result of applied external forces and loads.

These are examples of internal loads:

- ✓ Tension
- ✓ Compression
- ✓ Bending
- ✓ Torsion
- ✓ Shear

It is the magnitude and direction of the internal loads that determine the sizing and type of materials and fasteners needed to meet the load requirements [21].

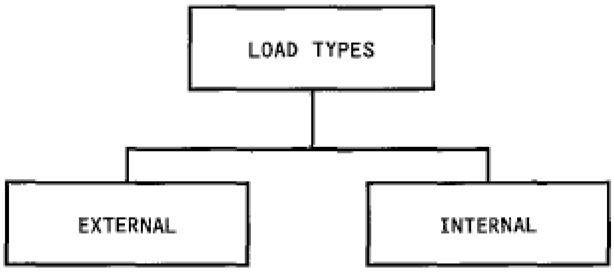


Figure 1.2 Loads – External and Internal

The basic fuselage structure is essentially a single cell thin walled tube with many transverse frames or rings and longitudinal stringers to provide a combined structure which can absorb and transmit the many concentrated and distributed applied forces safely and efficiently. The fuselage is essentially a beam structure subjected to bending, torsional and axial forces. The ideal fuselage structure would be one free of cut-outs or discontinuities, however a practical fuselage must have many cutouts [21].

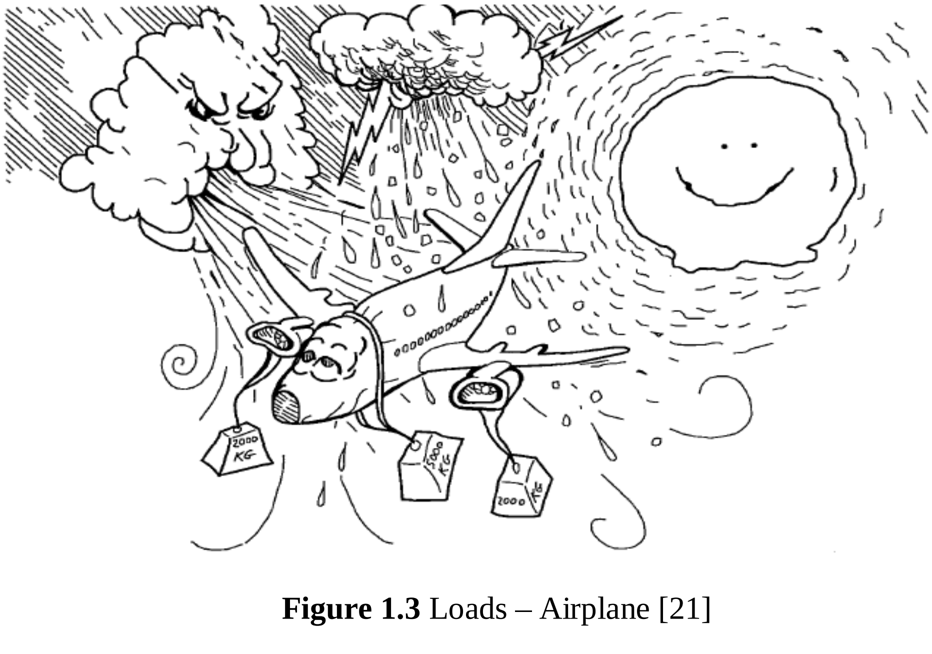


Figure 1.3 Loads – Airplane [21]

Figure 1.4 shows the basic interior fuselage structure of a small airplane with skin removed. It consists of transverse frames and longitudinal stringers [12].

Figure 1.4 Fuselage Construction

Turkish Airlines is a maintenance company. Producing companies (Airbus and Boeing) do not share loads of aircrafts. They give Turkish Airlines limited information such as Structural Repair Manuel (SRM), material and rivet to repair aircraft, so reverse engineering is done for repair design in this thesis.

1.4.Explanation of Used Terminology

A brief summary of that subject is presented in the following paragraphs to emphasize principles of importance regarding the use of allowables for various metallic materials.

<u>Basis</u>; Primary static design properties are provided for the following conditions:

- ✓ Tension . . . . F_{tu} and F_{ty}
- ✓ Compression . . . . . F_{cy}
- ✓ Shear . . . . F_{su}
- ✓ Bearing . . . . F_{bru} and F_{bry}

These design properties are presented as A- and B- or S-basis room temperature values for each alloy. Elongation and reduction of area design properties listed in room temperature property tables represent procurement specification minimum requirements, and are designated as S-values. Elongation and reduction of area at other temperatures, as well as moduli, physical properties, creep properties, fatigue properties and fracture toughness properties are all typical values unless another basis is specifically indicated.

Use of B-Values — The use of B-basis design properties is permitted in design by the Air Force, the Army, the Navy, and the Federal Aviation Administration, subject to certain limitations specified by each agency. Reference should be made to specific requirements of the applicable agency before using B-values in design [1].

Stress; The term "stress" as used in this thesis implies a force per unit area and is a measure of the intensity of the force acting on a definite plane passing through a given point.

$$Stress = \frac{F}{A} = \frac{Deforming\ Force}{Area}$$

Strain; Strain is the change in length per unit length in a member or portion of a member. As in the case of stress, the strain distribution may or may not be uniform in a complex structural element, depending on the nature of the loading condition. Strains usually are present also in directions other than the directions of applied loads.

$$Strain = \frac{Change\ in\ Dimension}{Original\ Dimension}$$

There are three types of strain;

- Linear Strain is the ratio of the change in length to the original length.

$$Linear\ Strain = \frac{Change\ in\ \ Length}{Original\ Length}$$

- Volume or Bulk Strain is the ratio of the change in volume to the original volume.

$$Volume\ Strain = \frac{Change\ in\ \ Volume}{Original\ Volume}$$

- Shearing Strain is equal to the angle of shear θ.

$$\text{Shearing } Strain = Angle\ of \text{ Shear } \theta$$

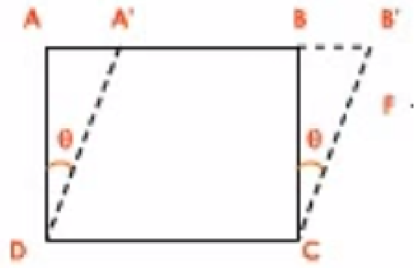


Figure 1.5. Angle of Shear

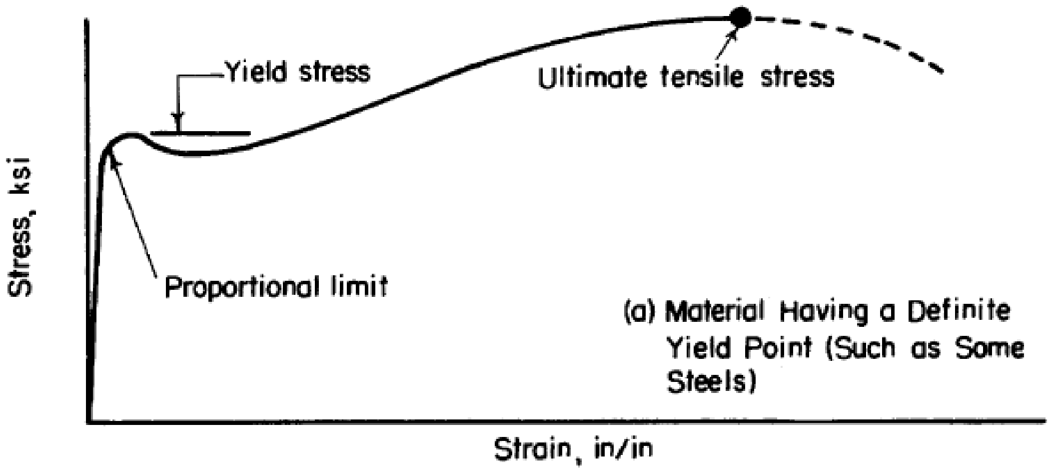
1.4.1.Tensile properties

When a metallic specimen is tested in tension using standard procedures of ASTM E 8, it is customary to plot results as a “stress-strain diagram.” Typical tensile stress strain diagrams are characterized in Figure 1.6. The general format of such diagrams is to provide a strain scale nondimensionally (in./in.) and a stress scale in 1000 lb/in. (ksi) [1].

1.4.1.1.Tensile yield stress (TYS or F_{ty})

Stress-strain diagrams for some ferrous alloys exhibit a sharp break at a stress below the tensile ultimate strength. At this critical stress, the material elongates considerably with no apparent change in stress. See the upper stress-strain curve in Figure 1.6.

The stress at which this occurs is referred to as the yield point. Most nonferrous metallic alloys and most high strength steels do not exhibit this sharp break, but yield in a monotonic manner. This condition is also illustrated in Figure 1.6. Permanent deformation may be detrimental, and the industry adopted 0.002 in./in. plastic strain as an arbitrary limit that is considered acceptable by all regulatory agencies. For tension and compression, the corresponding stress at this offset strain is defined as the yield stress (see Figure 1.6). This value of plastic axial strain is 0.002 in./in. and the corresponding stress is defined as the yield stress. For practical purposes, yield stress can be determined from a stress-strain diagram by extending a line parallel to the elastic modulus line and offset from the origin by an amount of 0.002 in./in. strain. The yield stress is determined as the intersection of the offset line with the stress-strain curve [1].



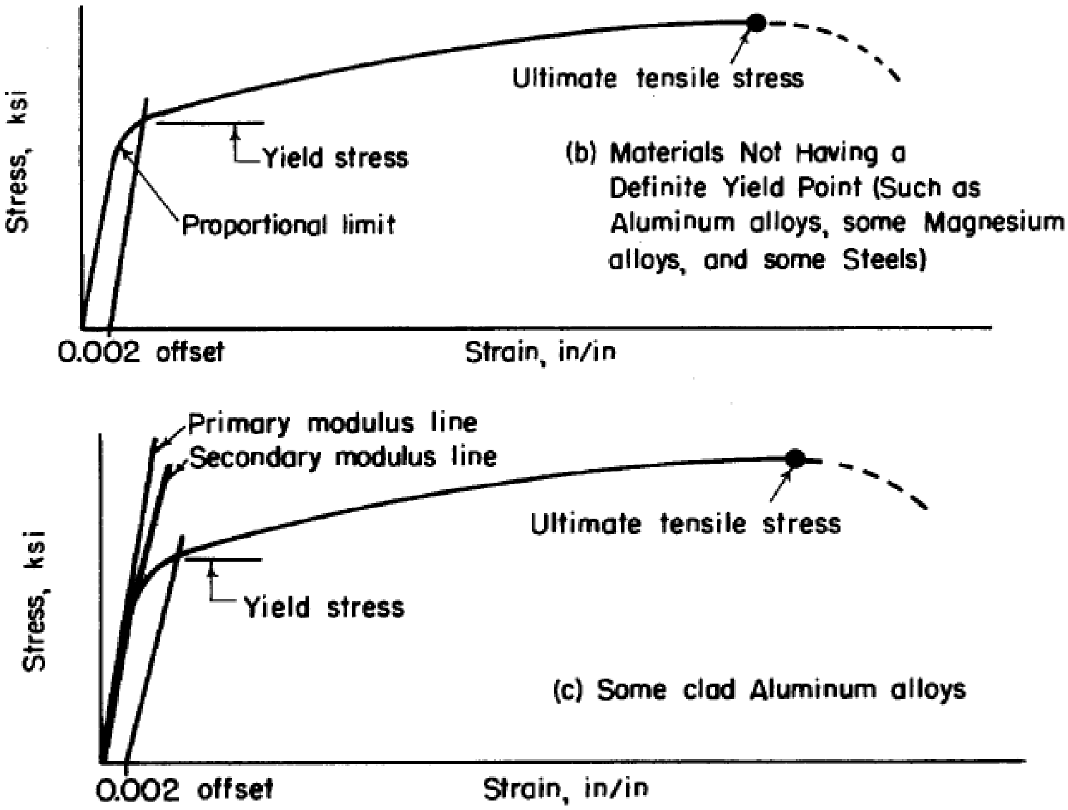


Figure 1.6 Typical Tensile Stress-Strain Diagrams [1]

1.4.1.2. Tensile ultimate stress (TUS or F_{ty})

Figure 1.6 shows how the tensile ultimate stress is determined from a stress-strain diagram. It is simply the maximum stress attained. It should be noted that all stresses are based on the original cross-sectional dimensions of a test specimen, without regard to the lateral contraction due to Poisson's ratio effects. That is, all strains used herein are termed engineering strains as opposed to true strains which take into account actual cross sectional dimensions. Ultimate tensile stress is commonly used as a criterion of the strength of the material for structural design, but it should be recognized that other strength properties may often be more important.

1.4.2. Compressive properties

Results of compression tests completed in accordance with ASTM E 9 are plotted as stress-strain curves similar to those shown for tension in Figure 1.6. Preceding remarks concerning tensile properties of materials, except for ultimate stress and elongation, also apply to compressive properties. Moduli are slightly greater in compression for most of the commonly used structural metallic alloys. Special considerations concerning the ultimate compressive stress are described in the following section [1].

1.4.3. Shear properties

Results of torsion tests on round tubes or round solid sections are plotted as torsion stress-strain diagrams. The shear modulus of elasticity is considered a basic shear property. Other properties, such as the proportional limit stress and shear ultimate stress, cannot be treated as basic shear properties because of “form factor” effects. The theoretical ratio between shear and tensile stress for homogeneous, isotropic materials is 0.577 [1].

1.4.4. Bearing properties

Bearing stress limits are of value in the design of mechanically fastened joints and lugs. Only yield and ultimate stresses are obtained from bearing tests. Bearing stress is computed from test data by dividing the load applied to the pin, which bears against the edge of the hole, by the bearing area. Bearing area is the product of the pin diameter and the sheet or plate thickness.

In the definition of bearing values, t is sheet or plate thickness, D is the pin diameter, and e is the edge distance measured from the center of the hole to the adjacent edge of the material being tested in the direction of applied load.

1.4.5. Fatigue properties

Repeated loads are one of the major considerations for design of both commercial and military aircraft structures. Static loading, preceded by cyclic loads of lesser magnitudes, may result in mechanical behaviors (Ftu , Fty , etc.) lower than those published in room temperature allowables tables. Such reductions are functions of the material and cyclic loading conditions [1].

1.4.5.1. Fatigue of materials

In materials science, fatigue is the progressive and localized structural damage that occurs when a material is subjected to cyclic loading. The nominal maximum stress values are less than the ultimate tensile stress limit, and may be below the yield stress limit of the material [8].

Fatigue lifes of materials are different from each other. For example, as it is seen in the Figure 1.7; Aluminum (2x24) is 1.0, Aluminum (7xxx) is 0.9, Titanium (Ti-6AL-4V) is 1.8 and Steels are 2.6.

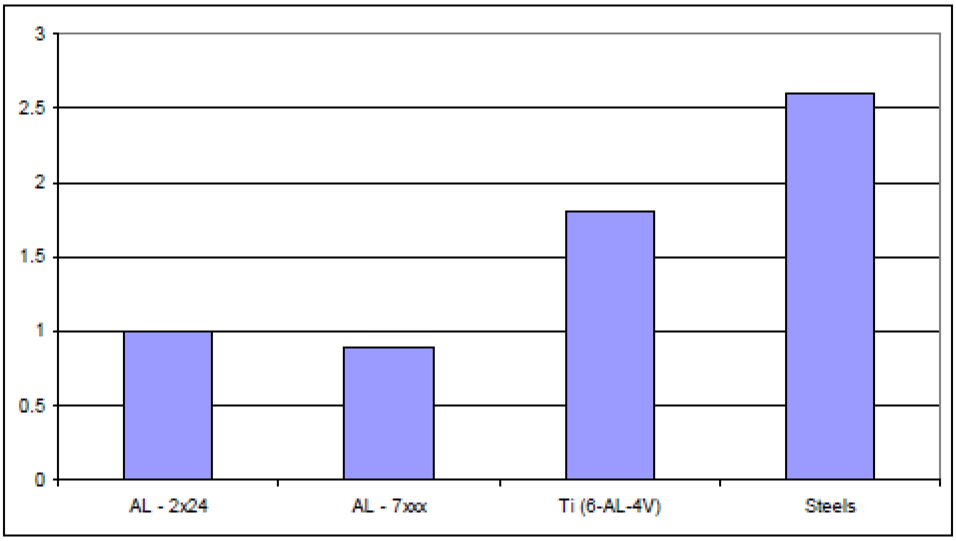


Figure 1.7 Factors Influencing Fatigue Life [18]

Fatigue occurs when a material is subjected to repeated loading and unloading. If the loads are above a certain threshold, microscopic cracks will begin to form at the *stress concentrators* such as the surface, persistent slip bands (PSBs), and grain interfaces. Eventually a crack will reach a critical size, and the structure will suddenly fracture. The shape of the structure will significantly affect the *fatigue life*; square holes or sharp corners will lead to elevated local stresses where fatigue cracks can initiate. Round holes and smooth transitions or fillets are therefore important to increase the fatigue strength of the structure [8].

1.4.5.2. Fatigue life

ASTM defines fatigue life, Nf, as the number of stress cycles of a specified character that a specimen sustains before failure of a specified nature occurs. One method to predict fatigue life of materials is the Uniform Material Law (UML). UML was developed for fatigue life prediction of aluminum and titanium alloys by the end of 20th century and extended to high-strength steels and cast iron. For some materials, there is a theoretical value for stress amplitude below which the material will not fail for any number of cycles, called a fatigue limit, endurance limit or fatigue strength.

In metals and alloys, when there are no macroscopic or microscopic discontinuities, the process starts with dislocation movements, eventually forming persistent slip bands that nucleate short cracks. Macroscopic and microscopic discontinuities as well as component design features which cause stress concentration (key ways, sharp changes of direction etc) are the preferred location for starting the fatigue process. Fatigue is a stochastic process, often showing considerable scatter even in controlled environments. The greater the applied stress range, the shorter the life. Fatigue life scatter tends to increase for longer fatigue lives. Damage is cumulative. Materials do not recover when rested. Fatigue life is influenced by a variety of factors, such as temperature, surface finish, micro-structure, presence of oxidizing or inert chemicals, residual stresses, contact (fretting), etc [8].

1.4.5.3. Aircraft rivet fatigue performance

An important criteria for aircraft rivets is their ability to resist vibration damage. Smoking rivets pictured above have vibrated loose; the longer they remain in service the more the hole is damaged. The rivet's ability to transmit stress across the lap is hindered.

Figure 1.8 Smoking Rivets [15]

Joint fatigue life depends on the ability of rivets to resist vibration damage. Below are some generalized factoids on preventing rivet loosening and increasing joint fatigue life.

1. Joint fatigue life is highly dependent on proper rivet installation.

2. Good quality holes improve joint fatigue life. Hole preparation that minimizes roughness, scratches, and other drilling defects improves fatigue life.

3. Tolerance stack up may work against you and decrease fatigue life. A -5 blind rivet has a hole tolerance of .160-.164 inch. #20 drill has a nominal diameter of 0.1610 inch. Its actual size can be from 0.1603 to 0.1610. Drilled holes, even using good techniques, very by -0.0010 to + 0.0030 tolerance. A drill on the high side of tolerance .1610 and a hole on the high side of tolerance .003 takes you to the extreme of 1640.

4. Rivets in tension reduce joint fatigue life.

5. Do not substitute rivets unless you understand the difference between shear critical and bearing critical joints. Poor fatigue life if you get it wrong.

6. Knife edge conditions reduce fatigue life. Per Boeing SRM, the countersink depth must not exceed 60% of the material thickness. Anything greater, the material is considered knife-edged and a poor fatigue detail. May be fine for static loads but poor fatigue in cyclic loads.

Corollary: Avoid knife edge conditions on pressurized skin [15].

Figure 1.9 Crack Propagation on the Fuselage [15]

1.5. Aircraft Accidents

As can be seen in the example below, calculation of rivets is important in the aircraft design in terms of static and fatigue. If aircraft designs or repair designs are wrong, numerous people can die or get injured.

1.5.1.Aloha Airlines flight 243

On April 28, 1988, at 1346, a Boeing 737-200, N73711, operated by Aloha Airlines Inc., as flight 243, experienced an explosive decompression and structural failure at 24,000 feet, while en route from Hilo, to Honolulu, Hawaii. Approximately 18 feet from the cabin skin and structure aft of the cabin entrance door and above the passenger floor line separated from the airplane during flight. There were 89 passengers and 6 crewmembers on board. One flight attendant was swept overboard during the decompression and is presumed to have been fatally injured; 7 passengers and 1 flight attendant received serious injuries. The flight crew performed an emergency descent and landing at Kahului Airport on the Island of Maui [16].

Damage extended from the main entrance door, aft about 18 feet (5.5 m). The airplane was determined to be damaged beyond repair, and was dismantled on site. According to the official NTSB report of the investigation, Gayle Yamamoto, a passenger, noticed a crack in the fuselage upon boarding the aircraft prior to the ill-fated flight but did not notify anyone [9].

Figure 1.10 Aloha Airlines Flight 243

The fuselage failure initiated in the lap joint; the failure mechanism was a result of *multiple site fatigue cracking* of the skin adjacent to rivet holes along the lap joint upper rivet row and tear strap disband which negated the fail-safe characteristics of the fuselage. Finally, the fatigue cracking initiated from the knife edge associated with the countersunk lap joint rivet holes; the knife edge concentrated stresses that were transferred through the rivets because of lap joint disbanding [9].

1.5.2. Rapid decompression due to fuselage rupture

On April 1, 2011, Southwest Airlines flight 812, a Boeing 737-300 registration N632SW, experienced a rapid depressurization caused by a rupture in the fuselage. The flight was at 34,000 feet when the depressurization occurred. The flight crew conducted an emergency descent and diverted the flight to Yuma International Airport, Yuma, AZ. At the time of the accident, the aircraft had accumulated 48,740 hours of service and 39,781 cycles (a cycle is a takeoff and landing). The accident aircraft was delivered to Southwest Airlines on June 13, 1996 [17].

Figure 1.11 The Boeing 737–300

On-scene inspection by NTSB investigators revealed an approximately 9-inch wide by 59-inch long rectangular-shaped hole in the fuselage crown on the left side of the airplane, aft of the over-wing exit. The 59-inch longitudinal fracture occurred in the aluminum fuselage skin along the lap joint at stringer-4 left (S-4L) between body station (BS) 666 and BS 725. At S-4L, the crown skin overlaps the lower skin forming a lap joint. The two skins are connected at the lap joint by three rows of rivets (referred to as lower, middle, and upper row of rivets.) The fracture was through the lower skin and connected 58 consecutive rivet holes in the lower row of lap joint rivets. The exterior surface of the skin in the area of S-4L is painted blue. Evidence of blue paint was also found inside the joint between the upper and lower skin and on several areas of the skin fracture surface.

Following an on-scene examination of the accident aircraft, a portion of the fuselage skin that contained the hole and another portion of the skin located forward of the hole (total size 116 inches by 19 inches) were excised from the accident aircraft and transported to the NTSB Materials Laboratory in Washington, DC. The airplane was then released back to the operator.

At the NTSB Materials Laboratory, microscope examination of the fracture faces of the ruptured skin revealed *fatigue cracks* emanating from at least 42 of the 58 rivet holes connected by the fracture. Electrical conductivity measurements, hardness tests, and X-ray energy dispersive spectroscopy elemental analysis of the skin in the area of the fracture revealed that the aluminum skin material was consistent with the specified material. The skin was the specified thickness.

Non-destructive eddy current inspections conducted around intact rivets on the removed skin section forward of the rupture revealed crack indications at nine rivet holes in the lower rivet row of the lap joint. To assess the condition of the intact rivets and the skin rivet holes, X-ray inspections were performed on the skin located forward of the rupture location. This inspection revealed gaps between the shank portions of several rivets and the corresponding rivet holes for many rivets associated with S-4L. Upon removing selected rivets, the holes in the upper and lower skin were found to be slightly offset relative to each other and many of the holes on the lower skin were out of round [17].

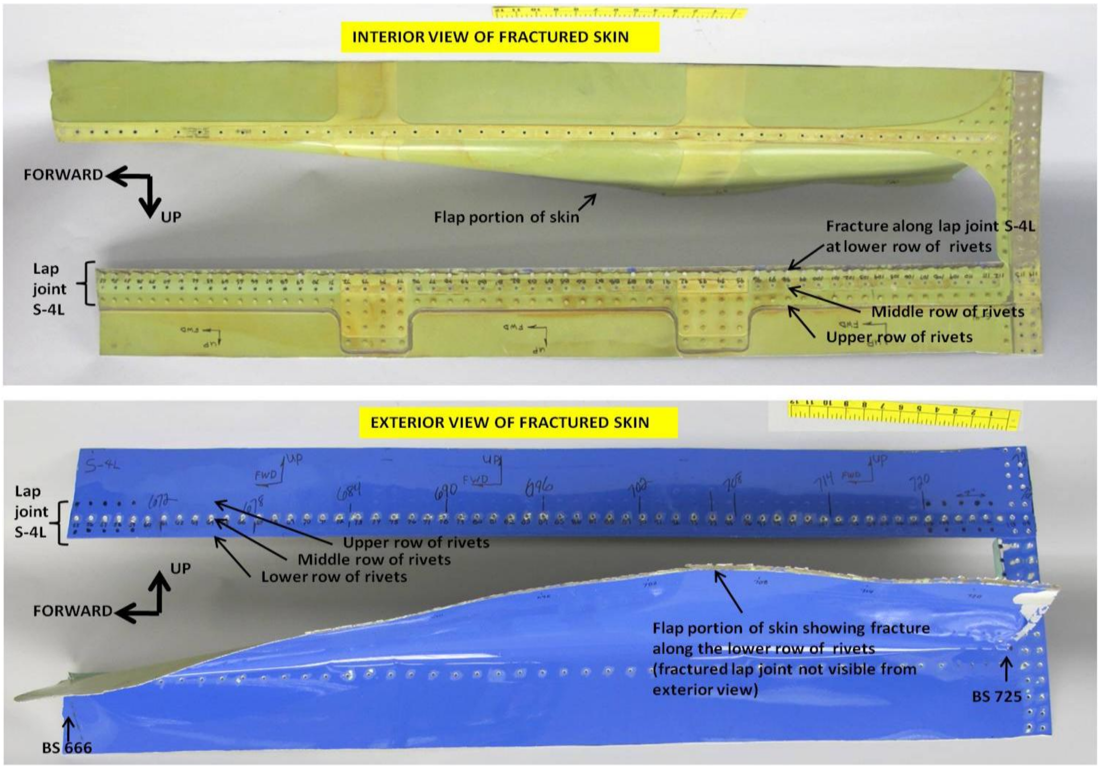


Figure 1.12 Fractured Skin of the Boeing 737–300 [17]

1.6. Stress Concentration

The elementary stress formulas used in the design of structural members are based on the members having a constant section or a section with gradual change of contour (Figure 1.13). Such conditions, however, are hardly ever attained throughout the highly stressed region of actual machine parts or structural members. The presence of shoulders, grooves, holes, keyways, threads, and so on, results in modifications of the simple stress distributions of Figure 1.13 so that localized high stresses occur as shown in Figures 1.14 and 1.15. This localization of high stress is known as stress concentration, measured by the *stress concentration factor*. The stress concentration factor K can be defined as the ratio of the peak stress in the body (or stress in the perturbed region) to some other stress (or stresslike quantity) taken as a

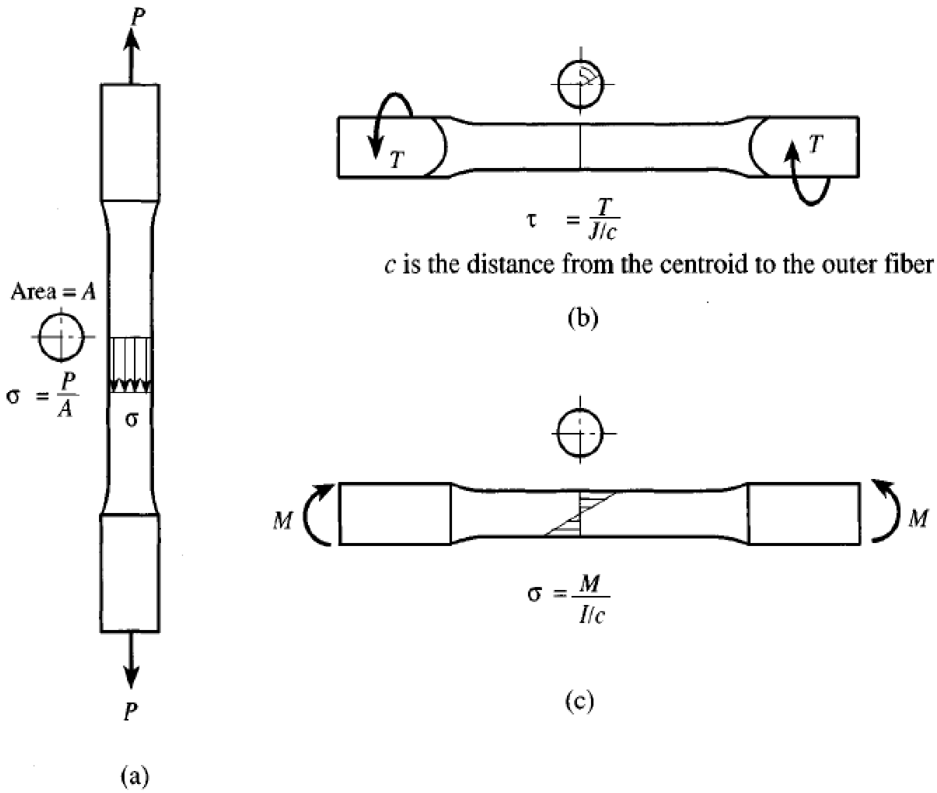
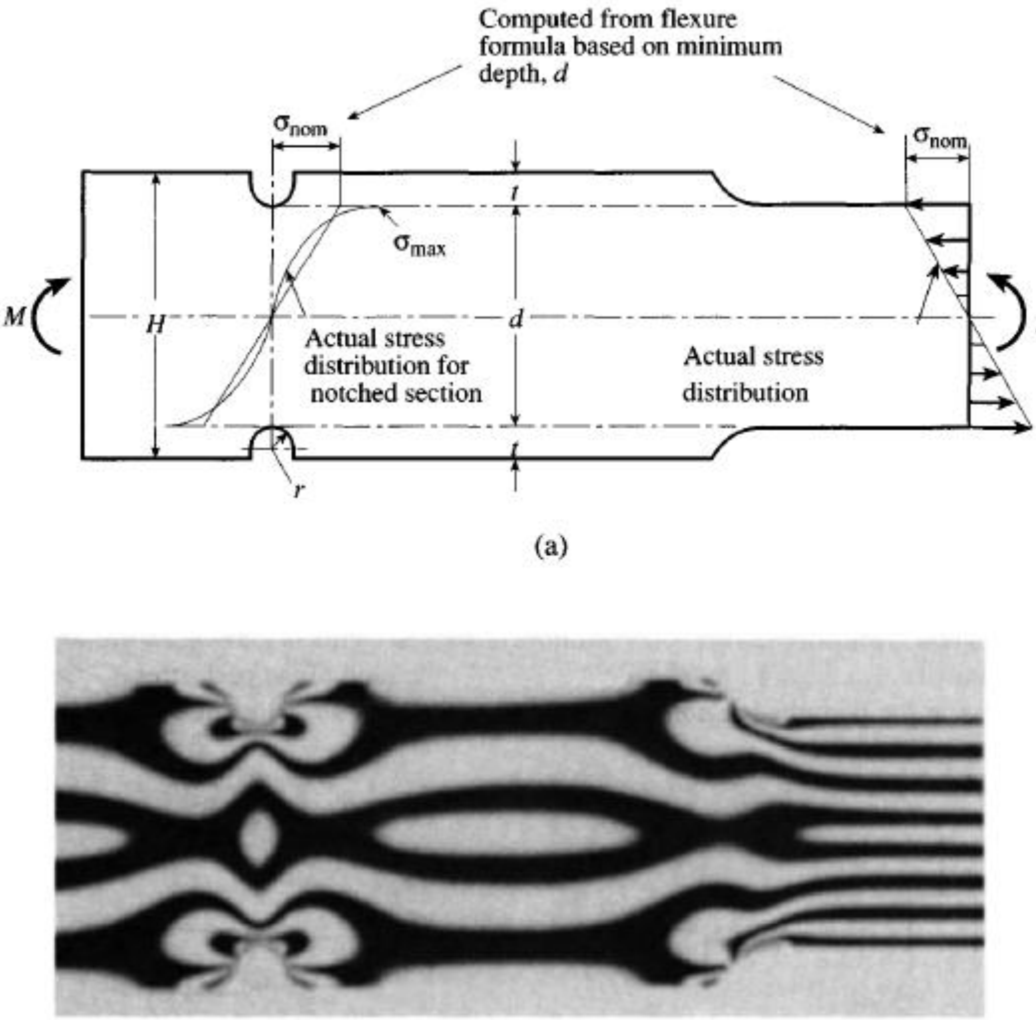


Figure 1.13 Elementary stress cases for specimens of constant cross section or with a gradual cross-sectional change: *(a)* Tension; *(b)* torsion; *(c)* bending [19].

Reference stress:

$K_t = \frac{\sigma_{max}}{\sigma_{nom}}$ for normal stress (tension or bending)

$K_{ts} = \frac{\tau_{max}}{\tau_{nom}}$ for shear stress (torsion)



(b)

Figure 1.14 Stress concentrations introduced by a notch and a cross-sectional change which is not gradual: *(a)* Bending of specimen; *(b)* photoelastic fringe photograph[19].

where the stresses σ_{max}, τ_{max} represent the maximum stresses to be expected in the member under the actual loads and the *nominal stresses* σ_{nom}, τ_{nom} are reference normal and shear stresses. The subscript t indicates that the stress concentration factor is a theoretical factor. That is to say, the peak stress in the body is based on the theory of elasticity, or it is derived from a laboratory stress analysis experiment [19].

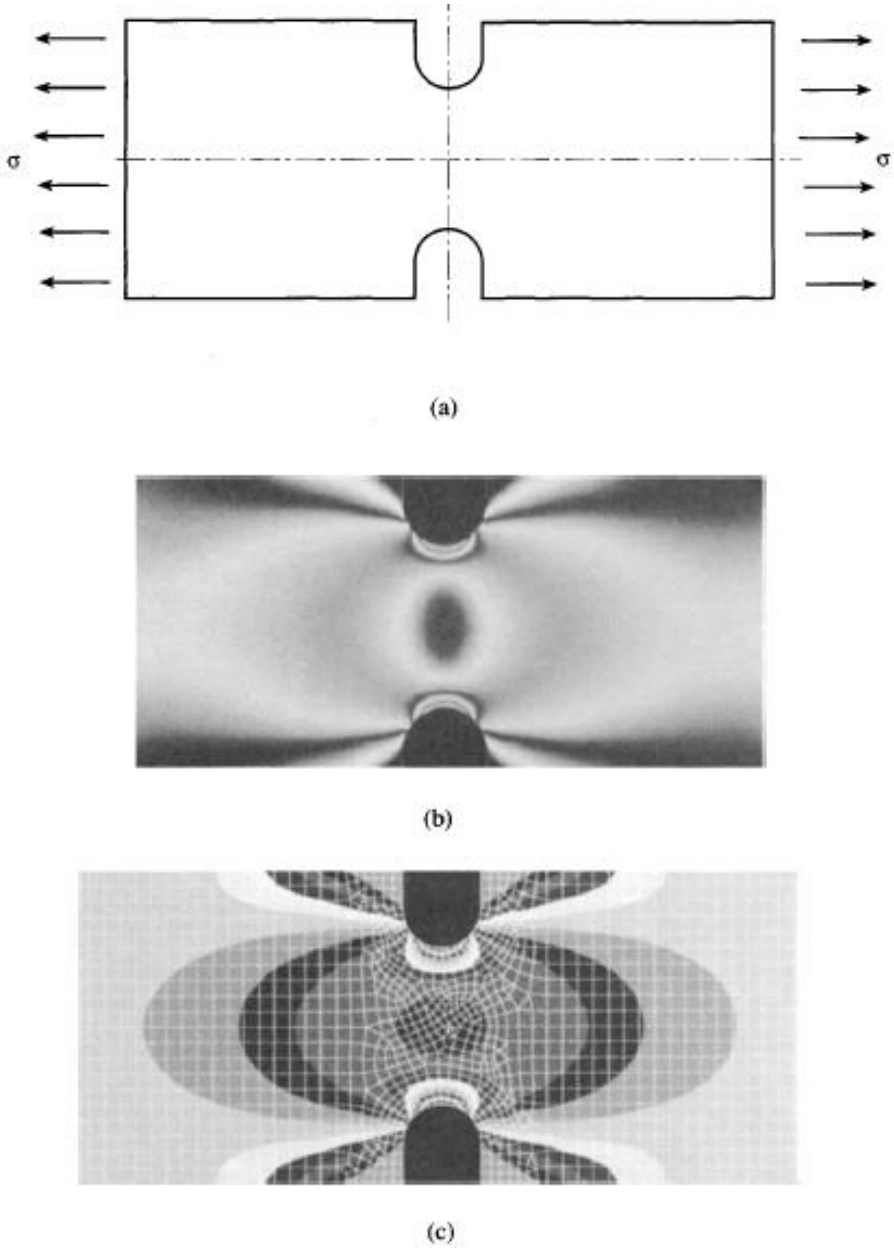


Figure 1.15 Tension bar with notches: *(a)* The specimen; *(b)* photoelastic fringe photograph (Doz. Dr.-Ing. habil. K. Fethke, Universitat Rostock); *(c)* finite element solution (Guy Nerad, University of Virginia) [19].

1.6.1. Selection of nominal stresses

The definitions of the reference stresses σ_{nom}, τ_{nom} depend on the problem at hand. It is very important to properly identify the reference stress for the stress concentration factor of interest. In this study the reference stress is usually defined at the same time that a particular stress concentration factor is presented. Consider an example to explain the selection of reference stresses.

Example – Tension Bar with Hole: Uniform tension is applied to a bar with a single circular hole, as shown in Figure 1.16. The maximum stress occurs at point A, and the stress distribution can be shown to be as in Figure 1.16. Suppose that the thickness of the plate is h, the width of the plate is H, and the diameter of the hole is d [19].

Use the stress based on the cross section at the hole, which is formed by removing the circular hole from the gross cross section. The corresponding area is referred to as the *net cross-sectional area*. If the stresses at this cross section are uniformly distributed and equal to σ_n:

$$\sigma_n = \frac{P}{(H-d)h}$$

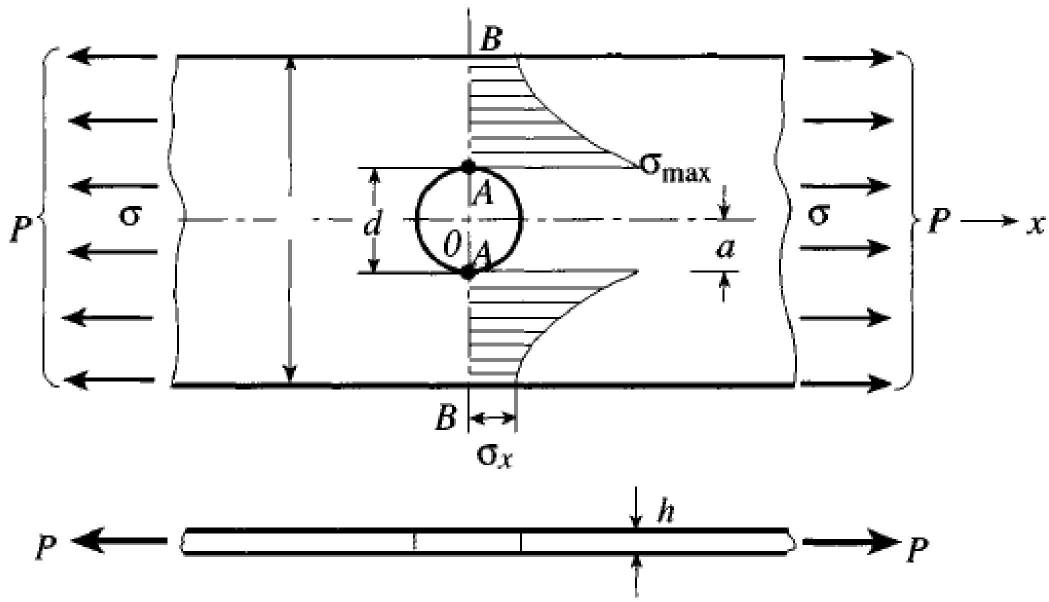


Figure 1.16 Tension Bar with Hole [19]

1.7. Airbus 321

The benchmark A320 Family's largest member – the A321 – offers airline customers the best seat-mile costs of any single-aisle aircraft and seating capacities comparable to that of a wide body jetliner.

Figure 1.17 Airbus 321 - Turkish Airlines

This aircraft has a stretched fuselage with an overall length of 44.51 metres, along with an extended operating range of up to 3,000 nautical miles while carrying a maximum passenger payload. Like each member in Airbus' A320 Family of jetliners, the A321 offers the lowest fuel burn, emissions and noise footprint in its class.

The A321 typically accommodates 185 passengers in a two-class configuration (16 in first class and 169 in economy) – while offering unbeatable economics in high-density seating (with up to 220 passengers) for charter and low-cost operators. The twin-engine A321 can be powered by either of two engine options: the CFM International CFM56 or International Aero Engines' V2500 [10].

Figure 1.18 The A321 is the A320 Family’s largest member [10]

1.8. Design for Long Service Life

Design for long service life is a compromise between maintenance and performance. Operators are interested primarily in an airplane which will have a good payload/range capability with moderate structural maintenance requirements.

Large commercial airplanes are designed for a minimum of 20 years of economic service life with a high level of reliability. This is known as the design service objective (DSO). It may be expressed in years, flight hours or flight cycles.

Some structural damage may occur prior to the DSO due to corrosion, fatigue, accidental damage, or environmental deterioration.

If an airplane is designed to be so robust that no structural maintenance is necessary, it will be too heavy to be economical to operate. Current Boeing designs are considered to be a good compromise between low maintenance and high economic service life.

Manufacturers have used the best analysis and test methods available to design and verify the airplane will have sufficient service life.

In the 1950s through the 1970s, airplane structure was analyzed and tested for static strength and fail-safety (multiple loads paths and backup structures). Starting in 1978, the FAA required damage tolerance analysis to be performed on all large commercial airplanes. The goal of damage tolerance is for the airplane to be able to safely operate with damage present until it can be found and repaired.

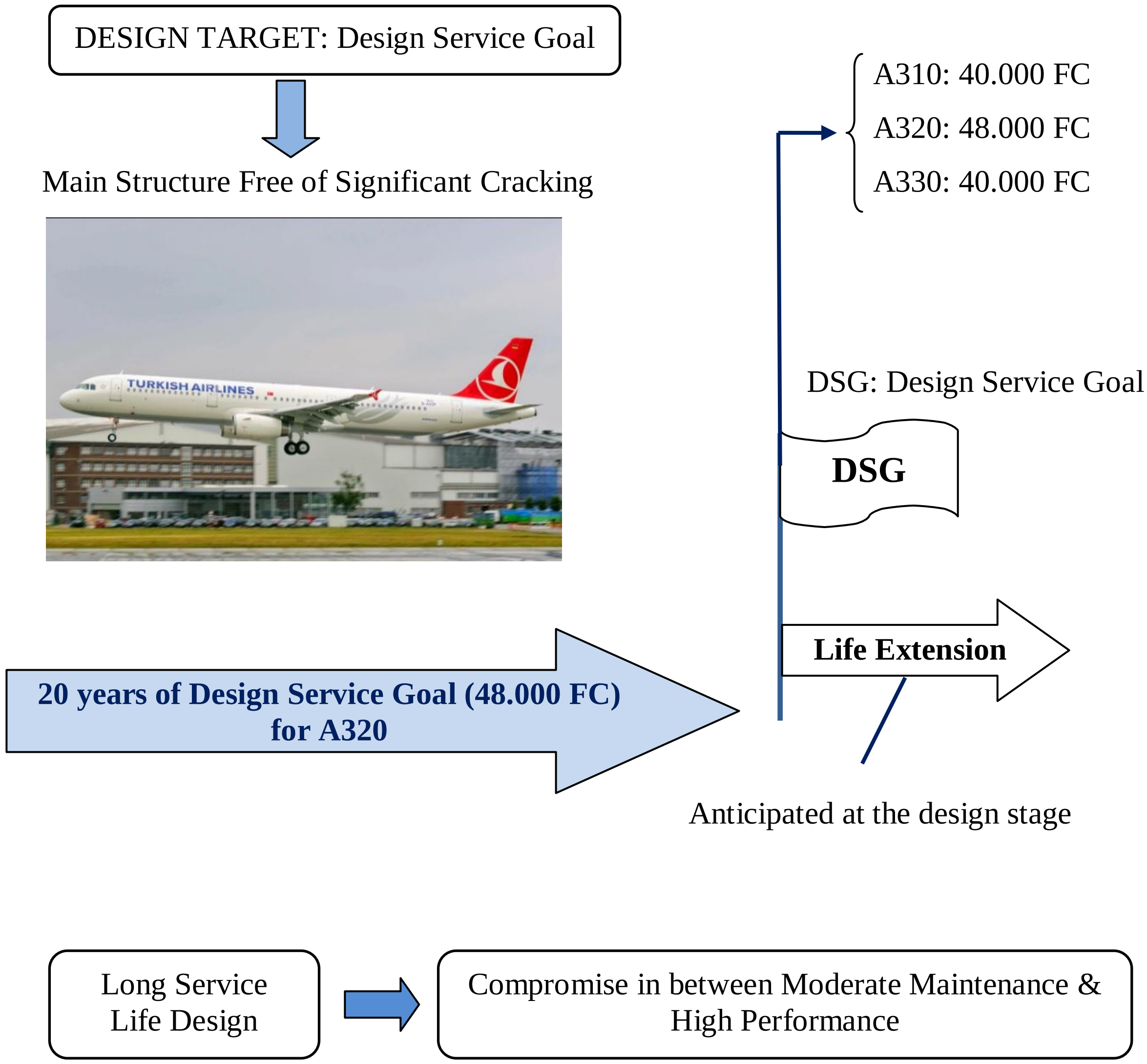


Figure 1.19 Service Life of A320 Family

2. FUSELAGE REPAIR

This topic contains allowable damage data for the fuselage - general. Allowable damage is a damage for which a structural repair is not necessary. You must remove the damage down to a smooth contour and compare with the allowable damage limits.

After rework the damaged area must be checked to make sure that the allowable limits have not been exceeded. When the limits are exceeded a repair is necessary. No allowable damage data for fuselage - general specified. In case of any damage contact producing companies such as Airbus or Boeing.

Figure 2.1 Contact with Airbus or Boeing

2.1. A321 - 231 Repair Location

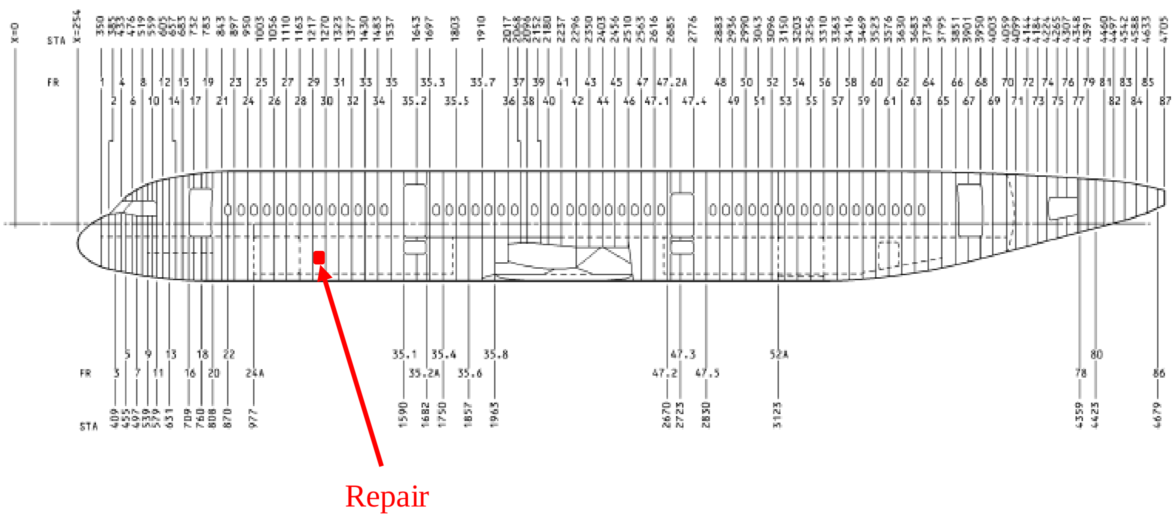


Figure 2.2 A321 - 231 Repair Location [2]

The current repair is located between Frame 29 – 30 & Stringer 35 Right Hand Side – 36 Right Hand Side, Figure 2.2 and Figure 2.3.

To identify the surrounding stringers, this illustration extracted from the Aircraft Maintenance Manuel (AMM) Chapter 06 can be used but will have to be confirmed during the detailed location step. Damage located between Stringers 35 Right Hand Side and 36 Right Hand Side).

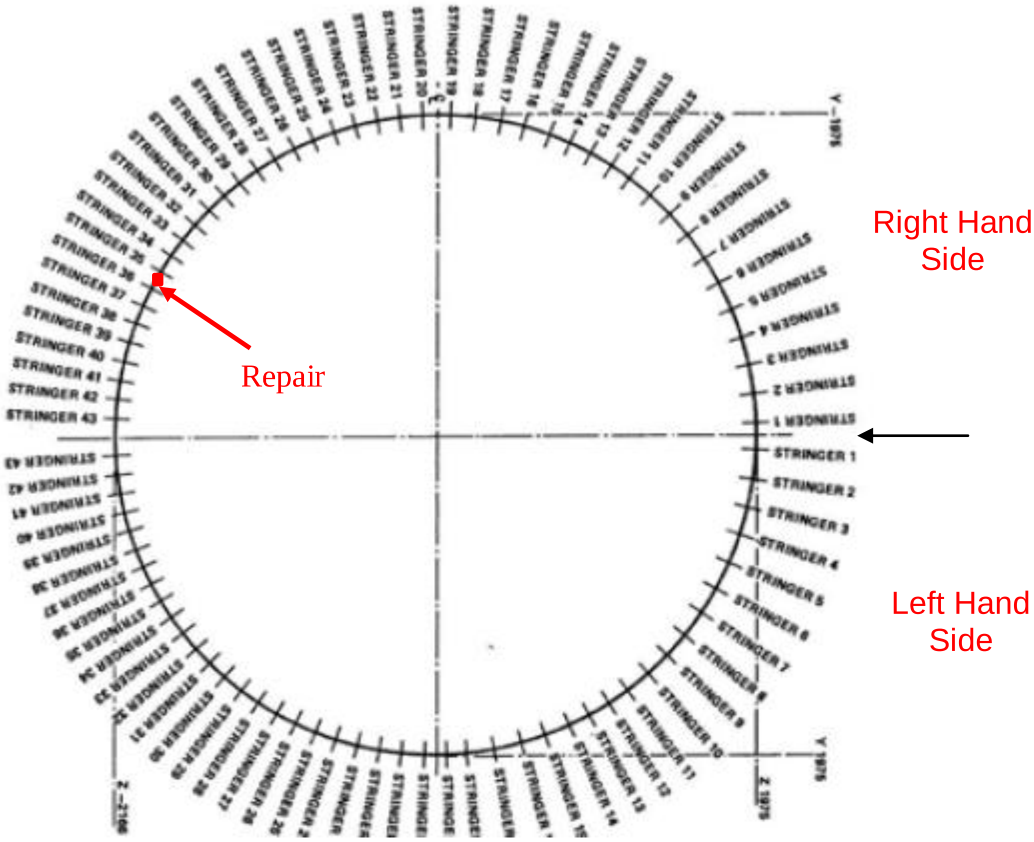


Figure 2.3 A321 - 231 Repair Location [20]

2.2. Skin Thickness

E region is between Frame 29 – 35 & Stringer 32 Right Hand Side – 41 Right Hand Side in Figure 2.4.

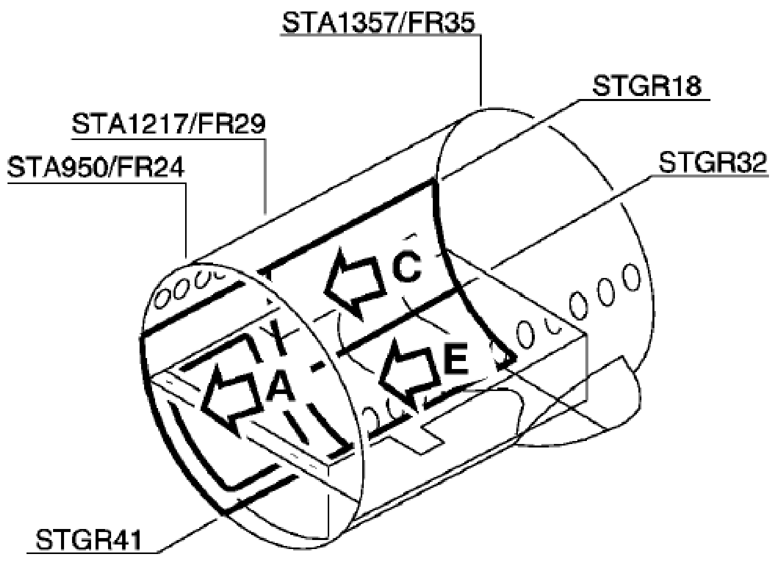


Figure 2.4 Skin Plates between Frame 29 – 35 & Stringer 32 Right Hand Side – 41 Right Hand Side

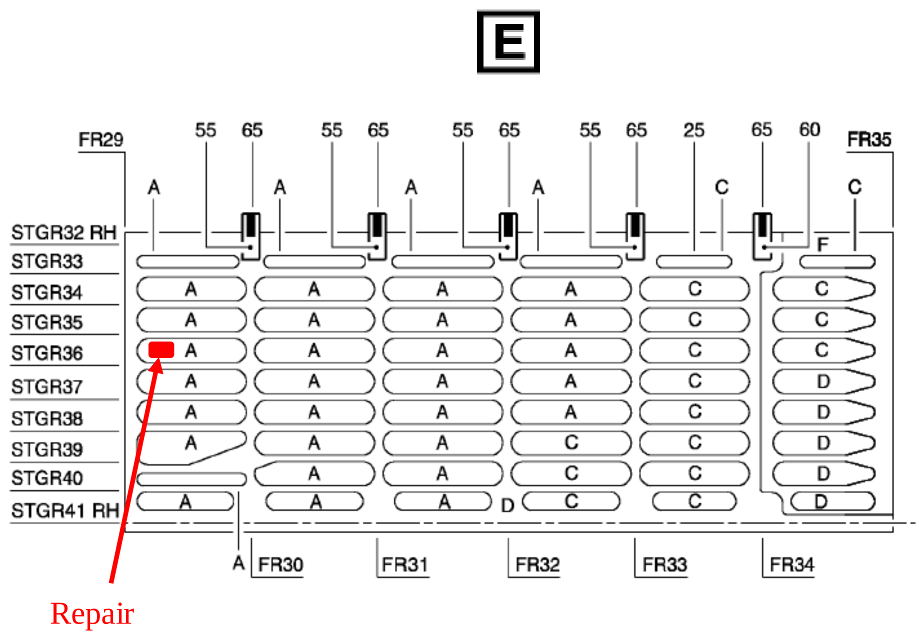


Figure 2.5 Skin Plates between Frame 29 – 35 & Stringer 32 Right Hand Side – 41 Right Hand Side

The code letters show the panel thickness of fuselage. As we look Code Table 2.1 for thickness, the repair area has A code and its thickness is 1.2 mm (0.047”).

Table 2.1. Thickness Code Table

CODE	THICKNESS mm (in.)
A	1.2 (0.047)
B	1.4 (0.055)
C	1.5 (0.059)
D	1.8 (0.071)
E	2 (0.079)
F	2.2 (0.087)
G	3.6 (0.142)
H	6 (0.236)

2.3. Skin Plates

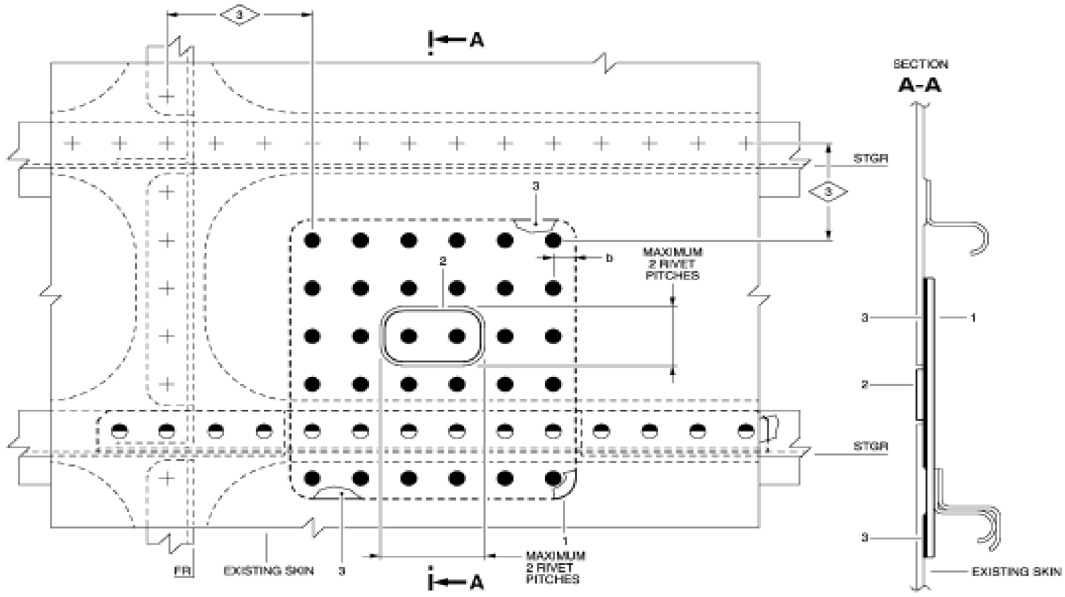
This topic contains repair procedures for the fuselage skin plates. The appropriate repairs are founded and are described in more detail for Frame 29 – 30 & Stringer 35 Right Hand Side – 36 Right Hand Side in the following text.

3. STRUCTURE

An instruction is indicated for small cut-out for skin thickness between 1.2 mm (0.047″) and 2.2 mm (0.087″) in SRM A321 - 231. It can be seen that SRM has 6 rows and 6 columns for a permanent repair. This repair is applicable for damage to the skin as 4 rows and 4 columns in this study for a fast and temporary repair. We have two repair options for a permanent repair in SRM. They can be external or internal doubler.

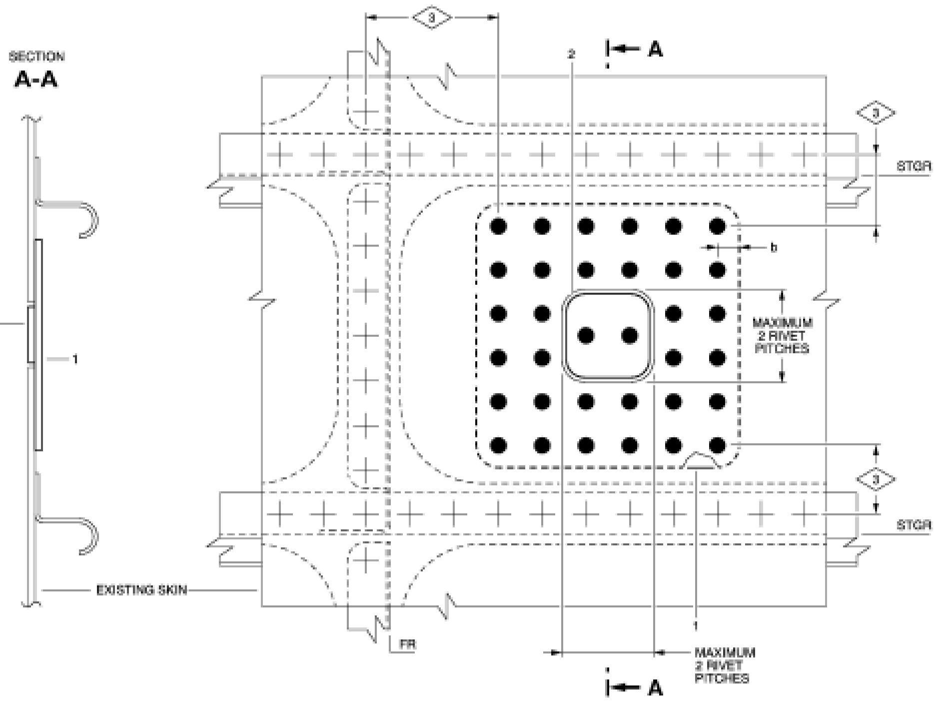
CAUTION: Hidden damage can lead to failure of the repair or surrounding structure.

3.1. Internal Doubler



REPAIR MATERIAL										
ITEM	NOMENCLATURE	MATERIAL	EXISTING SKIN 1.2 mm (0.047 in)	1.4 mm (0.055 in)	1.5 mm (0.059 in)	1.6 mm (0.063 in)	1.7 mm (0.067 in)	1.8 mm (0.071 in)	2 mm (0.079 in)	2.2 mm (0.087 in)
1	DOUBLER	CLAD2024T3	1.4 mm (0.056 in) ◇1	1.4 mm (0.056 in) ◇1	1.6 mm (0.063 in)		1.8 mm (0.071 in)		2 mm (0.08 in) ◇1	2.2 mm (0.09 in) ◇1
2	FILLER	CLAD2024T3	SAME THICKNESS AS EXISTING SKIN							
3	FILLER	CLAD2024T3	THICKNESS AS REQUIRED							
FASTENER SYMBOLS		+	REFERENCE ONLY							
		●	NAS1097DD5 ◇2 / NSA5410-32	NAS1097DD5			NAS1097DD6			
		◓	FASTENER IN ACCORDANCE WITH STRINGER REPAIR REFER TO CHAPTER 53-00-13, PAGEBLOCK 201							
MARGIN b			10 mm (0.394 in)							

Figure 3.1 Small Cut-out Internal Repair for Skin Thickness between 1.2 mm (0.047″) and 2.2 mm (0.087″) (Refer. to 53-00-11, Figure 221)



REPAIR MATERIAL										
ITEM	NOMENCLATURE	MATERIAL	EXISTING SKIN 1.2 mm (0.047 in)	1.4 mm (0.055 in)	1.5 mm (0.059 in)	1.6 mm (0.063 in)	1.7 mm (0.067 in)	1.8 mm (0.071 in)	2 mm (0.079 in)	2.2 mm (0.087 in)
1	DOUBLER	CLAD2024T3	1.4 mm (0.056 in) ⟨1⟩	1.4 mm (0.056 in) ⟨1⟩	1.6 mm (0.063 in)		1.8 mm (0.071 in)		2 mm (0.08 in) ⟨1⟩	2.2 mm (0.09 in) ⟨1⟩
2	FILLER	CLAD2024T3	SAME THICKNESS AS EXISTING SKIN							
FASTENER SYMBOLS		+	REFERENCE ONLY							
		●	NAS1097DD5 ⟨2⟩ NSA5410-32	NAS1097DD5			NAS1097DD6			
MARGIN b			10 mm (0.394 in)							

Figure 3.2 Small Cut-out Internal Repair for Skin Thickness between 1.2 mm (0.047″) and 2.2 mm (0.087″) (Refer. to 53-00-11, Figure 221)

CAUTION: This repair between Frame 1 through Frame 35, Frame 35.8 through Frame 47 above window line, Frame 47.5 through Frame 64 below window line and Frame 70 through Frame 78 must be inspected at defined intervals. The inspection instruction is described in the structural repair inspections section of the SRM. Inform your planning department and provide them with the necessary information.

CAUTION: This repair between Frame 47.5 through Frame 64 above window line is either permanent with associated repetitive inspections or temporary with or without repetitive inspections. This repair must be inspected at defined intervals.

NOTE: For the area between Frame 35 through Frame 86, this repair is superseded by Figure 226.

This repair is applicable for damage to the skin where the skin thickness is between 1.2 mm (0.047 in) and 2.2 mm (0.087 in) and is effective as follows:

- Frame 1 through Frame 35
- Frame 35.8 through Frame 86.

For excluded areas, refer to the relevant paragraph.

For applicability above Stringer 8 Left Hand Side through Stringer 8 Right Hand Side.

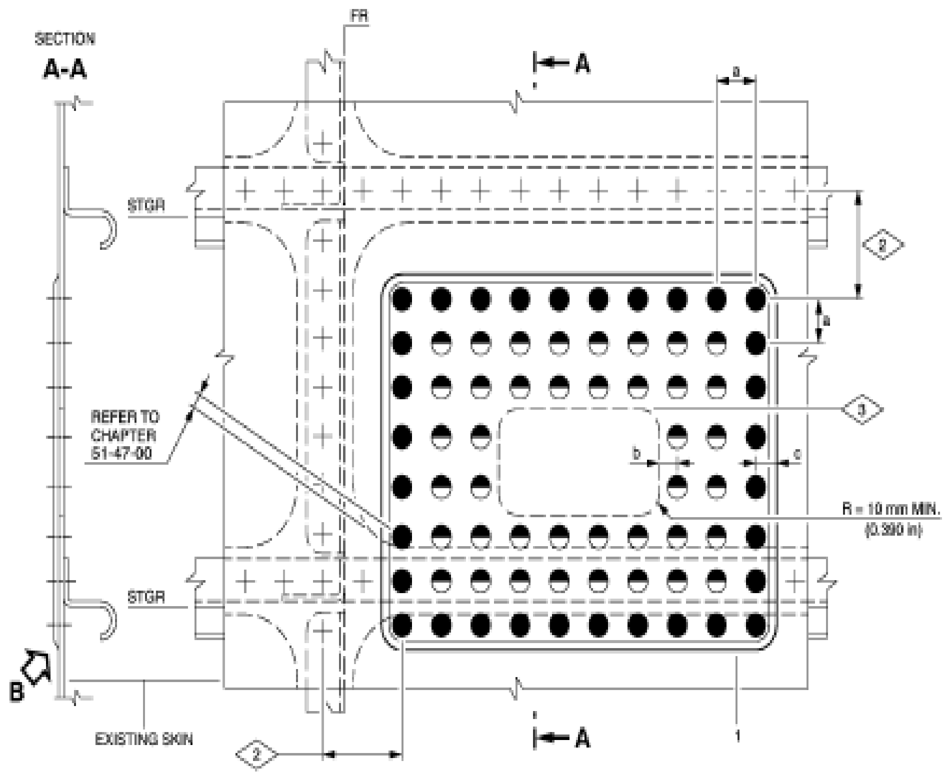
For repair solution with filler $thickness \geq 0.8\ mm\ (0.03\ in)$.

1 The mm (in) conversion for the material thickness corresponds to the US standard aluminum sheet metal gage and is not necessarily the exact conversion.

2 If the rivet head protrudes, mill it to make it flush with the skin.

3 Minimum 30 mm (1.181 in)

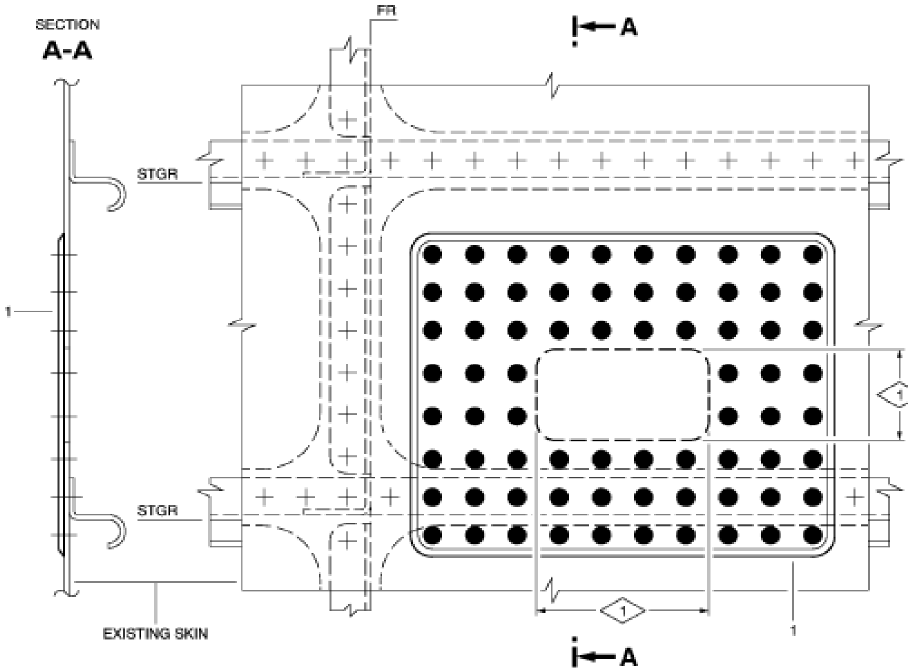
3.2. External Doubler



REPAIR MATERIAL							
ITEM	NOMEN-CLATURE	MATERIAL	EXISTING SKIN 4				
			1.2 mm (0.047 in)	> 1.4 mm (0.055 in) — 1.45 mm (0.057 in)	> 1.45 mm (0.057 in) — 1.65 mm (0.065 in)	> 1.65 mm (0.065 in) — 2.0 mm (0.079 in)	> 2.0 mm (0.079 in) — 2.2 mm (0.087 in)
1	DOUBLER 1	CLAD2024T3	1.4 mm (0.056 in)	1.6 mm (0.063 in)	1.8 mm (0.071 in)	2.0 mm (0.080 in)	2.2 mm (0.090 in)
FASTENER SYMBOLS							
+	REFERENCE ONLY						
●			NAS1097DD5				
◓			NAS1097DD5			NAS1097DD6	
PITCH a			ACCORDING TO EXISTING PITCH OR REFER TO CHAPTER 51-47-00				
MARGIN b			10 mm (0.39 in)				
MARGIN c			REFER TO CHAPTER 51-47-00				

Figure 3.3 Limited Cutout − External Repair for Skin Thickness between 1.2 mm (0.047 in) and 2.2 mm (0.087 in) − Skin between Stringers (Refer. to 53-21-11, Figure 218)

This repair is inactive and superseeded by Chapter 53-00-11 Page Block 201, Paragraph 4.W, Figure 238.



REPAIR MATERIAL								
ITEM	NOMENCLATURE	MATERIAL	EXISTING SKIN					
			1.2 mm (0.047 in)	1.4 mm - 1.45 mm (0.055 in - 0.057 in)	>1.45 mm - <1.6 mm (>0.057 in - <0.063 in)	1.6 mm - 1.65 mm (0.063 in - 0.065 in)	>1.65 mm - 2 mm (>0.065 in - 0.079 in)	>2 mm - 2.2 mm (>0.079 in - 0.087 in)
1	DOUBLER	CLAD2024T3	1.4 mm (0.056 in) 2	1.6 mm (0.063 in)	1.8 mm (0.071 in)		2 mm (0.08 in) 2	2.2 mm (0.09 in) 2
FASTENER SYMBOLS		+	REFERENCE ONLY					
		●	NAS1097DD5			NAS1097DD6		

Figure 3.4 Limited Cut−out − External Repair for Skin Thickness between 1.2 mm (0.047 in) and 2.2 mm (0.087 in) − Skin between Stringers, (Refer. to 53-00-11, Figure 238)

CAUTION: This repair between Frame 1 through Frame 35 and Frame 70 through Frame 78 must be inspected at defined intervals. The inspection instruction is described in the structural repair inspections section of the SRM. Inform your planning department and provide them with the necessary information.

CAUTION: This repair between Frame 35.8 through Frame 47 above window line and Frame 47.5 through Frame 64 is either permanent with associated repetitive inspections or temporary with or without repetitive inspections. This repair must be inspected at defined intervals. The inspection instruction is described in the structural repair inspections section of the SRM. Inform your planning department and provide them with the necessary information.

CAUTION: This repair procedure is not applicable after modification between Frame 21 through Frame 34, Stringer 5 Left Hand Side through Stringer 11 Left Hand Side and Stringer 5 Right Hand Side through Stringer 11 Right Hand Side.

NOTE: This repair is applicable for damage to the skin where the skin thickness is between 1.2 mm (0.047 in) and 2.2 mm (0.087 in) and is effective as follows:

- From Frame 1 through Frame 35
- From Frame 35.8 through Frame 47
- From Frame 47.5 through Frame 64
- From Frame 70 through Frame 86.

For excluded areas, refer to the relevant paragraph.

1 Cut-out in skin is limited to a length of half a frame bay and a width of one stringer bay.

2 The mm (in) conversion for the material thickness corresponds to the US standard Aluminum sheet metal gage and is not necessarily the exact conversion.

3.3. Repair Materials

- ✓ Doubler
- ✓ Filler
- ✓ Sealant (Interfay Sealant)
- ✓ Sealant (Corrosion Inhibiting Brush Consistency)
- ✓ Cleaning Agent (Methyl−Ethyl−Ketone)
- ✓ Storage Preservation (Corrosion Preventive Compound, Temporary)
- ✓ Structure Paint (Anti Corrosion Primer Polyurethane)

- ✓ Structure Paint (Polyurethane Top Coat, Grey)
- ✓ Structure Paint (Flexible Polyurethane)
- ✓ Structure Paint (External Top Coat)
- ✓ Structure Paint (Wash Primer)
- ✓ Chromic Acid Anodizing (CAA)

3.4. Safety Precautions

WARNING: Be careful when you use consumable materials. Obey the material manufacturer's instructions and your local regulations.

WARNING: Obey the manufacturer's instructions when you use cleaning agents, bonding and adhesive compound, sealant, special material and structure paint. These materials are dangerous.

CAUTION: Hidden damage can lead to failure of the repair or surrounding structure.

CAUTION: Repairs applicable in areas around static ports, pitot probes, total air temperature probes and angle of attack probes, have to comply with aerodynamic smoothness requirements.

CAUTION: Use only specified cleaning materials and solutions or their equivalents. The surface protection could be damaged if unspecified materials are used. It is important that the manufacturer's mixing, application and treatment instructions are followed.

CAUTION: There must be a minimum distance of four fastener spacings between the outer rivet/fastener rows of adjacent repairs and any existing doublers.

CAUTION: There must be a minimum distance of four fastener spacings between the outer rivet/fastener row of the doubler and any existing cutout (e.g. door/window).

CAUTION: There must be a minimum distance of three fastener spacings between the outer rivet/fastener row of the doubler to the first rivet/fastener row of a longitudinal or circumferential joint. In case this distance cannot be maintained, refer to the instructions for the relevant SRM joint repair scheme.

CAUTION: To prevent damage to the surface protection, mechanical and electrical systems, the area surrounding the repair must be covered with plastic foil and masking tape.

CAUTION: Obey the given inspection instruction reference which leads to the applicable inspection program defined in the structural repair inspections, if necessary.

These precautions are taken into account in the following calculations.

3.5. Repair Instructions

(a) Mark the repair area

<u>*CAUTION:*</u> Make sure not to damage internal structure or system when drilling.

(b) Remove the existing fasteners in the repair area.

<u>*CAUTION:*</u> A metal sheet must be placed underneath the cutting area to avoid damage to the structure.

(c) Cut out the damaged skin area and deburr the edges of the cutout.

(d) Clean the edges of the cutout with cleaning agent (methyl−ethyl−ketone).

(e) Treat the edges of the cutout with structure paint (wash primer).

(f) Apply structure paint (anti corrosion primer polyurethane) to the edges of the cutout.

(g) Prepare the repair part(s).

<u>*NOTE:*</u> Cut the doubler to oversize and pre−shape it to match the fuselage contour to avoid stressing.

(h) Put the repair part(s) in position, transfer drill the existing fastener holes and drill the additional holes, if necessary.

<u>*NOTE:*</u> Rivet pitch and margin data are given the following topic.

(i) Remove the repair parts from the repair area and deburr all the fastener holes.

(j) Cut the doubler to size and deburr.

(k) Chamfer the doubler.

<u>*WARNING:*</u> Chromic acid anodizing (CAA) is dangerous.

(l) Clean the repair parts and repair area with cleaning agent (methyl−ethyl−ketone).

(m) Apply structure paint (anti corrosion primer polyurethane) to the repair parts.

(n) Apply sealant (corrosion inhibiting brush consistency) to the mating surfaces of the repair parts.

(o) Fill existing countersinks in fuselage skin with countersunk repair washers if necessary.

(p) Put the repair parts in position and install the fasteners wet with sealant (corrosion inhibiting brush consistency).

<u>*NOTE:*</u> Make sure that the installation process is completed, or, at least one fastener is set in every third hole, during the curing time of the sealant.

(q) Seal all the edges of the doubler with sealant (corrosion inhibiting brush consistency).

(r) Apply structure paint (flexible polyurethane) to the sealant bead.

(s) Apply structure paint (polyurethane top coat, grey) to the internal surface of the repair area.

(t) Apply structure paint (external top coat) to the external surface of the repair area to match the surrounding structure.

WARNING: Do not apply storage preservation in area of oxygen equipment and oxygen pipes.

(v) Apply storage preservation (corrosion preventive compound, temporary) or equivalent to the repair area inside the lower shell area.

3.6. Pitch Value

Rivet spacing is measured between the centerlines of rivets in the same row. On most repairs, the general practice is to use the same rivet spacing and edge distance (distance from the center of the hole to the edge of the material) that the manufacturer used in the area surrounding the damage. The SRM for the particular aircraft may also be consulted [14].

Table 3.1 Pitch Data for Installation of Pins and Bolts in Aluminum Alloy Assemblies (Refer. to 51-47-00, Table 2)

NOMINAL FASTENER DIAMETER		MINIMUM PITCH VALUE			MAXIMUM PITCH VALUE		
'D'mm	'D'in.	Factor	mm	in.	Factor	mm	in.
4.0	0.1560	4.0 D	16.000	0.624	5.0 D	20.000	0.780
4.8	0.1900	4.0 D	19.200	0.760	5.0 D	24.000	0.950
5.6	0.2205	4.0 D	22.400	0.882	5.0 D	28.000	1.102
6.4	0.2500	4.0 D	25.600	1.000	5.0 D	32.000	1.250
7.9	0.3125	3.875 D	30.613	1.211	5.0 D	39.500	1.563
9.5	0.3750	3.750 D	35.625	1.406	5.0 D	47.500	1.875

The minimum spacing between blind bolts [diameter = $(5/32)'' = 0.1560$"] shall not be less than 4 times the diameter of the blind bolt and more than 5 times the diameter of the blind bolt (Table 3.1).

$$\text{Min. Pitch Value} = 4 * \text{D}$$
$$\text{Max. Pitch Value} = 5 * \text{D} \quad \text{for } (5/32)'' = 0.1560'' = 4\,mm$$

3.7. Edge Distance

Edge distance, also called edge margin by some manufacturers, is the distance from the center of the first rivet to the edge of the sheet.

If rivets are placed too close to the edge of the sheet, the sheet may crack or pull away from the rivets. If they are spaced too far from the edge, the sheet is likely to turn up at the edges (Figure 2-5) [14].

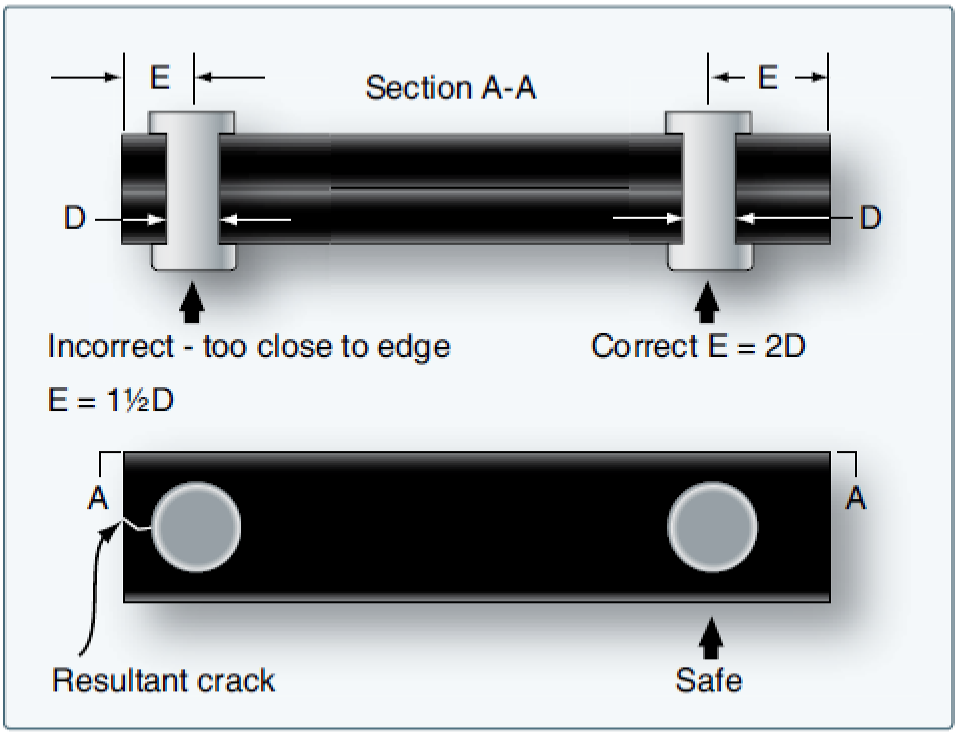


Figure 3.5 Minimum Edge Distance [14]

Table 3.2 Edge Distance Values for Installation of Pins and Bolts in Aluminum Alloy Assemblies (Refer. to 51-47-00, Table 3)

NOMINAL FASTENER DIAMETER		E (for protruding head fastener) <1>		E1 (for countersunk head fastener) <1>		B min	
mm	in.	mm	in.	mm	in.	mm	in.
4.0	5/32	7.0	0.276	8.0	0.315	6.0	0.236
4.8	3/16	9.0	0.354	10.0	0.394	7.0	0.276
5.6	7/32	9.5	0.374	10.5	0.413	8.0	0.315
6.4	1/4	10.0	0.394	11.0	0.433	9.0	0.354
7.9	5/16	12.0	0.472	13.0	0.512	10.0	0.394

The minimum edge distance of a blind bolt [diameter = $(5/32)'' = 0.1560$"] shall not be less than 0.276” $= 7\ mm$ (Table 3.2).

4. MATERIAL

The Aluminum Clad 2024 – T3 skin is 1.2 mm (0.047″) thick, however, doubler used for the repair is 1.4 mm (0.056″) thick made of the same material as skin.

Table 4.1 Material Properties of Aluminum Clad 2024 – T3 [1]

Specification	AMS-QQ-A-250/5ª							
Form	Flat sheet							
Temper	T3							
Thickness, in.	0.008-0.009		0.010-0.062		0.063-0.128		0.129-0.249	
Basis	A	B	A	B	A	B	A	B
Mechanical Properties:								
F_{tu}, ksi:								
L	59	60	60	61	62	63	63	64
LT	58	59	59	60	61	62	62	63
F_{ty}, ksi:								
L	44	45	44	45	45	47	45	47
LT	39	40	39	40	40	42	40	42
F_{cy}, ksi:								
L	36	37	36	37	37	39	37	39
LT	42	43	42	43	43	45	43	45
ST	...	...	...	...	...	...	...	...
F_{su}[b], ksi	37	37	37	38	38	39	39	40
F_{bru}[b,c], ksi:								
(e/D = 1.5)	96	97	97	99	101	102	102	104
(e/D = 2.0)	119	121	121	123	125	127	127	129
F_{bry}[b,c], ksi:								
(e/D = 1.5)	68	70	68	70	70	73	70	73
(e/D = 2.0)	82	84	82	84	84	88	84	88
e, percent:								
LT	10	...	[d]	...	15	...	15	...
E, 10^3 ksi:								
Primary	10.5							
Secondary	9.5				10.0			
E_c, 10^3 ksi:								
Primary	10.7							
Secondary	9.7				10.2			
G, 10^3 ksi	...							
μ	0.33							
Physical Properties:								
ω, lb/in.^{3}	0.100							
C, K, and α	...							

'A' basis values used in the L_t material direction except for F_{cy} where the L material direction has been used due to the lower value.

$$F_{tu} = Allowable\ Tensile\ Stress = 61000\ psi$$

$$F_{su} = Allowable\ Ultimate\ Stress\ in\ Pure\ Shear = 38000\ psi$$

5. STRUCTURAL FASTENERS

Structural fasteners, used to join sheet metal structures securely, come in thousands of shapes and sizes with many of them specialized and specific to certain aircraft. Since some structural fasteners are common to all aircraft, this section focuses on the more frequently used fasteners. For the purposes of this discussion, fasteners are divided into two main groups: solid shank rivets and special purpose fasteners that include blind rivets [14].

5.1. Solid Shank Rivet

The solid shank rivet is the most common type of rivet used in aircraft construction. Used to join aircraft structures, solid shank rivets are one of the oldest and most reliable types of fastener. Widely used in the aircraft manufacturing industry, solid shank rivets are relatively low-cost, permanently installed fasteners. They are faster to install than bolts and nuts since they adapt well to automatic, high-speed installation tools. Rivets should not be used in thick materials or in tensile applications, as their tensile strengths are quite low relative to their shear strength. The longer the total grip length (the total thickness of sheets being joined), the more difficult it becomes to lock the rivet.

Riveted joints are neither airtight nor watertight unless special seals or coatings are used. Since rivets are permanently installed, they must be removed by drilling them out, a laborious task.

Before installation, the rivet consists of a smooth cylindrical shaft with a factory head on one end. The opposite end is called the bucktail. To secure two or more pieces of sheet metal together, the rivet is placed into a hole cut just a bit larger in diameter than the rivet itself. Once placed in this predrilled hole, the bucktail is upset or deformed by any of several methods from hand-held hammers to pneumatically driven squeezing tools. This action causes the rivet to expand about 11⁄2 times the original shaft diameter, forming a second head that firmly holds the material in place [14].

5.1.1. Rivet head shape

Solid rivets are available in several head shapes, but the universal and the 100° countersunk head are the most commonly used in aircraft structures. Universal head rivets were developed specifically for the aircraft industry and designed as a replacement for both the round and brazier head rivets. These rivets replaced all protruding head rivets and are used primarily where the protruding head has no aerodynamic significant. They have a flat area on the head, a head diameter twice the shank diameter, and a head height approximately 42.5 percent of the shank diameter (Figure 5.1).

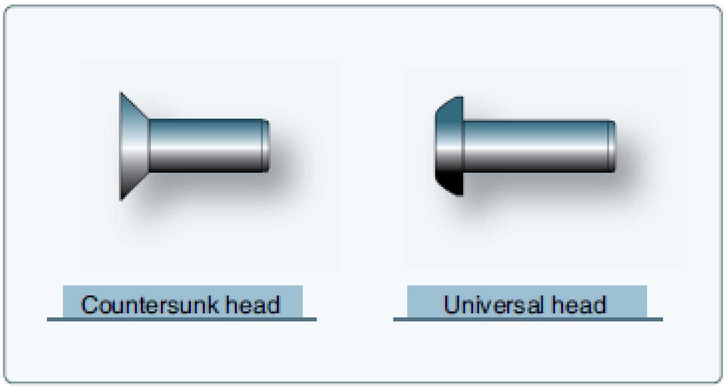


Figure 5.1 Solid Shank Rivet Styles [14]

The countersunk head angle can vary from 60° to 120°, but the 100° has been adopted as standard because this head style provides the best possible compromise between tension/shear strength and flushness requirements. This rivet is used where flushness is required because the rivet is flat-topped and undercut to allow the head to fit into a countersunk or dimpled hole. The countersunk rivet is primarily intended for use when aerodynamics smoothness is critical, such as on the external surface of a high-speed aircraft.

Typically, rivets are fabricated from aluminum alloys, such as 2017-T4, 2024-T4, 2117-T4, 7050, and 5056. Titanium, nickel-based alloys, such as Monel® (corrosion-resistant steel), mild steel or iron, and copper rivets are also used for rivets in certain cases.

Rivets are available in a wide variety of alloys, head shapes, and sizes and have a wide variety of uses in aircraft structure. Rivets that are satisfactory for one part of the aircraft are often unsatisfactory for another part. Therefore, it is important that an aircraft technician know the strength and driving properties of the various types of rivets and how to identify them, as well as how to drive or install them.

Solid rivets are classified by their head shape, by the material from which they are manufactured, and by their size. Identification codes used are derived from a combination of the Military Standard (MS) and National Aerospace Standard (NAS) systems, as well as an older classification system known as AN for Army/Navy. For example, the prefix MS identifies hardware that conforms to written military standards. A letter or letters following the head-shaped code identify the material or alloy from which the rivet was made. The alloy code is followed by two numbers separated by a dash. The first number is the numerator of a fraction, which specifies the shank diameter in thirty-seconds of an inch. The second number is the numerator of a fraction in sixteenths of an inch and identifies the length of the rivet. Rivet head shapes and their identifying code numbers are shown in Figure 5.2.

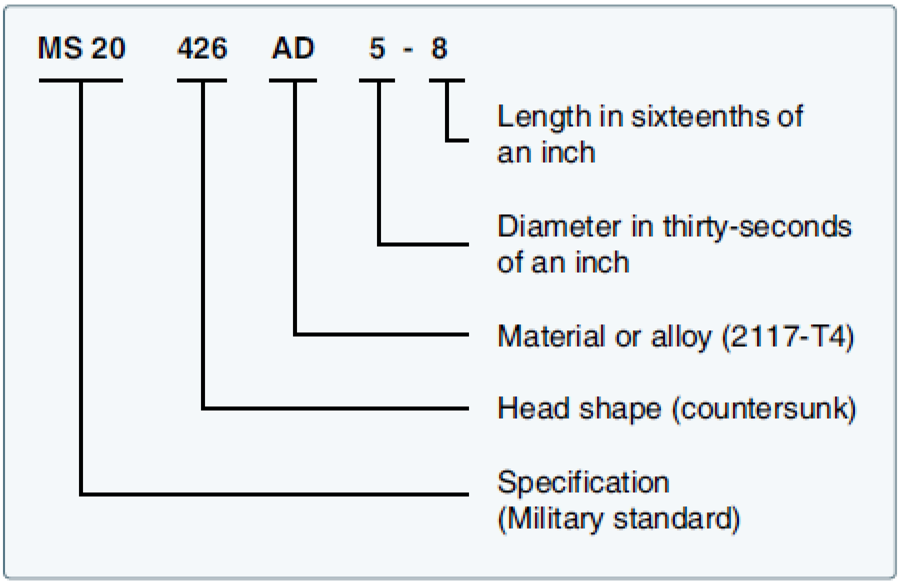


Figure 5.2 Rivet Head Shapes and Their Identifying Code Numbers [14]

The most frequently used repair rivet is the AD rivet because it can be installed in the received condition. Some rivet alloys, such as DD rivets (alloy 2024-T4), are too hard to drive in the received condition and must be annealed before they can be installed. Typically, these rivets are annealed and stored in a freezer to retard hardening, which has led to the nickname “ice box rivets.” They are removed from the freezer just prior to use. Most DD rivets have been replaced by E-type rivets which can be installed in the received condition.

The head type, size, and strength required in a rivet are governed by such factors as the kind of forces present at the point riveted, the kind and thickness of the material to be riveted, and the location of the part on the aircraft. The type of head needed for a particular job is determined by where it is to be installed. Countersunk head rivets should be used where a smooth aerodynamic surface is required. Universal head rivets may be used in most other areas.

The size (or diameter) of the selected rivet shank should correspond in general to the thickness of the material being riveted. If an excessively large rivet is used in a thin material, the force necessary to drive the rivet properly causes an undesirable bulging around the rivet head. On the other hand, if an excessively small rivet diameter is selected for thick material, the shear strength of the rivet is not great enough to carry the load of the joint. As a general rule, the rivet diameter should be at least two and a half to three times the thickness of the thicker sheet. Rivets most commonly chosen in the assembly and repair of aircraft range from 3⁄32-inch to 3⁄8 inch in diameter. Ordinarily, rivets smaller than 3⁄32-inch in diameter are never used on any structural parts that carry stresses.

The proper sized rivets to use for any repair can also be determined by referring to the rivets (used by the manufacturer) in the next parallel row inboard on the wing or forward on the fuselage. Another method of determining the size of rivets to be used is to multiply the skin's thickness by 3 and use the next larger size rivet corresponding to that figure. For example, if the skin is 0.040 inch thick, multiply 0.040 inch by 3 to get 0.120 inch and use the next larger size of rivet, 1⁄8-inch (0.125 inch).

When rivets are to pass completely through tubular members, select a rivet diameter equivalent to at least 1⁄8 the outside diameter of the tube. If one tube sleeves or fits over another, take the outside diameter of the outside tube and use oneeighth of that distance as the minimum rivet diameter. A good practice is to calculate the minimum rivet diameter and then use the next larger size rivet.

Whenever possible, select rivets of the same alloy number as the material being riveted. For example, use 1100 and 3003 rivets on parts fabricated from 1100 and 3003 alloys, and 2117-1 and 2017-T rivets on parts fabricated from 2017 and 2024 alloys.

The size of the formed head is the visual standard of a proper rivet installation. The minimum and maximum sizes, as well as the ideal size, are shown in Figure 5.3 [14].

5.1.2. Installation of rivets

5.1.2.1. Repair layout

Repair layout involves determining the number of rivets required, the proper size and style of rivets to be used, their material, temper condition and strength, the size of the holes, the distances between the holes, and the distance between the holes and the edges of the patch. Distances are measured in terms of rivet diameter.

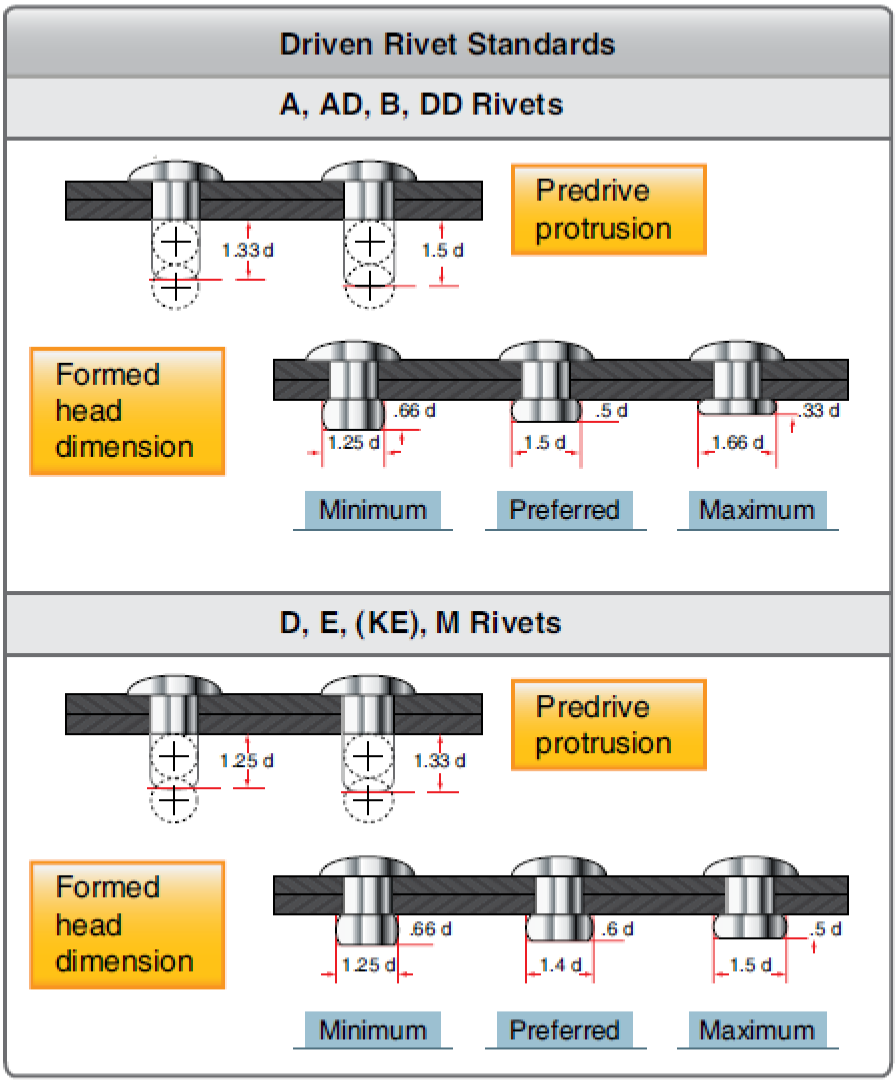


Figure 5.3 Rivet Formed Head Dimensions [14]

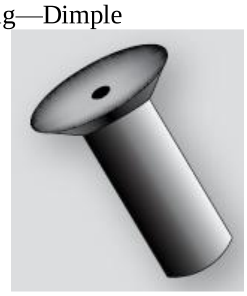
5.1.2.2. Rivet length

To determine the total length of a rivet to be installed, the combined thickness of the materials to be joined must first be known. This measurement is known as the grip length. The total length of the rivet equals the grip length plus the amount of rivet shank needed to form a proper shop head. The latter equals one and a half times the diameter of the rivet shank. Where A is total rivet length, B is grip length, and C is the length of the material needed to form a shop head, this formula can be represented as A = B + C (Figure 5.3).

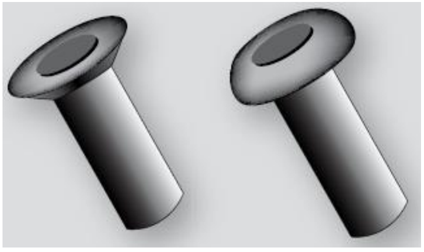
5.1.2.3. Rivet strength

For structural applications, the strength of the replacement rivets is of primary importance (Table 5.1). Rivets made of material that is lower in strength should not be used as replacements unless the shortfall is made up by using a larger rivet.

Table 5.1 Rivet Allow Strength [14]

Standard Rivet Alloy Code Markings	
Alloy code—A Alloy—1100 or 3003 aluminum Head marking—None  Shear strength—10 KSI Nonstructural uses only	Alloy code—B Alloy—5056 aluminum Head marking—Raised cross  Shear strength—28 KSI
Alloy code—AD Alloy—2117 aluminum Head marking—Dimple  Shear strength—30 KSI	Alloy code—D Alloy—2017 aluminum Head marking—Raised dot  Shear strength—38 KSI 38 KSI When driven as received 34 KSI When re-heat treated
Alloy code—DD Alloy—2024 aluminum Head marking—Two bars (raised)  Shear strength—41 KSI Must be driven in "W" condition (Ice-Box)	Alloy code—E, [KE*] *Boeing code Alloy—2017 aluminum Head marking—Raised Ring  Shear strength—43 KSI Replacement for DD rivet to be driven in "T" condition
Alloy code—E	

Alloy—7050 aluminum
Head marking—raised circle



Shear strength—54 KSI

For example, a rivet of 2024-T4 aluminum alloy should not be replaced with one of 2117-T4 or 2017-T4 aluminum alloy unless the next larger size is used.

The 2117-T rivet is used for general repair work, since it requires no heat treatment, is fairly soft and strong, and is highly corrosion resistant when used with most types of alloys. Always consult the maintenance manual for correct rivet type and material.

The type of rivet head to select for a particular repair job can be determined by referring to the type used within the surrounding area by the manufacturer. A general rule to follow on a flush-riveted aircraft is to apply flush rivets on the upper surface of the wing and stabilizers, on the lower leading edge back to the spar, and on the fuselage back to the high point of the wing. Use universal head rivets in all other surface areas. Whenever possible, select rivets of the same alloy number as the material being riveted [14].

5.2. Blind Rivets

The first blind fasteners were introduced in 1940 by the Cherry Rivet Company (now Cherry® Aerospace), and the aviation industry quickly adopted them. The past decades have seen a proliferation of blind fastening systems based on the original concept, which consists of a tubular rivet with a fixed head and a hollow sleeve. Inserted within the rivet’s core is a stem that is enlarged or serrated on its exposed end when activated by a pulling-type rivet gun. The lower end of the stem extends beyond the inner sheet of metal. This portion contains a tapered joining portion and a blind head that has a larger diameter than the stem or the sleeve of the tubular rivet.

When the pulling force of the rivet gun forces the blind head upward into the sleeve, its stem upsets or expands the lower end of the sleeve into a tail. This presses the inner sheet upward and closes any space that might have existed between it and the outer sheet. Since the exposed head of the rivet is held tightly against the outer sheet by the rivet gun, the sheets of metal are clamped, or clinched, together [14].

5.2.1. Blind bolts

Bolts are threaded fasteners that support loads through predrilled holes. Hex, close-tolerance, and internal wrenching bolts are used in aircraft structural applications. Blind bolts have a higher strength than blind rivets and are used for joints that require high strength. Sometimes, these bolts can be direct replacements for the Hi-Lok® and lockbolt. Many of the new generation blind bolts are made from titanium and rated at 90 KSI shear strength, which is twice as much as most blind rivets.

Determining the correct length of the fastener is critical to correct installation. The grip length of a bolt is the distance from the underhead bearing surface to the first thread. The grip is the total thickness of material joined by the bolt. Ideally, the grip length should be a few thousands of an inch less than the actual grip to avoid bottoming the nut. Special grip gauges are inserted in the hole to determine the length of the blind bolt to be used. Every blind bolt system has its own grip gauge and is not interchangeable with other blind bolt or rivet systems.

Blind bolts are difficult to remove due to the hardness of the core bolt. A special removal kit is available from the manufacturer for removing each type of blind bolt. These kits make it easier to remove the blind bolt without damaging the hole and parent structure. Blind bolts are available in a pull type and a drive type.

5.2.1.1. Pull-type blind bolt

Several companies manufacture the pull-type of blind bolt fastening systems. They may differ in some design aspects, but in general they have a similar function. The pull-type uses the drive nut concept and is composed of a nut, sleeve, and a draw bolt. Frequently used blind bolt systems include but are not limited to the Cherry Maxibolt® Blind Bolt system and the HuckBolt® fasteners which includes the Ti-Matic® Blind Bolt and the Unimatic® Advanced Bolt (UAB) blind bolt systems.

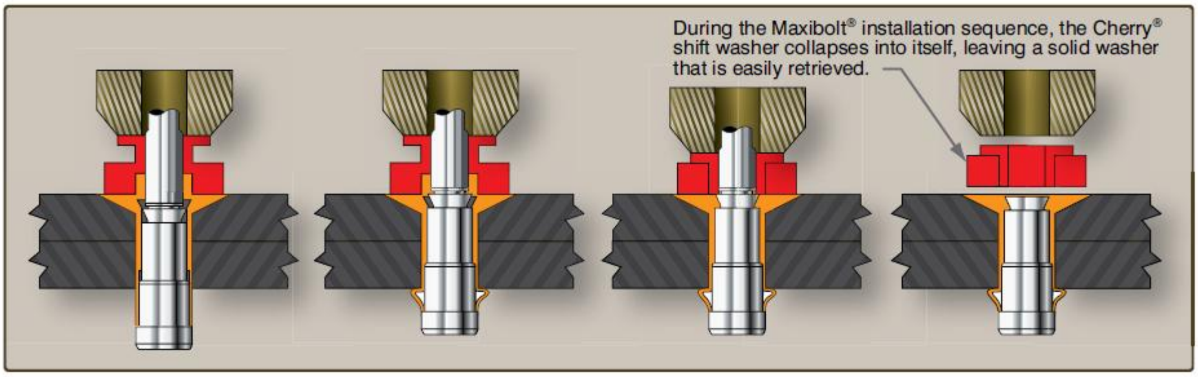


Figure 5.4 Blind Bolt System Installation [14]

Blind bolt installation is easier and fast than solid rivet, so blind bolt is more convenient for fast repair. In the new repair design, pull – type blind bolts (MS21141 – 0502, MS21141 – 0602 and MS90354 – 0502) are used instead of solid rivets (NAS1097DD5).

5.2.1.2. Cherry Maxibolt® blind bolt system

The Cherry Maxibolt® blind bolt, available in alloy steel and A-286 CRES materials, comes in four different nominal and oversized head styles (Figure 5.4). One tool and pulling head installs all three diameters. The blind bolts create a larger blind side footprint and they provide excellent performance in thin sheet and nonmetallic applications. The flush breaking stem eliminates shaving while the extended grip range accommodates different application thicknesses. Cherry Maxibolts® are primarily used in structures where higher loads are required. The steel version is 112 KSI shear. The A286 version is 95 KSI shear. The Cherry® G83, G84, or G704 installation tools are required for installation.

5.3. Fasteners in Available Design

NAS1097DD5 fasteners have to be used for the permanent repair in SRM.

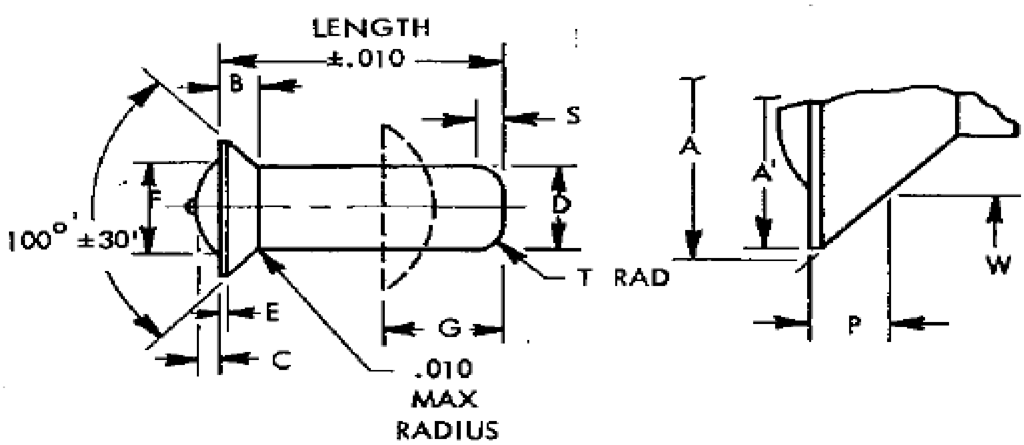


Figure 5.5 NAS1097DD5 $(5/32\,'')$ Countersinking Head Solid Rivet [3]

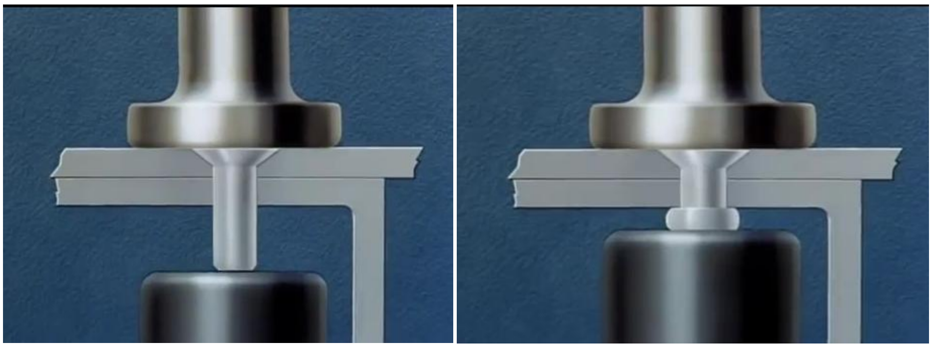


Figure 5.6 Solid Rivet Installation

In the available design, solid rivets are used in SRM. We must reach to repair areas’ back [3].

Table 5.2 Single Shear Strength of NAS1097DD5 [1]

Undriven			Driven		Rivet Designation	Rivet Size							
Rivet Material	F_{su} (ksi) Min	F_{su} (ksi) Max	Rivet Material	F_{su}^{b} (ksi)		1/16	3/32	1/8	5/32	3/16	1/4	5/16	3/8
						Driven Single Shear Strength, lbs[c]							
5056-H32	24	n/a	5056-H321[d]	28[e]	B[f]	99	203	363	556	802	1450	2290	3275
2117-T4	26	n/a	2117-T3	30[e]	AD	106	217	389	596	860	1555	2455	3510
2017-T4	35	42	2017-T3	38[e]	D	134	275	493	755	1085	1970	3115	4445
2024-T4	37	n/a	2024-T31	41[g]	DD	145	297	532	814	1175	2125	3360	4795
7050-T73	41	46	7050-T731[d]	43[e]	E[h]	152	311	558	854	1230	2230	3520	5030
Monel	49	59	Monel	52[e]	M	183	376	674	1030	1490	2695	4260	6085
Ti-45Nb	50	59	Ti-45Nb	53[e]	T	187	384	687	1050	1515	2745	4340	6200
A-286	85	95	A-286	90[e]	-	317	651	1165	1785	2575	4665	7375	10500

Shear strength for NAS1097DD5 $(Diameter = 5/32'')$ is $854\,lb$. Its material is 7050 – T73. $P_{shear} = 854\ lb$ for NAS1097DD5 (5/32)

6. AIRCRAFT METAL STRUCTURAL REPAIR IN TURKISH AIRLINES

The satisfactory performance of an aircraft requires continuous maintenance of aircraft structural integrity. It is important that metal structural repairs be made according to the best available techniques because improper repair techniques can pose an immediate or potential danger. The reliability of an aircraft depends on the quality of the design, as well as the workmanship used in making the repairs. The design of an aircraft metal structural repair is complicated by the requirement that an aircraft be as light as possible. If weight were not a critical factor, repairs could be made with a large margin of safety. In actual practice, repairs must be strong enough to carry all of the loads with the required factor of safety [14].

6.1. Steps of a Fuselage Repair

STEP 1: Aircraft producing companies are issued Structural Repair Manuel (SRM) which has repair methods. Firstly, SRM is researched whether the repair is mentioned in SRM or not. Generally, SRM has minor repair procedures. The following repair takes part into SRM because of a minor repair. If repair design is out of the SRM, Turkish Technic sends mail to Airbus or Boeing to learn and apply a new repair design (Figure 6.1).

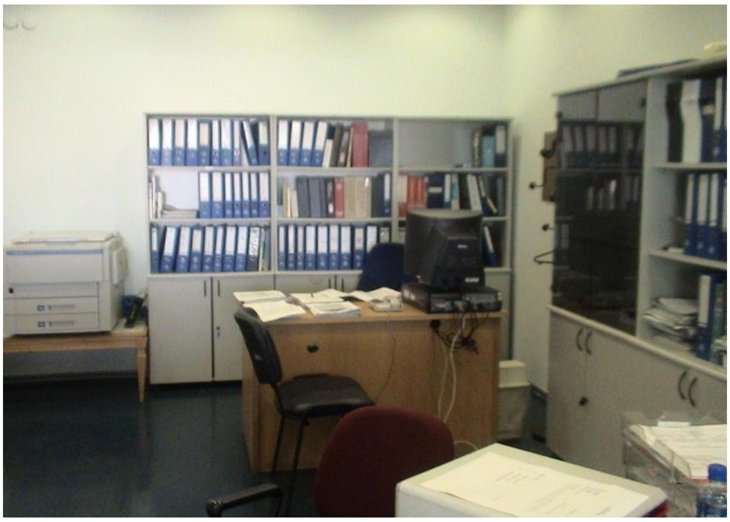


Figure 6.1 Research of Structural Repair Manuel (SRM)

STEP 2: The material of damaged aircraft skin is Aluminum and its thickness 0.047".

- Mark the repair
- Cut out the damaged skin and deburr the edges of the cutout
- Remove all rivets/fasteners in the area of the repair doubler
- Radii of cutout not smaller than 10 mm (Figure 6.2)

Figure 6.2 Cut-out of the Damaged Skin

- Measure cutout and compare with SRM to find the correct repair reference e.g. limited or unlimited cutout
- Define frame cut – lines refer to SRM

STEP 3: Manufacturing the necessary an external doubler and a filler (Figure 6.3 and 6.4)

- Define the doubler and filler size and thickness refer to SRM
- Drill the new holes refer to SRM for rivet type

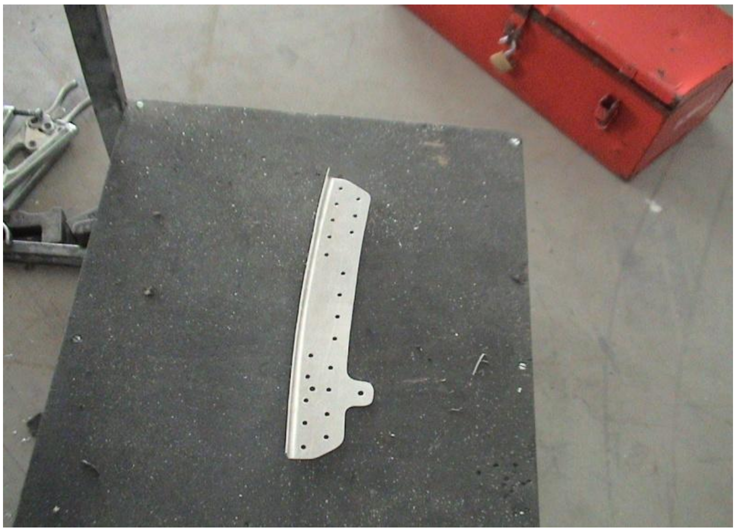


Figure 6.3 Preparing Filler for Fuselage Skin Repair

A metal sheet must be placed underneath the cutting area to avoid damage to the structure.

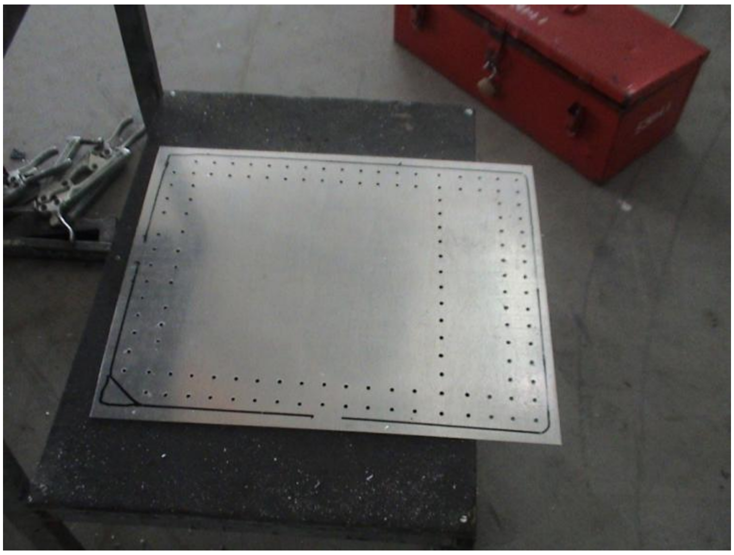


Figure 6.4 Preparing an External Doubler for Fuselage Skin Repair

STEP 4: The technical prepares the doubler (Figure 6.5)

- Cut the doubler to oversize and pre-shape it to the fuselage contour to avoid stressing
- Cut the doubler to its final dimensions
- Deburr the edges of the doubler and the filler

Figure 6.5 Preparing an External Doubler

STEP 5:

- Make sure not to damage internal structure or system when drilling (Figure 6.6).
- Clean the repair parts and repair area with cleaning agent (methyl – ethyl – ketone)

Figure 6.6 Checking of the Repair Area

WARNING: Chromic acid anodizing (CAA) is dangerous.

STEP 6:

- Position the repair parts on the repair area (Figure 6.7)
- Transfer drill the existing fastener holes and drill the additional holes, if necessary.

NOTE: For rivet pitch and margin data, refer to SRM.

Figure 6.7 Position the Repair Parts

- Apply Sealant (Interfay Sealant) to the mating surfaces of the repair parts

<u>STEP 7:</u>

- Fix the external doubler to fuselage with temporary fasteners (Figure 6.8).

Figure 6.8 Fixing the Doubler

STEP 8: Put the repair parts in position and install the fasteners wet with sealant (corrosion inhibiting brush consistency) (Figure 6.9)

Figure 6.9 Installation of the Fasteners

STEP 9: Apply the items in Structural Repair Manuel (SRM) step by step (Figure 6.10).

Figure 6.10 Following of Structural Repair Manuel (SRM)

STEP 11: Seal all the edges of the doubler with sealant (Figure 6.11).

Figure 6.11 Sealing all the Edges of the Doubler with Sealant (Corrosion Inhibiting Brush Consistency)

STEP 12: Prevent to spread of the sealant (Figure 6.12 and 6.13).

Figure 6.12 White Duct Tape

Figure 6.13 Removing of the White Duct Tape

6.2. Another Fuselage Repairs

Repairs are which different sizes and in different locations are shown below. Especially, doors and windows are damaged parts very much (Figure 6.14, 6.15, 6.16. 6.17, 6.18, 6.19, 6.20 and 6.21).

Figure 6.14 Near Passenger Door

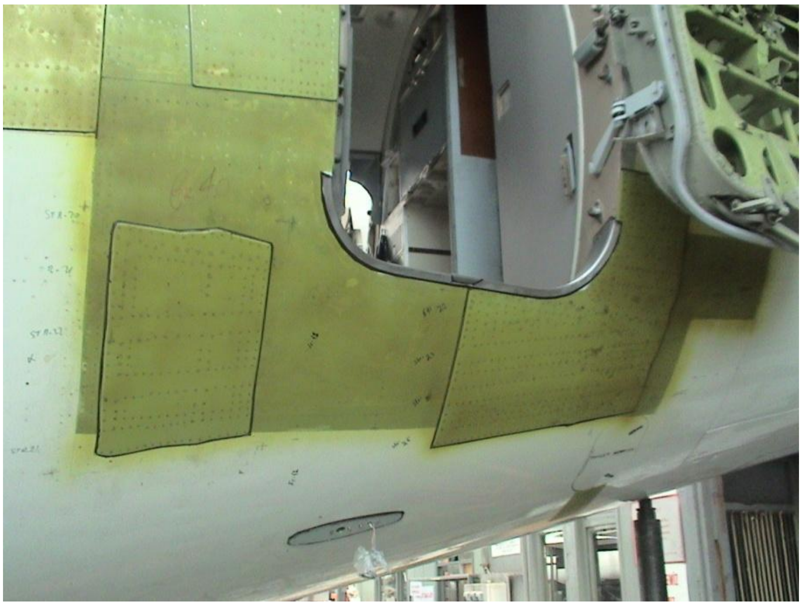


Figure 6.15 Aircraft Passenger Door Edges

Figure 6.16 Aircraft Window Edges

Figure 6.17 Aircraft Cargo Door Edges

Figure 6.18 A Fuselage Repair

Figure 6.19 Fuselage Repair (Cut-out)

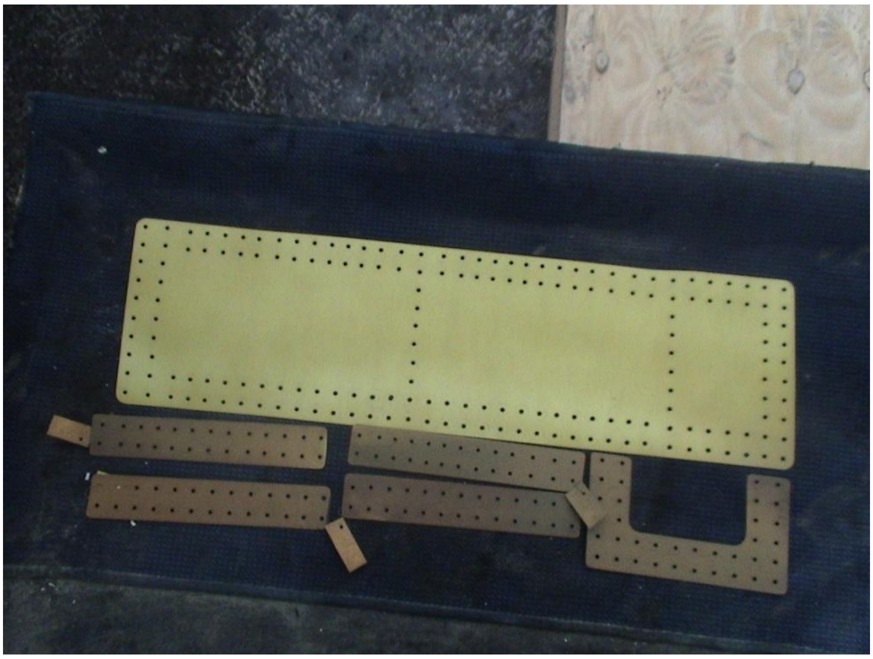


Figure 6.20 Doublers for Fuselage Repair

Figure 6.21 Fuselage Repair (Installation of the Rivets)

After the structural repair is done on the fuselage, the repair area is painted in the Paint Shop of Turkish Technic. First of all, structure paint (wash primer) is applied. Second of all, anti corrosion primer polyurethane is applied and polyurethane top coat (grey) is dyed to the edges of the cutout. Finally, the repair area is painted white color.

7. FASTENERS IN NEW DESIGN

Blind rivets with protruding head will be used in new design. Thus, workers do not lose time with countersink rivets.

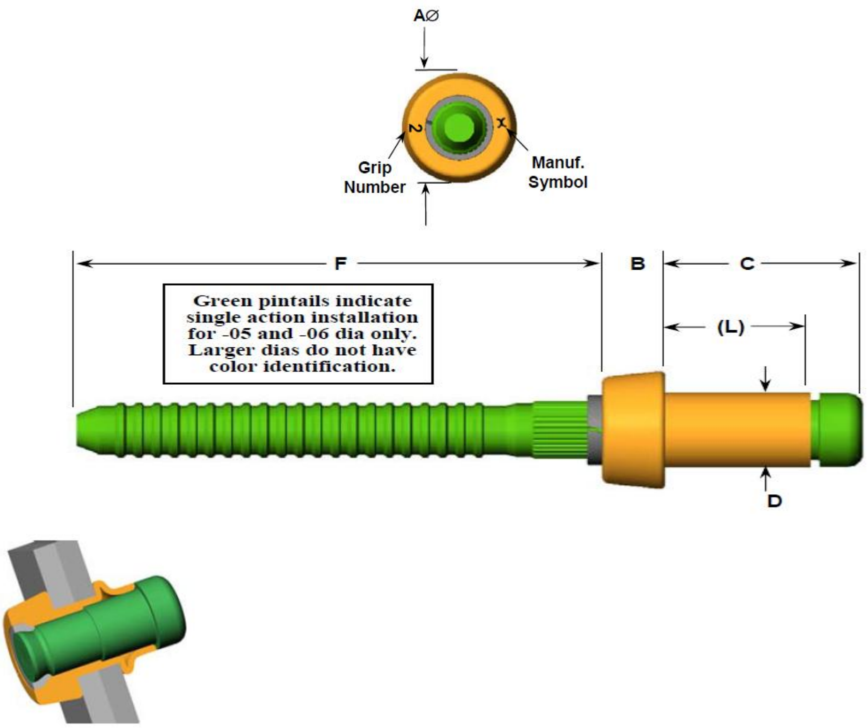


Figure 7.1 Blind Bolt Protruding Head [3]

Table 7.1 MS90354 Blind Protruding Head's Properties [4]

Dia Dash Number	Basic Dia	øA	B	øD	F Min	Single Shear Min	Tensile Min	Hole Limits
-05	5/32	.272 / .250	.070 / .062	.164 / .162	.844	2,340	1,350	.164 / .167
-06	3/16	.332 / .305	.135 / .125	.199 / .197	.875	3,450	2,100	.199 / .202
-08	1/4	.432 / .400	.140 / .130	.260 / .258	1.000	5,900	3,650	.260 / .263
-10	5/16	.522 / .480	.141 / .131	.312 / .310	1.218	8,500	5,200	.312 / .315
-12	3/8	.627 / .580	.205 / .195	.374 / .372	1.562	12,200	7,500	.374 / .377
-14	7/16	.727 / .675	.207 / .197	.437 / .435	1.562	16,700	10,150	.437 / .441
-16	1/2	.832 / .770	.270 / .260	.499 / .496	1.562	21,800	13,500	.500 / .504

Shear strength for MS90354 blind protruding head's bolt (5/32" diameter) is 2340 lb from Table 7.1.

$$Pshear = 2340\ lb$$

Table 7.2 Dimensions of MS90354 [4]

Fastener Diameter			5/32 (-05)		3/16 (-06)		1/4 (-08)		5/16 (-10)		3/8 (-12)	
Size Dash	Min Grip	Max Grip	L (Ref)	C (Max)	L (Ref)	C (Max)	L (Ref)	C (Max)	L (Ref)	C (Max)	L (Ref)	C (Max)
-XX01	0.031	0.095	.217	.372	-	-	-	-	-	-	-	-
-XX02	.094	.157	.280	.434	.303	.479	.344	.553	.381	.618	.423	.695
-XX03	.156	.220	.342	.497	.366	.542	.406	.616	.443	.680	.485	.757
-XX04	.219	.282	.405	.559	.428	.604	.469	.679	.506	.743	.548	.820
-XX05	.281	.345	.467	.622	.491	.667	.531	.741	.569	.806	.611	.883
-XX06	.344	.407	.530	.684	.553	.729	.594	.804	.631	.868	.673	.945
-XX07	.406	.470	.592	.747	.616	.792	.656	.866	.694	.931	.736	1.008
-XX08	.469	.532	.655	.809	.678	.854	.719	.929	.756	.993	.798	1.070
-XX09	.531	.595	.717	.872	.741	.917	.781	.991	.819	1.056	.861	1.133
-XX10	.594	.657	.780	.934	.803	.979	.844	1.054	.881	1.118	.923	1.195
-XX11	.656	.720	.842	.997	.866	1.042	.906	1.116	.944	1.181	.986	1.258
-XX12	.719	.782	.905	1.059	.928	1.104	.969	1.179	1.006	1.243	1.048	1.320
-XX13	.781	.845	.967	1.122	.991	1.167	1.031	1.241	1.069	1.306	1.111	1.383
-XX14	.844	.907	1.030	1.184	1.053	1.229	1.094	1.304	1.131	1.368	1.173	1.445
-XX15	.906	.970	1.092	1.247	1.116	1.292	1.156	1.366	1.194	1.431	1.236	1.508
-XX16	.969	1.032	1.155	1.309	1.178	1.354	1.219	1.429	1.256	1.493	1.298	1.570

$$t_{skin} + t_{doubler} = 0.047" + 0.056" = 0.103"$$

The total of skin and doubler thickness is 0.103", so we find grip of fasteners. It is between 0.094" and 0.157" in Table 7.2. Its dash size is XX02 and diameter 05 for (5/32" diameter). Thus, the fastener is selected MS90354 – 0502 in the new repair design.

Table 7.3 Dimensions of MS21141J [4]

Size Dash Number (b)			Nom	ØA	B	ØD	F	K		P	R	X	Installed Fastener		Hole Limits (Ref)
Double Action	Single Action		Size								Rad	(a)	Min Ultimate Single	Min Ultimate Tensile	
(c)	(c)	(d)					Min	Max	Min	Max	Max	Min	Shear (Lbs.)	(Lbs.)	
-	S04	-	.1250	.215 .197	.060 .054	.129 .127	.812	.150	.095	.190	.010	.170	1222	675	.132 .129
-05	S05	U05	.1562	.272 .250	.070 .058	.164 .162	.844	.202	.120	.215	.010	.195	1980	1150	.167 .164
-06	S06	U06	.1875	.332 .305	.135 .120	.199 .197	.875	.231	.146	.250	.015	.238	2925	1690	.202 .199
-	-08	U08	.2500	.432 .400	.140 .120	.260 .258	1.000	.279	.188	.305	.020	.315	5000	2900	.263 .260
-	-10	U10	.3125	.522 .480	.141 .131	.312 .310	1.218	.319	.224	.350	.025	.373	7200	4170	.315 .312
-	-12	U12	.3750	.627 .580	.205 .195	.374 .372	1.562	.364	.264	.405	.030	.448	10380	5970	.377 .374

Shear strength for MS21141J blind protruding head's bolt ($5/32''$ diameter) is $1980\ lb$. $Pshear = 1980\ lb$

8. REPAIR DESIGN GUIDELINE

8.1. Purpose of the Repair Design Guideline

Each repair on an aircraft structure has to be capable of sustaining the original justified design loads for static, fatigue and damage tolerance requirements. In most cases only the static requirements need to be met for the repair principles on secondary structure and only those will be addressed in this part.

A static or stress analysis is performed to make sure that the structure can withstand the applied ultimate loads without failure and without permanent deformation at limit load.

There are two main approaches in demonstrating the static strength capability of the repair design. The approach taken depends on how much is known concerning the existing loading in the repaired part.

(1) All details of the applied loading are known:

In this case, a detailed stress analysis of the part to be repaired results in an optimized repair design. This is generally the approach taken by the manufacturer.

(2) No details of the applied loading are known:

In this case the <u>*REVERSE ENGINEERING METHOD*</u> may be employed to evaluate the repair static strength capability [2].

For the reverse engineering path, the engineer must evaluate the allowable loads in the structure before the damage, and then compare the result with the evaluated post-repair loads.

8.1.1. Reverse engineering principle

The allowable load in the pre-damage state can only be calculated if a stress in the area is assumed. This assumed stress is related to the material of the original structure (e.g the assumed stress is equal to the ultimate tensile stress Ftu of the material for the tension check and related to the component geometry for the compression check).

In the case where the item is considered as "fatigue critical", this assumption for the tension check (i.e. use of Ftu) can be too conservative and a more realistic approach is to assume a maximum stress equal to the material yield stress (Fty).

Important Remark: It must be emphasized that this assumption should only be used for "fatigue critical" components, (e.g fuselage skin and attached frames or stringers). For items that are not "fatigue critical" and designed in a more traditional manner (e.g. fuselage cross beams, floor struts, etc.), the allowable ultimate tensile stress of the material (Ftu) should be used.

$$Reserve\ Factor\ (R.F.)\ =\ Allowable\ Load\ /\ Applied\ Load$$

$$R.F. \geq\ 1.00$$

Note: It must be noted that the reverse engineering path cannot take into account the undamaged structure Reserve Factor (which is greater than 1.00). The RF derived using the reverse engineering method will therefore be larger than that calculated using the manufacturer's path.

8.2. Scope and Limitations

This topic presents some guidelines which may be used when designing repairs for extruded/formed sections. These guidelines are based on the reverse engineering method.

These guidelines are based on the restoration of the static strength of the repaired part. The resulting repair will therefore comply with the static strength requirements. These guidelines, when used together with the other standard SRM principles and guidelines (fastener hole and drill data, spacing and margin data, sealing, etc.) may be used repairs to secondary structure without referring back to AIRBUS.

These guidelines may also be used when designing repairs, which are to be submitted to AIRBUS for approval, however it should be noted that the assumption taken when designing a repair based on the reverse engineering method are often conservative and may result in over−designed repairs.

The proposed method addresses only the static performance of the repair since no fatigue and damage tolerance assessment is required for repair solutions referring to this part. Nevertheless these repair solutions ensure minimum acceptable fatigue and damage tolerance performances [2].

8.3. List of Symbols

This list shows which symbols has been incorporated into the repair design guideline. It also shows the measurement unit of the used symbol and its description.

Table 8.1. List of Symbols

SYMBOL	UNIT	DESCRIPTION
A_i	mm^2	Cross Sectional Area of the Element
a_i	mm	External Breadth of the Element
b_i	mm	Calculate Element Breadth
E_c	Mpa	Young's Modulus
F_{bru}	Mpa	Allowable Ultimate Bearing Stress. Examples use e/d = 2.0
F_{su}	MPa	Allowable Ultimate Stress in Pure Shear
F_{tu}	MPa	Allowable Tensile Stress
F_{ty}	MPa	Yield Strength at 0,2% Strain
L	mm	Profile Length
r	mm	Radius
s_i	mm	Element Thickness
P_{shear}	lb	Shear Strength

8.4. Factor Definitions

8.4.1. Safety Factor

The relation of the maximum allowable load without failure and without permanent deformation and the designed applied load is expressed in a value known as the safety factor. In aircraft design the factor of safety is 1.5.

$$Ultimate\ Load\ =\ 1.5\ *\ Limit\ Load$$

8.4.2. Reserve Factor

The degree to which a structure can withstand the ultimate load is expressed in a value known as the reserve factor (R.F.). The reserve factor must be greater or equal to 1.0.

$$Reserve\ Factor\ (R.F.)\ =\ Allowable\ Load\ /\ Applied\ Load$$

$$R.F. \geq\ 1.00$$

8.4.3. Margin of Safety

An other expression of the reserve factor is margin of safety (MS).

$$Margin\ of\ Safety\ =\ Reserve\ Factor\ -\ 1$$

$$M.S. \geq\ 0$$

8.5. Principle

8.5.1. Repair Material and Dimensions

In order to restore the original structure capability, a basic principle when reinforcing or splicing a part is to use the same material as the original. The thickness of the repair part should ensure that the cross–sectional area of the repair restores the original cross sectional area. Generally use one gauge thicker than the original.

8.5.2. Analysis

A static analysis for a repair requires normally the following checks:

- ✓ Tension check
- ✓ Shear check
- ✓ Compression check
- ✓ Check to ensure that the fasteners are able to transfer the applied loading.

For the reverse engineering approach the structures engineer must calculate the allowable load in the pre–damaged state and then compare that to the post–repair allowable load. The allowable load in the pre–damaged state can only be calculated if a stress in the repaired area is assumed. This assumed stress is related to the material of the repaired part, i.e. assume a stress equal to the ultimate tensile stress, the lower of Ftu or $1.5 * Fty$ of the material for the tension check and to the component geometry for the compression check.

The reverse engineering method uses the $'A'$ basis values from the tables. For extrusions the values in the L material direction are used as the material direction is known. For items made from sheet the values in the Lt material direction are taken except if the value in the L direction is lower than the Lt value. To calculate the allowable load in the pre–damaged state the section to be considered is:

- ✓ **for profiles:** The original net section of the part.

This section is the original profile section minus the area(s) corresponding to the original fastener holes. Additional repair fastener holes not considered. The repair fastener holes are not taken into account as we need to calculate the maximum potential strength of the original part in order to design the repair with this capability. If you take into account the repair fastener holes then the potential tensile strength of the original part will be underestimated.

- ✓ **for plates and webs:** The repair net section of the part.

This section is the one of the considered (cut out width) minus the area corresponding to the original and the additional repair fastener holes. For plates and web repairs the section to be taken corresponds to the width of the cutout (length of cutout * web thickness) minus the area of additional repair fastener holes (diameter of hole * original web thickness * number of fasteners in the cutout length).

8.5.2.1. Tension and shear check

For the tension and shear check, material allowable tables have been developed which include the main material properties for most common alloy.

8.5.2.2. Compression check

This method can be used to determine pre and post allowable loading.

8.5.2.3. Fastener check

For the fastener check, the pre−damage tension and compression loads should be compared and the maximum used as the applied fastener load. Tables of allowable values (Fasteners − Strength of Mechanically Joints) can be used to determine the allowable fastener load. The reserve factor must again be greater or equal to 1.

8.5.2.4. Allowable compressive load

The reverse engineering method can be used to calculate the compressive crippling load of short metal profiles of a general shape. The calculation is based on an empirical method following sufficient tests on various shapes and materials. The calculation method assumes that the profile can be idealized using flat rectangular plates of constant thickness.

8.6. Calculation

8.6.1. Calculation under tensile load

- ✓ **Maximum tensile load**

$$Ptmax = A * (Ftu\ or\ 1{,}5 * Fty \quad \text{or whichever is less})\ [N]$$

$\boldsymbol{Ptmax}$: $Maximum\ Tensile\ Load$

$\boldsymbol{A}$: $Cross\ Sectional\ Area$

$\boldsymbol{Fty}$: $Ultimate\ Tensile\ Stress$

- ✓ **Reserve factor**

$$R.F. = \frac{P_{t\ maxRep}}{P_{t\ max0ri}}$$

$\boldsymbol{P_{t\ maxRep}}$**:** Maximum Tensile Load of Repaired Part

R.F.: Reserve Factor ≥ 1.0

$\boldsymbol{P_{t\ max0ri}}$**:** Maximum Tensile Load of Origin Part

- ✓ **Determination of fastener number**

$$No_{Row} \geq \frac{P_{t\ max\ 0ri}}{F_{\text{ultim Riv}}}$$

$\boldsymbol{P_{t\ max0ri}}$**:** $Maximum\ Tensile\ Load\ of\ Origin\ Part$

$\boldsymbol{No_{Row}}$: Number of Fastener Rows

$\boldsymbol{F}_{\textbf{ultim Riv}}$: $Ultimate\ Handling\ Force\ of\ one\ Row\ of\ Rivets$

8.6.2. Calculation under shear load

- ✓ **Maximum shear load**

$$Psmax = A * Fsu$$

$\boldsymbol{Psmax}$: $Maximum\ Shear\ Load$

$\boldsymbol{A}$: $Net\ Section\ Area$

$\boldsymbol{Fsu}$: $Ultimate\ Shear\ Stress$

- ✓ **Reserve factor**

$$R.F. = \frac{P_{\text{smax Rep}}}{P_{\text{smaxOri}}}$$

$\boldsymbol{P}_{\textbf{smax Rep}}$: $Maximum\ Shear\ Load\ of\ Repaired\ Part$

R.F.: Reserve Factor ≥ 1.0

$\boldsymbol{P}_{\textbf{smaxOri}}$: Maximum Shear Load of Origin Part

- ✓ **Determination of fastener number**

$$No_{Row} \geq \frac{P_{t\ max\ 0ri}}{F_{\text{ultim Riv}}}$$

$\boldsymbol{P_{t\ max0ri}}$**:** $Maximum\ Tensile\ Load\ of\ Origin\ Part$

$\boldsymbol{No_{Row}}$: Number of Fastener Rows

$\boldsymbol{F}_{\textbf{ultim Riv}}$: $Ultimate\ Handling\ Force\ of\ one\ Row\ of\ Rivets$

9. CALCULATIONS OF REPAIR DESIGN

Aircraft structural members are designed to perform a specific function or to serve a definite purpose. The primary objective of aircraft repair is to restore damaged parts to their original condition. Very often, replacement is the only way this can be done effectively. When repair of a damaged part is possible, first study the part carefully to fully understand its purpose or function.

An inspection of the damage and accurate estimate of the type of repair required are the most important steps in repairing structural damage. The inspection includes an estimate of the best type and shape of repair patch to use; the type, size, and number of rivets needed; and the strength, thickness, and kind of material required to make the repaired member no heavier (or only slightly heavier) and just as strong as the original [5].

9.1. Case 1

9.1.1. Airbus calculation

A "static analysis" or stress analysis is performed to ensure that the structure of the aircraft can withstand the applied ultimate loads without catastrophic failure. Dimensions used for repair design are shown in Figure 9.4. Cutout size is $1'' x\, 1''$. Cut – loss of metal, usually to an appreciable depth over a relatively long and narrow area, by mechanical means, as would occur with the use of a saw blade, chisel or sharp – edged stone striking a glancing blow.

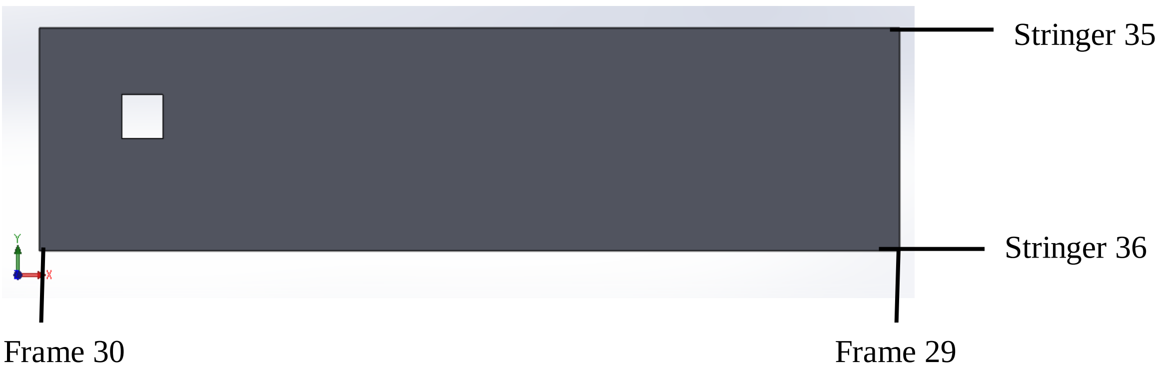
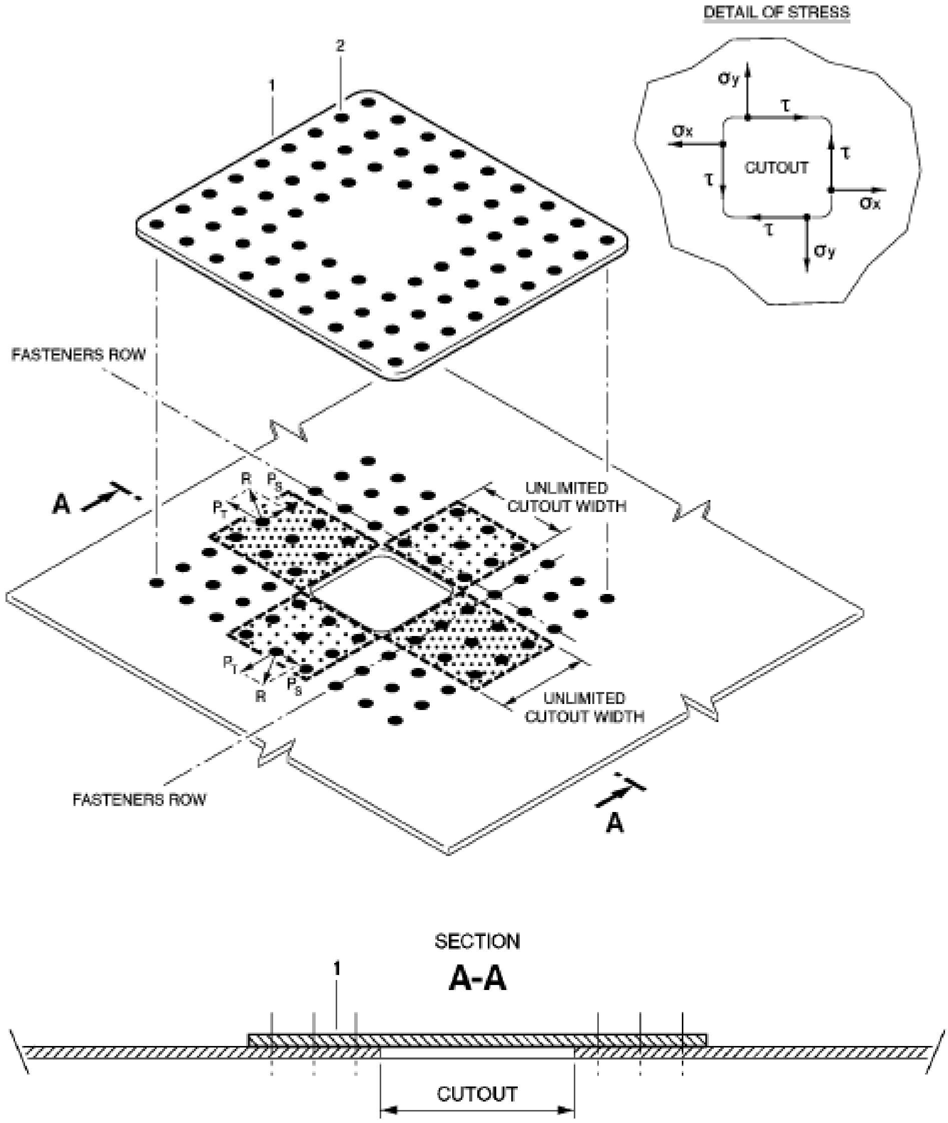


Figure 9.1 Location of Case 1 ($1'' x\, 1''$ Cut-out)

Structural Repair Manuel (SRM) offers three rows NAS1097DD5 rivets with external doubler. In this thesis, this repair’s details are given in Chapter 3.2



Area A: Tensile and Shear Load

Area B: Tensile and Shear Load

Figure 9.2 Available Repair Design

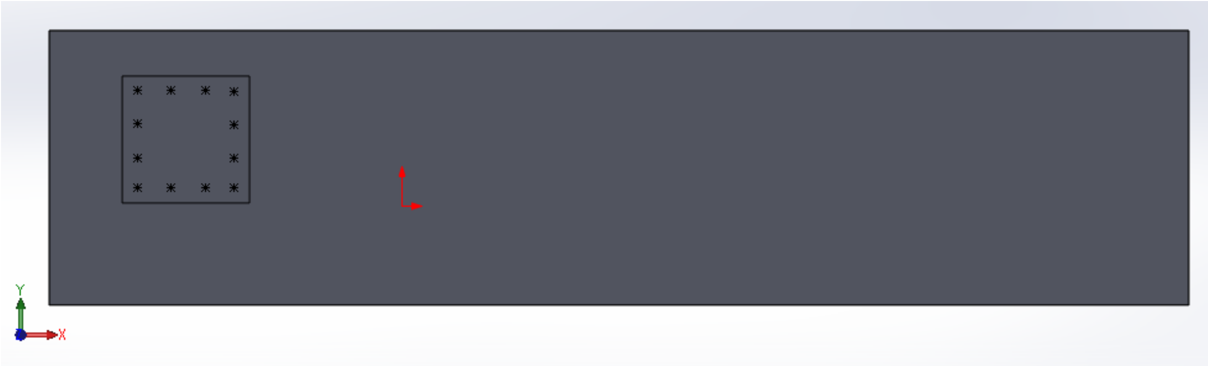


Figure 9.3 Assembly Doubler ($1'' x\ 1''$ Cut-out)

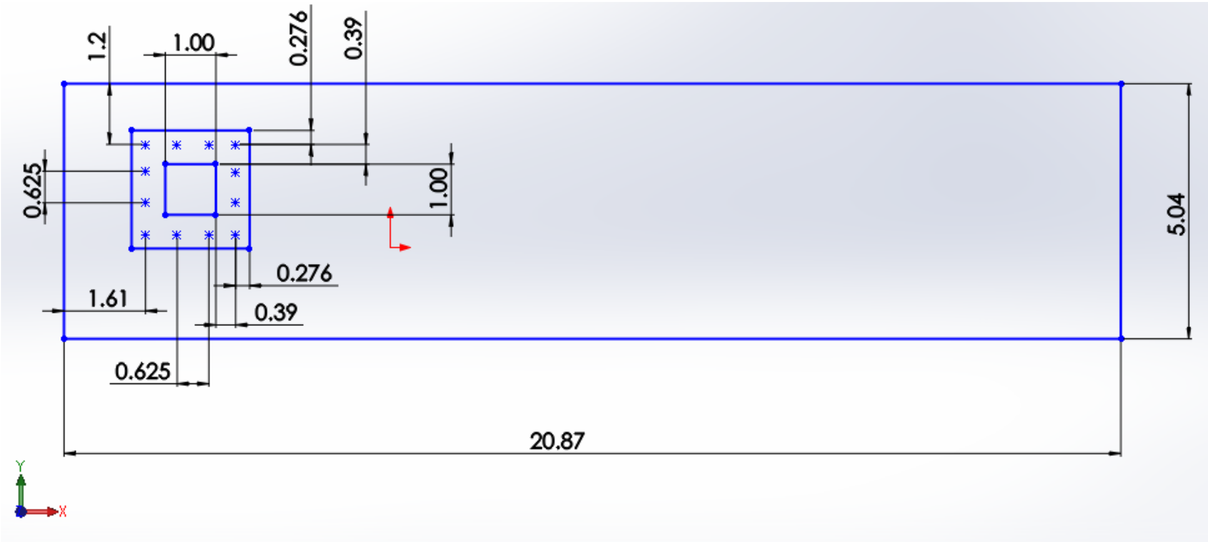


Figure 9.4 Dimensions of Case 1

9.1.1.1.Material capability (tension check)

The reverse engineering path is often the only way that airline engineers can take to design and check the repair capabilities, since they will not normally have access to manufacturer reports containing details of the applied loading. However, aircraft's material capability are known. In this case the reserve engineering method may be employed to evaluate the repair static strength capability. The following calculations are done according to the most critical places. Because, cracks probably start from the most critical places.

- Original Part:

Clad 2024 T-3 – thickness $1.2\ mm\ =\ 0.047''$

- Doubler:

Clad 2024 T-3 – thickness $1.4\ mm\ =\ 0.056''$

Cutout Original Part:

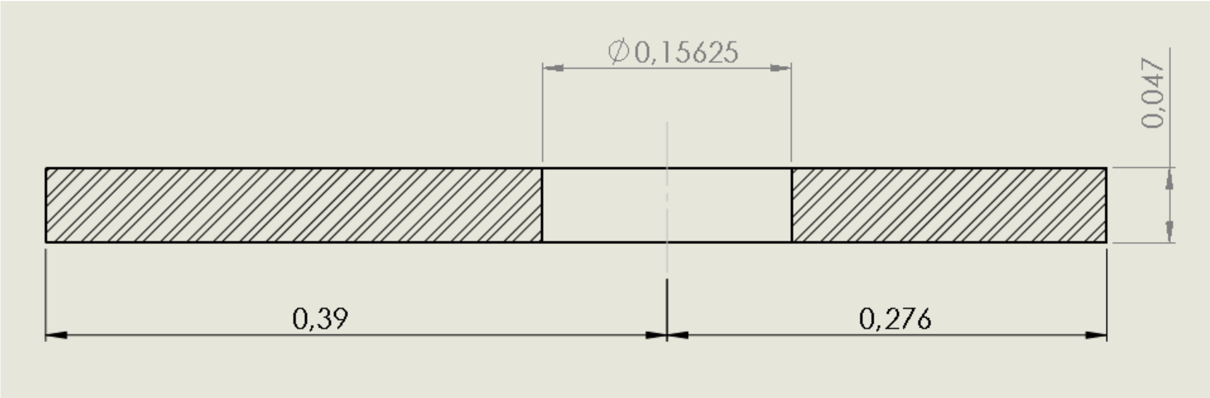


Figure 9.5 Sketch of Original Part Section (inc)

$$Section\ Area\ =\ Length\ *\ Thickness$$

$$Section\ =\ \left(0.666'' - \frac{5}{32}\right) * (0.047\ '') = 0.023958\ in^2$$

$$F_{tu} = Allowable\ Tensile\ Stress = 61000\ psi$$

$$P_{tens\ max.} = Section * F_{tu}$$

$$P_{tens\ max.} = (0.023958\ in^2) * (61000\ psi) = 1461.45\ lb$$

$$F_{su}\ = Allowable\ Ultimate\ Stress\ in\ Pure\ Shear$$

$$P_{shear\ max.} = Section * F_{su}$$

$$P_{shear\ max.} = (0.023958\ \ in^2) * (38000\ psi) = 910.41\ lb$$

Doubler Repair:

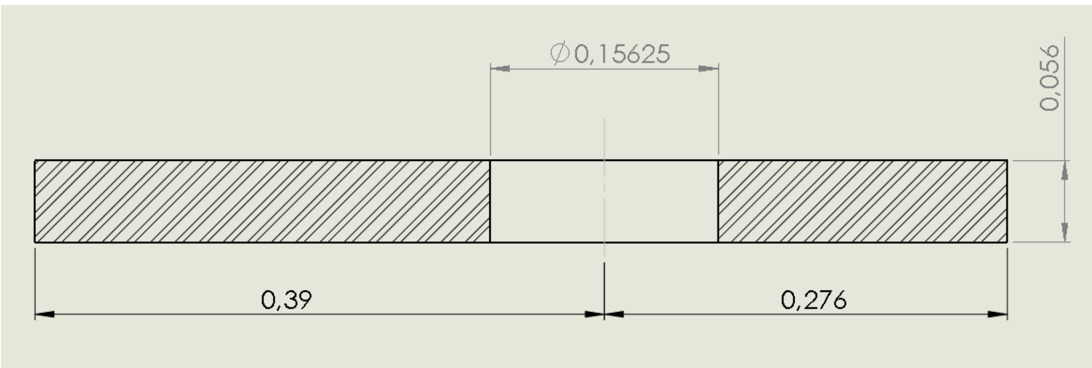


Figure 9.6 Sketch of Doubler Section (inc)

$$Section = length * thickness$$

$$Section = \left(0.666'' - \frac{5}{32}\right) * (0.056'') = 0.028546\ in^2$$

$$P_{tens\ max.} = Section * F_{tu}$$

$$P_{tens\ max.} = (0.028546\ in^2) * (61000\ psi) = 1741.306\ lb$$

$$P_{shear\ max.} = Section * F_{su}$$

$$P_{shear\ max.} = (0.028546\ in^2) * (38000\ psi) = 1084.748\ lb$$

- **Check of tensile reserve factor:**

$$Reserve\ Factor = \frac{P_{tens\ max.\ \ doubler}}{P_{tens\ max.\ \ orig}}$$

$$= \frac{1741.306}{1461.45} = 1.19 > 1 \quad ALLOWABLE$$

Reserve factor or margin of safety must be greater than 1.00, so it is acceptable.

A static analysis for a repair basically requires 3 checks:

- ✓ A tension check,
- ✓ A shear check,
- ✓ A compression check,
- ✓ A check to ensure that the fasteners are able to transfer the applied loading.

- **Check of shear reserve factor:**

$$Reserve\ Factor = \frac{P_{shear\ max.\ \ doubler}}{P_{shear\ max.\ \ orig}}$$

$$= \frac{1084.748}{910.41} = 1.19 > 1 \quad ALLOWABLE$$

- **Combination tensile and shear check:**

The maximum (principle) stress is limited to F_{tu}. The calculation in tensile covers any combined stress in tensile and shear.

- **Compression check:**

The original section is restored by the doubler section with increased thickness and equal or better material characteristics (F_{cy} and E_c).

9.1.1.2.Determination of fastener number

We can calculate the required number of rivets by using the rivet formula. Firstly, the resultant force are obtained from tensile and shear loads.

$$R = \sqrt{P_{tens\,max.orig.}{}^2 + P_{shear\,max\,orig}{}^2}$$

$$R = \sqrt{1461.45^2 + 910.41^2} = 1721.83\ lb$$

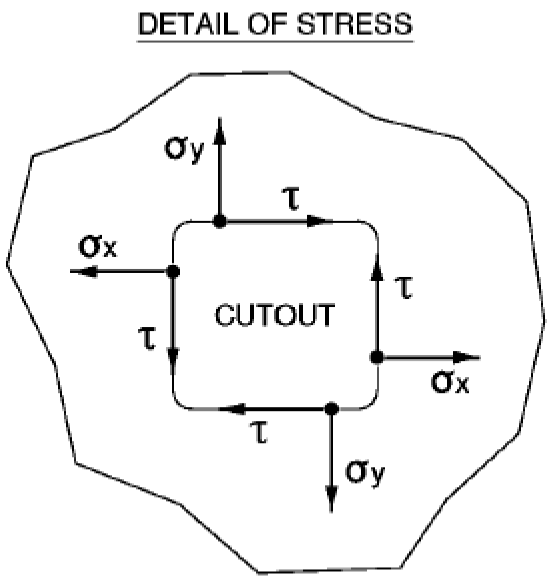


Figure 9.7 Resultant Force

In the new repair design, $MS21141 - 0502$ blind bolt is offered instead of solid rivet.

Capacity of 1 Row of 1 Fasteners:

$Shear\ Strength\ for\ MS21141 - 0502 =\ 1980\ lb$

$$1 * (1980\ lb)\ =\ 1980\ lb$$

Number of Fastener Rows:

$$Rows \geq \frac{1721.83}{1980} = 0.87$$

$$1 \geq 0.87 \quad ALLOWABLE$$

Thus, 1 row of MS21141 – 0502 blind bolt bears the applied loads.

9.1.2. Boeing calculation

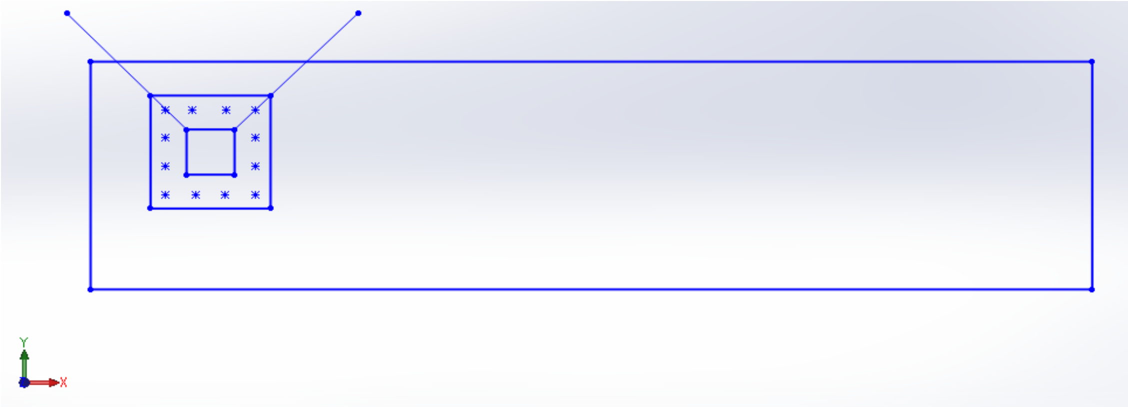


Figure 9.8 1 Row Fasteners for $1'' \; x \; 1''$ Cutout

9.1.2.1. Static strength restoration

$$Thickness\ of\ the\ skin\ = \ 0.047\,''\ \ (Material\ Al - T3)$$

$$Thickness\ of\ the\ repair\ doubler\ (same\ material) = \ 0.056''$$

$$Strength\ MS = \frac{0.056''}{0.047\,''} - 1 = 0.19 > 0$$

9.1.2.2.Joint capability check

Calculate Load Capacity Loss

Assuming load loss due to trim in the skin. Material: $0.047\,''\ \ (Material\ Al - T3)$

$$A_{lost}\ = (1'') * (0.047'') = 0.047\ in^2$$

$$P_{lost}\ = A_{lost} * F_{tu} = \ (0.047\ \mathrm{in}^2) * (61000\ psi) = \ 2867\ lb$$

Total joint load is carried by 12 fasteners on the repair parts as follows:

9.1.2.3.Total joint load cap

$$P_{JT}\ = Shear\ Strength\ for\ MS21141 - 0502 = \ 1980\ lb$$

$$P_{JT\ (Tot.)} = 2 * P_{JT}\ = 2 * (1980\ lb) = 3960\ lb$$

$$M.S.for\ Joint\ Load\ Capacity\ = \frac{P_{JT\ (Tot.)}}{P_{lost}} - 1$$

$$= \frac{3960}{2867} - 1 = 0.38\quad ALLOWABLE$$

We obtain the same result from Boeing calculation. The joint has enough load capacity to handle lost load. 1 row of MS21141 – 0502 blind bolt bears the applied loads. This repair is structurally satisfactory. This completes strength substantiation of this repair.

9.1.3. Fatigue of joints

Fatigue failures in structures frequently occur in joints. Various catastrophic accidents due to fatigue have been reported in the literature. As a consequence joints are a major issue for designing against fatigue. The prime purpose of a joint is to transmit loads from one element of the structure to other element [13].

9.1.3.1.Riveted joints with eccentricities

Single lap joints between sheets are used in various structures. Overlapping sheets are joined by a number of fasteners (rivets). The variety of such joints is large; the material, material thickness, type of fastener, pattern of fastener locations (how many rows and how many fasteners in a row), hole diameters, distances between fasteners, joining techniques, etc. These parameters affect the deformation behavior. Some essential aspects with respect to fasteners should be mentioned.

Riveting occurs by a deformation process on the rivet to fill up the hole and to produce the closing head (also called the driven head). Solid rivets are usually of a similar alloy as the sheets or plates. Riveted joints were applied already long ago in steel structures (e.g. bridges, cranes, ships, etc.) and also in aluminum structures, in particular aircraft structures. Riveting occurs by hammering or squeezing. Squeezing is not noisy, in contrast to hammering. Moreover, squeezing can be done in an automated production process which gives a more uniform production quality [13].

A few fastener types are shown in Figure 9.9. The protruding head rivet is the older one, which is also often used with a spherical head. The countersunk rivet was introduced to obtain a flat surface at one side of the joint which can be desirable for aerodynamic or hydrodynamic reasons. Due to the countersunk hole, the stress concentration at the hole is larger than for the cylindrical hole of a protruding head rivet. During riveting, the force applied on the rivet is pressing material of the rivet shank into the rivet hole. Hole filling depends on the rivet force, which is also called the squeeze force. If the force is low, the hole is just filled up, but if a larger force is used, the hole is plastically expanded and a much better contact between the entire rivet and the hole is obtained. This leads to better fatigue properties.

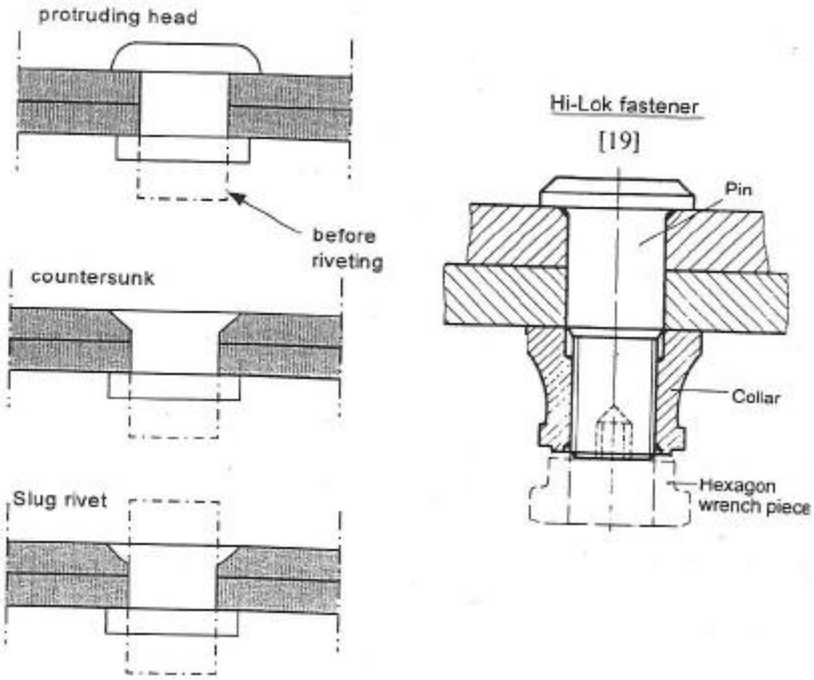


Figure 9.9 Different Types of Fasteners [13]

Figure 9.10 Hi-Lok [14]

Several types of fasteners were developed for aircraft applications for special reasons, e. g. easy and fast assembling, good fatigue properties, and high static shear strength. An example is the Hi-Lok fastener also shown in Figure 9.9 & 9.10. It is usually installed with a slight interference fit for good hole filling. The fastener is made from a high-strength steel or Ti-alloy to obtain a large static shear strength. The installation of the nut on the bolt is done from one side. The nut consists of two parts, a collar which is the real nut, and a hexagon wrench piece, see Figure 9.9. During installation, the hexagon piece is tightening the nut until this piece is sheared off from the nut. As a result, a well-controlled high torque is applied which ensures a significant clamping of the sheets by the Hi-Lok fastener, and also a high strength and good fatigue properties. Of course such a fastener is more expensive than a conventional rivet or bolt. A variety of high-tech fasteners is commercially available [13].

Riveted lap joints with eccentricities are schematically indicated in Figure 9.11. The eccentricity is equal to the sheet thickness. So if sheet thickness is thin, the eccentricity can be less.

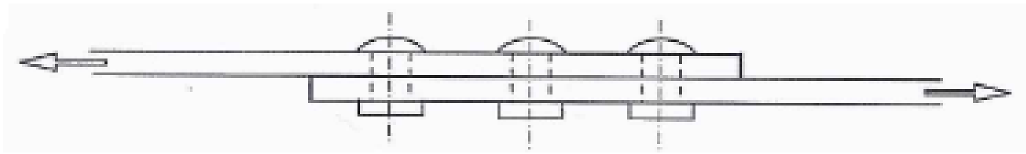


Figure 9.11 Riveted Lap Joint [13]

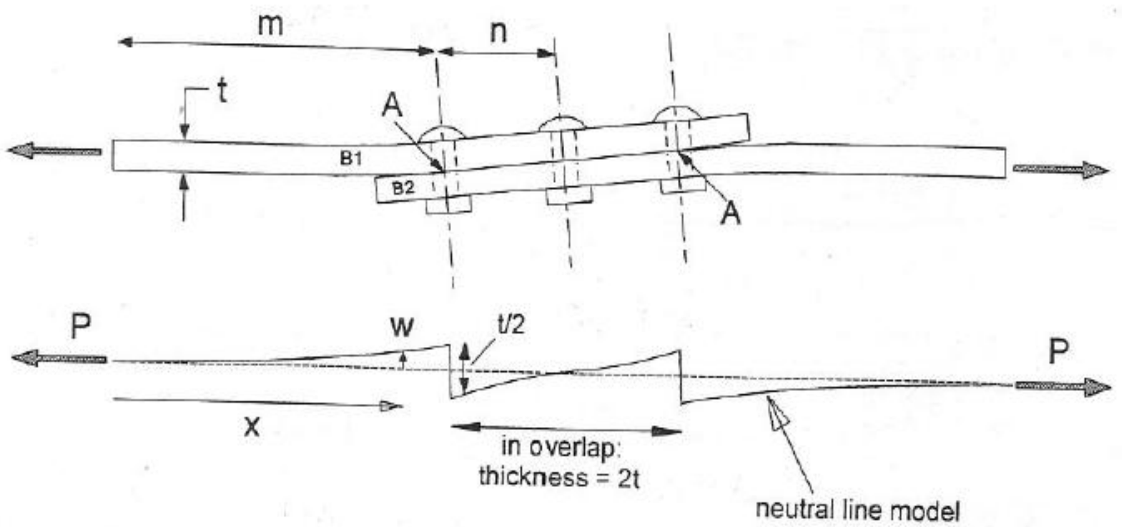


Figure 9.12 Secondary Bending in a Riveted Lap Joint Under Tensile Loading [13]

As a result of the eccentricities in a lap joint, a tensile load on the joint causes bending of the plates. Maximum bending stresses occur at the eccentricities, i.e. at the fastener rows (point A in Figure 9.12). This will cause an extra stress concentration at the holes of the fasteners. Bending caused by the tensile load is referred to as <u>secondary bending</u>. It is a by – product of the tension load. The additional bending, referred to as secondary bending, will increase notch effects in these joints. Furthermore, fasteners in a simple lap joints are loaded in single shear.

The fastener in a lap joint is asymmetrically loaded and the fastener will tilt in the hole if the hole filling by the fastener is poor. This gives an inhomogeneous bearing pressure along the hole [13].

The fatigue limit of joints can be very low due to severe stress concentrations, fretting corrosion & secondary bending. In spite of a high static strength of a joint, the fatigue limit can be low.

9.1.3.2. Predictions on the fatigue life of riveted lap joints

Predictions of the fatigue life of a riveted lap joint should be expected to be a complex problem in view of secondary bending, fastener type, fretting corrosion and rivet hole filling; all having a significant effect of the fatigue life. These aspects are not easily accounted for by analytical equations. Moreover, geometric variables of the joint must also be considered. Homan and Jongebreur suggested a prediction method for riveted lap joints in sheet material of aircraft fuselage structures loaded under constant-amplitude loading. The model starts from an S-N curve of a reference lap joint for $R = 0$ for which fatigue data are available. Predictions are extrapolated from this curve by accounting for three contributions to the stress concentration at the rivet holes of the critical end row. The contributions are associated with (i) load transmission by the rivets (pin loading on the hole), (ii) bypass loading of the other rivet rows, and (iii) increased stress by secondary bending. The equation used is the following:

$$K_t = \gamma * K_{t,pin} + (1-\gamma) * K_{t,hole,tension} + k * K_{t,hole,bending} \qquad \textbf{(5.13)}$$

$\boldsymbol{d} = Diameter$

$\boldsymbol{b} = Pitch$

$\boldsymbol{K} = Stress\ Concentration\ Factor$

$\boldsymbol{Kt} = Theoretical\ Stress\ Concentration\ Factor\ for\ Normal\ Stress$

$\boldsymbol{\gamma} = The\ Percentage\ of\ the\ Load\ Transmitted\ to\ the\ Other\ Sheet\ in\ the\ Critical\ Row$

$\boldsymbol{(1-\gamma)} = The\ Percentage\ of\ the\ Bypass\ Load$

Bending Factor: for 1 row rivet [7]

$$k = \frac{S_{bending}}{S_{tension}} = 3$$

$\boldsymbol{k}$ = The Secondary Bending Factor

$\boldsymbol{K_{t,hole,tension}}$:

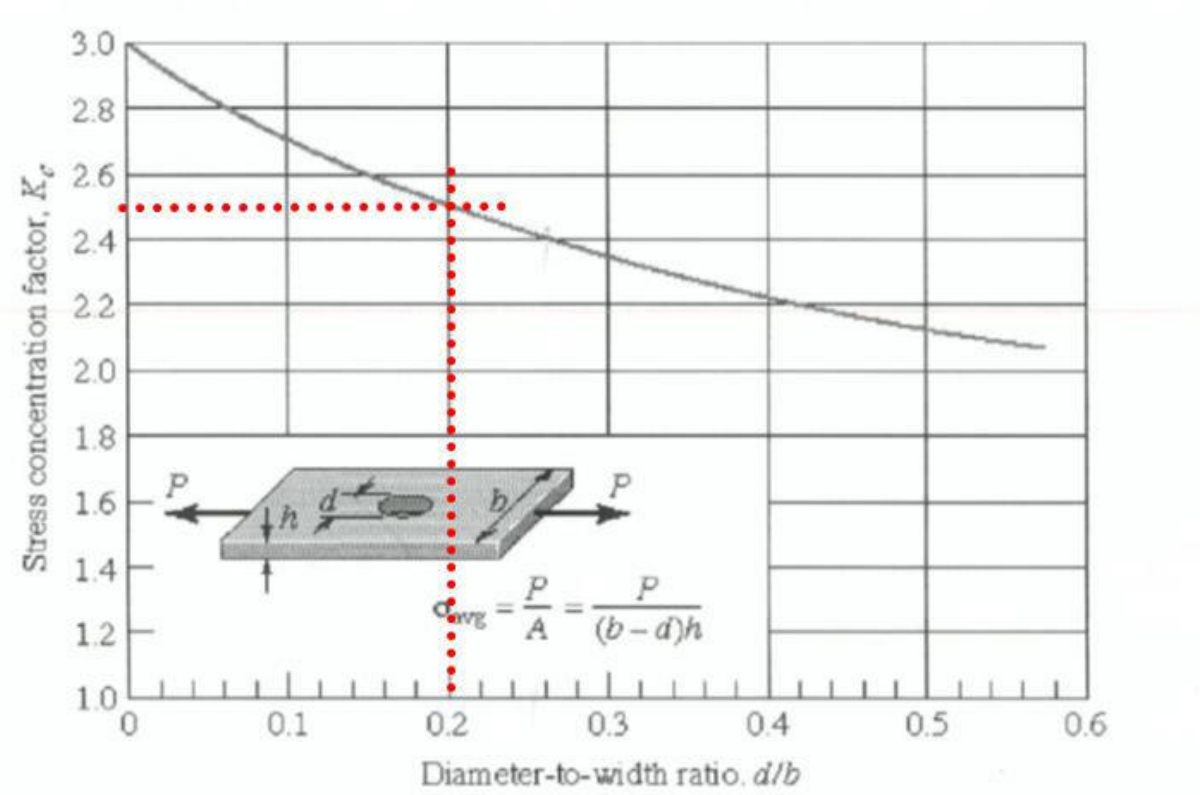


Figure 9.13 Stress Concentration Factors K_{tg} and K_{tn} for the Tension of a Finite – Width Thin Element with a Circular Hole

$$d = \frac{5}{32} = 0.15625''$$

$$pitch = 0.625''$$

$$\frac{d}{b} = \frac{0.15625''}{0.625''} = 0.25$$

$$K_{t,hole,tension} = 2.4$$

$\boldsymbol{K_{t,hole,bending}}$:

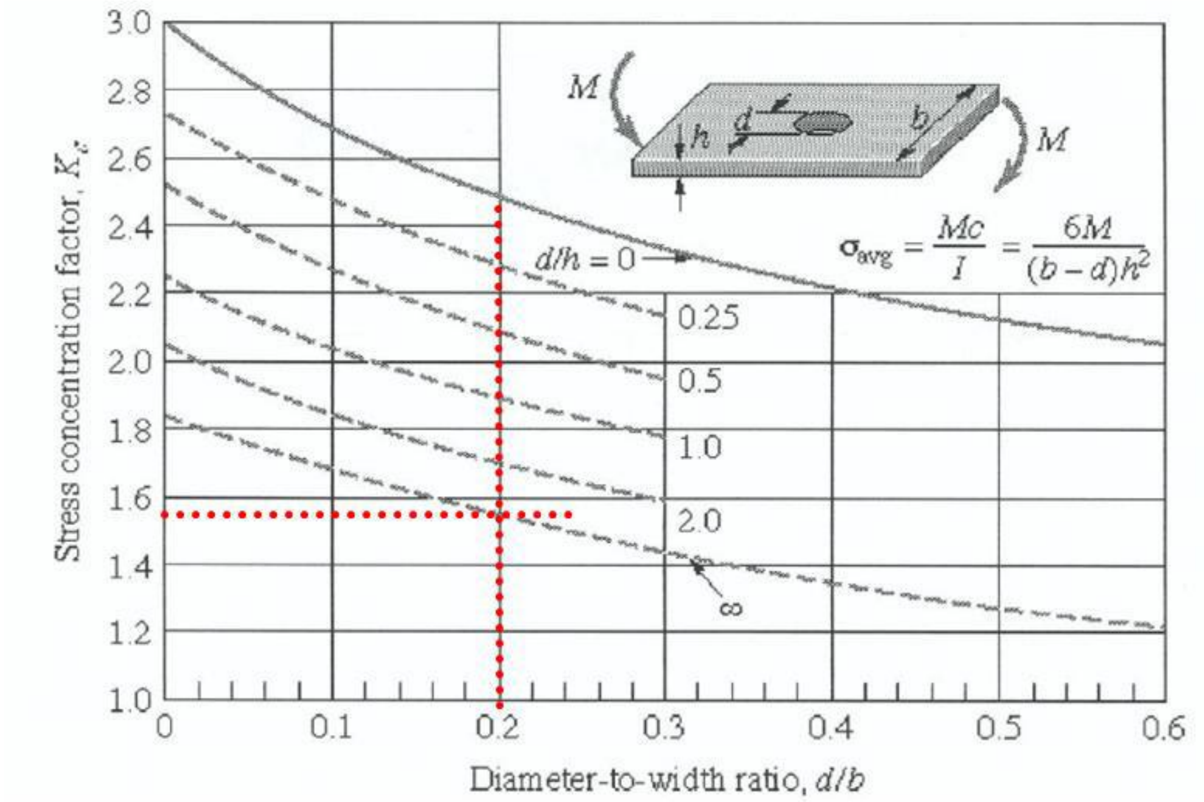


Figure 9.14 Stress Concentration Factors K_{tg} and K_{tn} for Bending of a Finite – Width Plate with a Circular Hole

$$d = \frac{5}{32} = 0.15625''$$

$$pitch = 0.625''$$

$$\frac{d}{b} = \frac{0.15625''}{0.625''} = 0.25$$

$$h = 0.047''$$

$$\frac{d}{h} = \frac{0.15625''}{0.047''} = 3.32 \rightarrow \infty$$

$$K_{t,hole,bending} = 1.5$$

$\boldsymbol{K_{t,pin}}$:

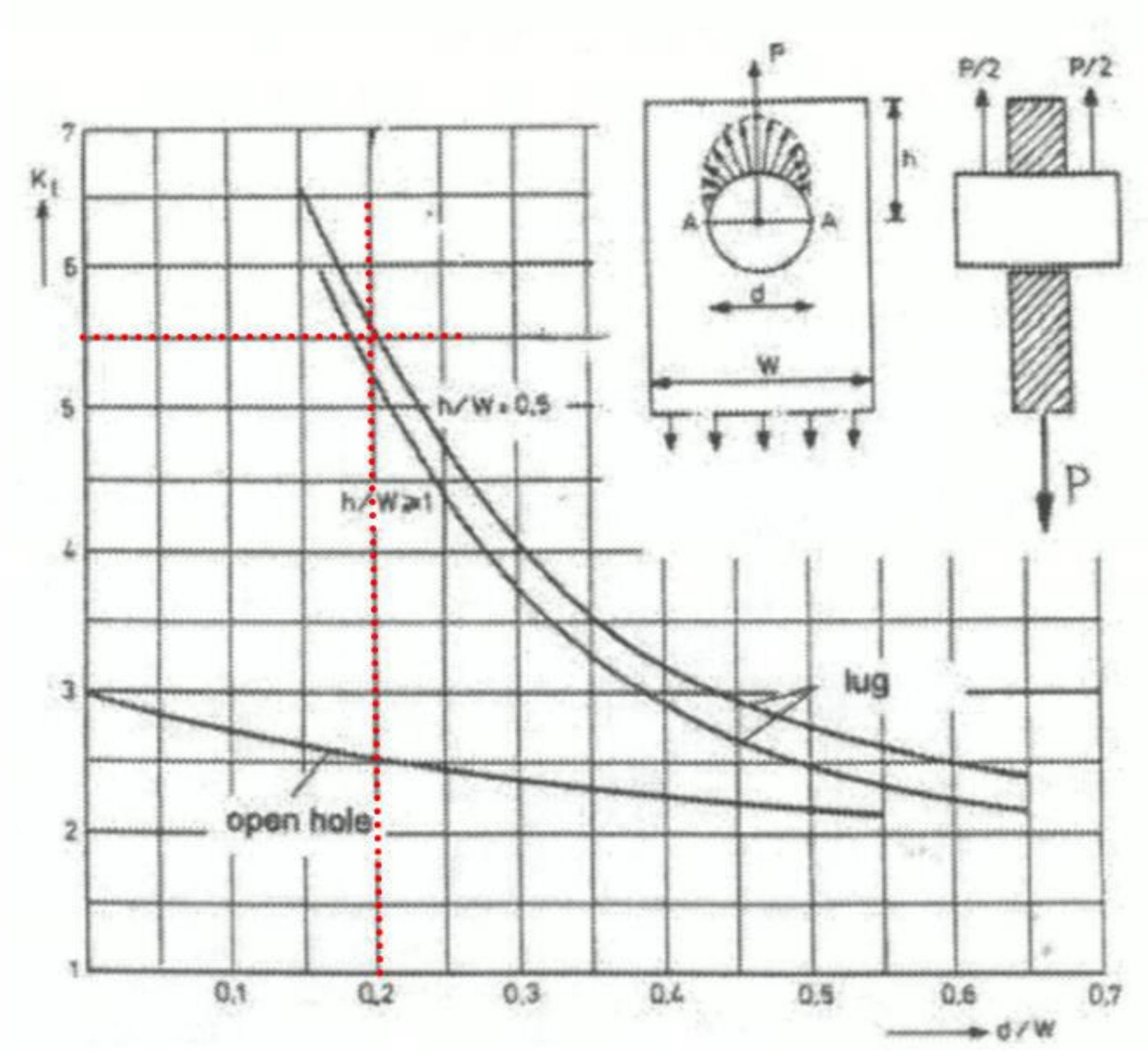


Figure 9.15 $K_t - values$ for a Lug. Comparison to an Open Unloaded Hole

$$d = \frac{5}{32} = 0.15625''$$

$$pitch\ =\ 0.625''$$

$$\frac{d}{\mathrm{w}} = \frac{0.15625''}{0.625''} = 0.25$$

$$\frac{h}{w} = \frac{0.047''}{0.625''} = 0.0752$$

$$K_{t,pin}\ =\ 4.75$$

$$K_t = \gamma * K_{t,pin} + (1-\gamma) * K_{t,hole,tension} + k * K_{t,hole,bending}$$

$\boldsymbol{\gamma}$: $the\ percentage\ of\ the\ load\ transmitted$ [7]

$\gamma = 1$ for 1 row fastener

The three stress concentration factors, $K_{t,pin}$, $K_{t,hole,tension}$ and $K_{t,hole,bending}$ depend on the joint geometry (rivet diameter / rivet pitch) . $K_{t,pin}$, $K_{t,hole,tension}$ and $K_{t,hole,bending}$ are found from Figure 9.13, 9.14 & 9.15, respectively.

$$K_t = 1 * 4.75 + (1 - 1) * 2.5 + 1.5 * 3$$

$$K_t = 9.25$$

9.1.4. Fatigue life calculation

The prediction method of Homan and Jongebreur is based on the peak stress calculated with the equation of K_t. The peak stress is calculated for the reference joint and for the actual joint for which life predictions should be made.

$$\sigma_{hoop} = \frac{\Delta P * R}{t} \quad \textbf{(5.16)}$$

$$= \frac{8.6 * 77.76''}{0.047''} = 14.23\ ksi \quad (Gross\ Stress)$$

$$\sigma_{net} = \sigma_{gross} * \frac{pitch}{d - pitch}$$

$$\sigma_{hoop}(net) = 14.23 * \left(\frac{0.625}{0.625 - \frac{5}{32}}\right)$$

$$\sigma_{hoop}(net) = 18.97\ ksi$$

<u>Peak Stress:</u>

$$\sigma_{peak} = \sigma_{net} * k_{t\,(net)}$$

$$\sigma_{peak} = (18.97\ ksi) * 9.25 = 175.5\ ksi$$

<u>MMPDS:</u>

S – N curve for notched specimen for $K_t = 5$

$$S_{peak} = 175.5\ ksi$$

$$S_{peak_{corrected}} = \frac{S_{eq}}{K_t} = \frac{175.5\ ksi}{5} = 35.1\ ksi$$

$$log\mathrm{N}_f = 8.9 - 3.73 * log\left(S_{eq} - 3.9\right)$$

$$S_{eq} = S_{max} * (1 - R)^{0.56}$$

$$S_{eq} = S_{peak_{corrected}} = 35.1\ ksi$$

$$logN_f = 8.9 - 3.73 * log(35.1 - 3.9)$$

$$N_f = 2122 \quad Cycles$$

20 years of design service goal (48000 FC) for A321. This information is mentioned in Chapter 1.8. 2122 flight cycles are obtained for a temporary repair of A321. It can be acceptable.

9.2. Case 2

9.2.1. Airbus calculation

Dimensions used for repair design are shown in Figure 9.18. Cutout size is $3'' x\ 1''$.

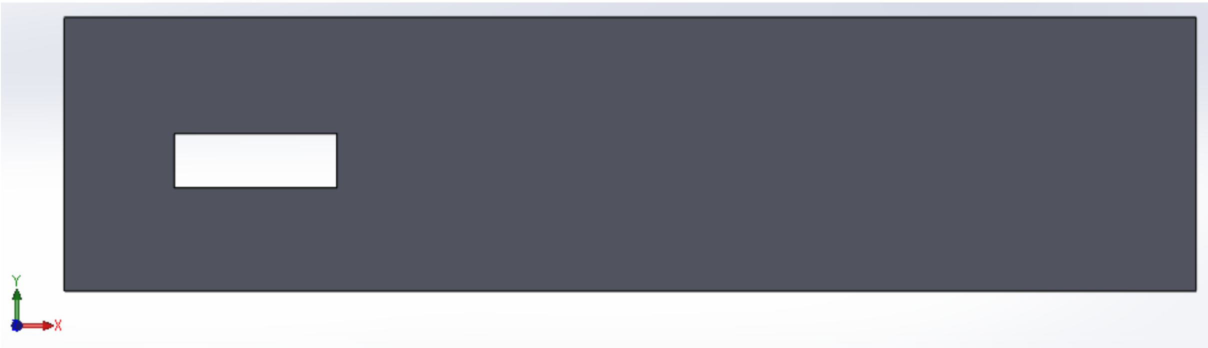


Figure 9.16 Skin Cutout ($3'' x\ 1''$)

There are chemical pockets between Frame 29 – 30 and Stringer 35 – 36. Chemical pockets are considered while rivets & cutout are designed.

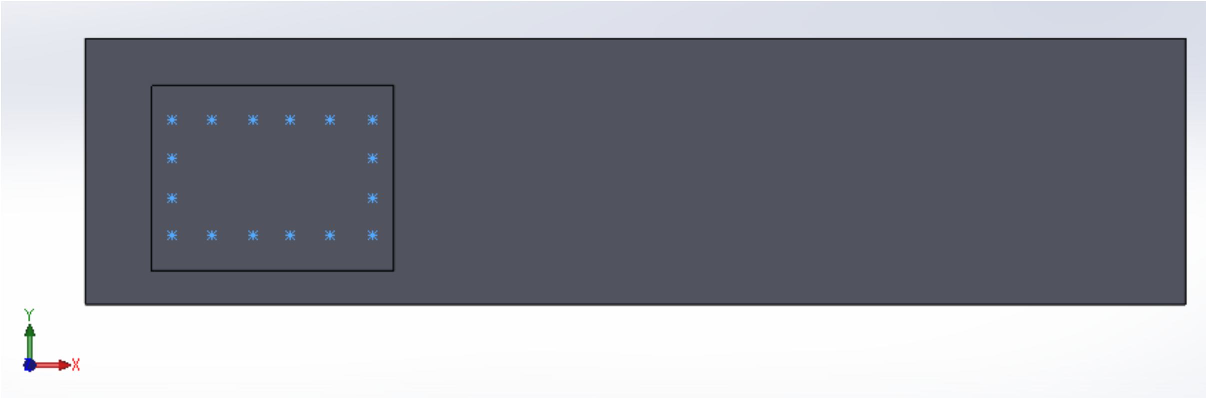


Figure 9.17 Assembly Doubler

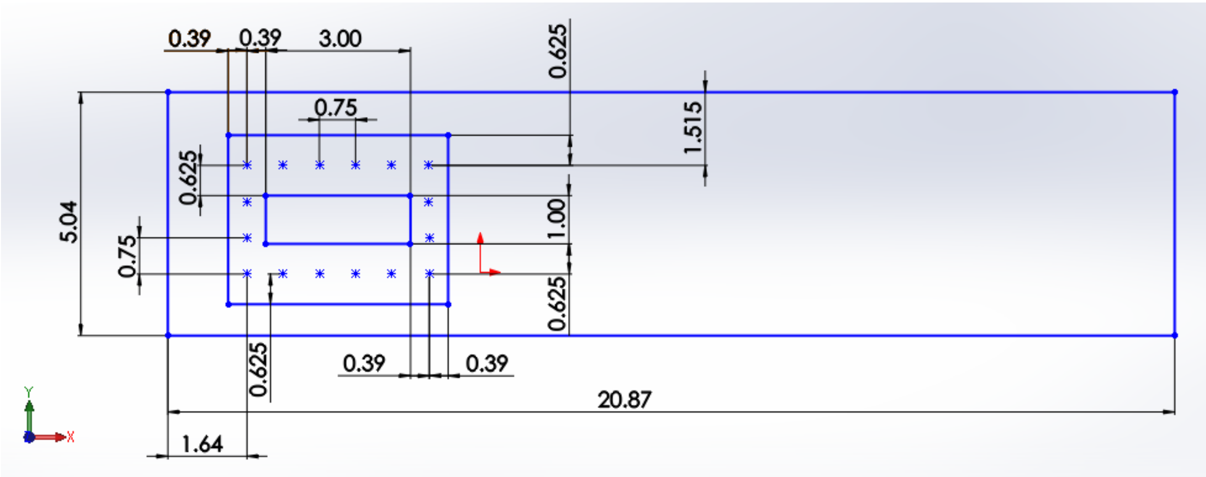


Figure 9.18 Dimensions of New Repair Design ($3'' x\ 1''$)

$Fastener\ Pitch\ = 0.75\,'' > 4\ * d\ = 0.625\,''$ Allowable (in longitudinal direction)

$Edge\ Margin = \ 2 * d = 0.3125\,''$

Fastener pitch is enough for oversizing both longitudinal and horizontal direction, so we can use the same holes in the permanent repair.

9.2.1.1.Material capability (tension check)

- <u>Original Part:</u>

Clad 2024 T-3 – thickness 1.2 $mm\ = \ 0.047''$

- <u>Doubler:</u>

Clad 2024 T-3 – thickness 1.4 $mm\ = \ 0.056''$

Cutout Original Part:

$0.39\,'' \geq 0.39\,''$

$0.39\,'' \geq 0.276\,''$ Edge margin constraints are supplied in SRM 51-47-00.

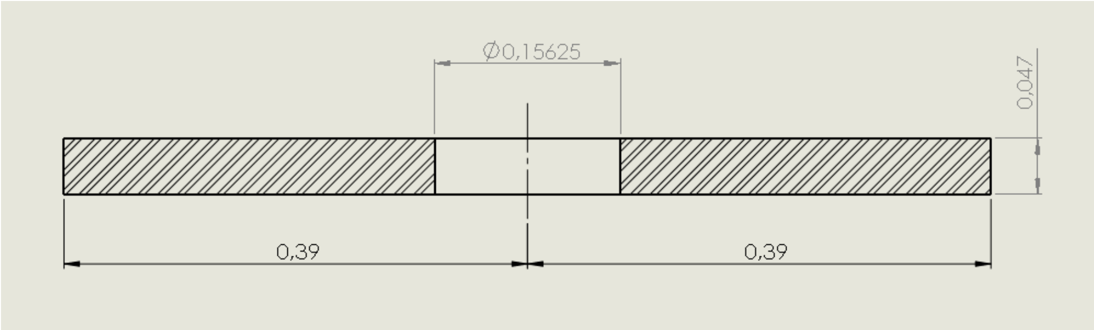


Figure 9.19 Sketch of Original Part Section (inc)

$$Section\ Area\ =\ Length\ *\ Thickness$$

$$Section\ =\ \left(0.78'' - \frac{5}{32}\right) * (0.047'') = 0.02931625\ in^2$$

$$F_{tu} = Allowable\ Tensile\ Stress = 61000\ psi$$

$$P_{tens\ max.} = Section * F_{tu}$$

$$P_{tens\ max.} = (0.02931625\ in^2) * (61000\ psi) = 1788.29\ lb$$

$$F_{su}\ = Allowable\ Ultimate\ Stress\ in\ Pure\ Shear$$

$$P_{shear\ max.} = Section * F_{su}$$

$$P_{shear\ max.} = (0.02931625\ in^2) * (38000\ psi) = 1114.02\ lb$$

Doubler Repair:

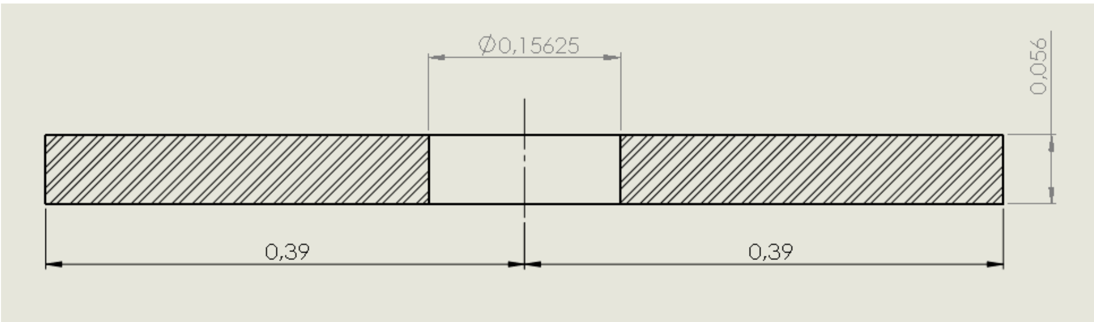


Figure 9.20 Sketch of Doubler Section (inc)

$$Section\ =\ length\ *\ thickness$$

$$Section\ =\ \left(0.78'' - \frac{5}{32}\right) * (0.056'') = 0.03493\ in^2$$

$$P_{tens\ max.} = Section * F_{tu}$$

$$P_{tens\ max.} = (0.03493\ in^2) * (61000\ psi) = 2130.73\ lb$$

$$P_{shear\ max.} = Section * F_{su}$$

$$P_{shear\ max.} = (0.03493\ in^2) * (38000\ psi) = 1327.34\ lb$$

- **Check of tensile reserve factor:**

$$Reserve\ Factor = \frac{P_{tens\ max.\ \ doubler}}{P_{tens\ max.\ \ \ orig}} = \frac{2130.73}{1788.29} = 1.19 > 1$$

- **Check of shear reserve factor:**

$$Reserve\ Factor = \frac{P_{shear\ max.\ \ doubler}}{P_{shear\ max.\ \ \ orig}} = \frac{1327.34}{1114.02} = 1.19 > 1$$

- **Combination tensile and shear check:**

The maximum (principle) stress is limited to F_{tu}. The calculation in tensile covers any combined stress in tensile and shear.

- **Compression check:**

The original section is restored by the doubler section with increased thickness and equal or better material characteristics (F_{cy} and E_c).

9.2.1.2.Determination of fastener number

$$R = \sqrt{P_{tens\ max.orig.}{}^2 + P_{shear\ max\ orig}{}^2}$$

$$R = \sqrt{1788.29^2 + 1114.02^2} = 2106.9\ lb$$

<u>*Capacity of 1 Row of 1 Fasteners:*</u>

$Shear\ Strength\ for\ MS90354 - 0502 = \ 1980\ lb$

$$1 * (2340\ lb) \ = \ 2340\ lb$$

<u>*Number of Fastener Rows:*</u>

$$Rows \geq \frac{2106.9}{2340} = 0.9$$

$$1 \geq 0.9 \quad ALLOWABLE$$

9.2.2. Boeing calculation

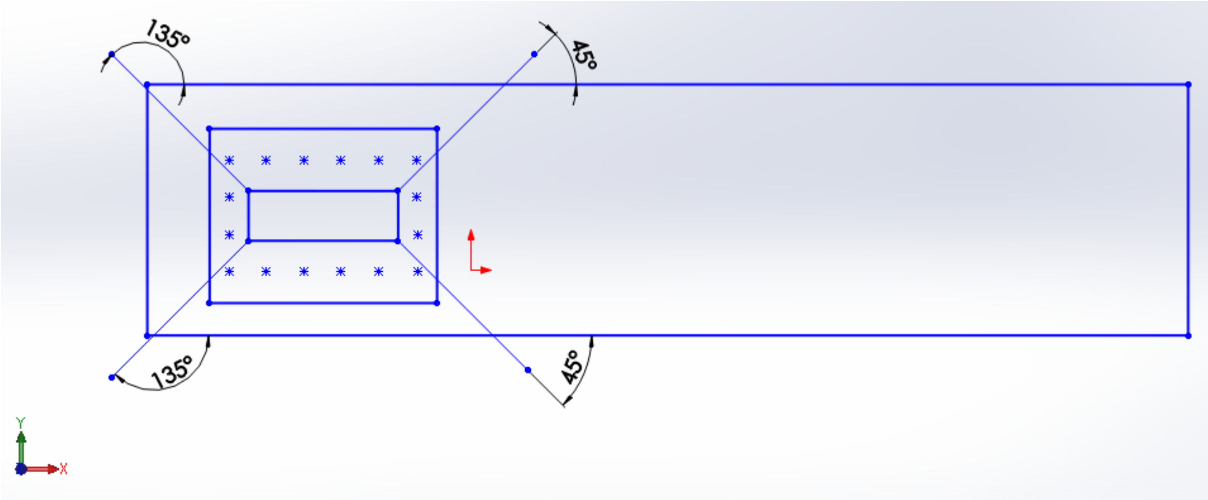


Figure 9.21 1 Row Fasteners for $3'' \ x \ 1''$ Cutout

9.2.2.1.Static strength restoration (transverse)

$$Thickness\ of\ the\ skin\ =\ 0.047\ ''\ \ (Material\ Al-T3)$$

$$Thickness\ of\ the\ repair\ doubler\ (same\ material)\ =\ 0.056''$$

$$Strength\ MS=\frac{0.056''}{0.047\ ''}-1=0.19$$

9.2.2.2.Joint capability check

Calculate Load Capacity Loss

Assuming load loss due to trim in the skin. Material: $0.047\ ''\ \ (Material\ Al-T3)$

$$A_{lost}\ =(3'')*(0.047'')=0.141\ in^2$$

$$P_{lost}\ =A_{lost}*F_{tu}\ =\ \ (0.141\ in^2)*(61000\ psi)=\ 8601\ lb$$

Total joint load is carried by 16 fasteners on the repair parts as follows:

9.2.2.3.Total joint load cap

$$P_{JT}\ =Shear\ Strength\ for\ MS21141J-0502=\ 1980\ lb$$

$$P_{JT\ (Tot.)}=6*P_{JT}\ =6*(1980\ lb)=11880\ lb$$

$$M.S.for\ Joint\ Load\ Capacity\ =\frac{P_{JT\ (Tot.)}}{P_{lost}}-1=\frac{11880}{8601}-1=0.38$$

$$0.38>\ 0\quad ALLOWABLE$$

The joint has enough load capacity to handle lost load. This repair is structurally satisfactory. This completes strength substantiation of this repair.

9.2.2.4.Static strength restoration (transverse)

$Fastener\ Pitch\ =\ 0.75\,''>4\ *d\ =\ 0.625\,''$ Allowable (in horizontal direction)

$Edge\ Margin=\ 0.39''\geq 0.39''$

$Edge\ Margin=\ 0.39''\geq 0.276''$

$Thickness\ of\ the\ skin\ =\ 0.047\,''\ \ (Material\ Al-T3)$

$Thickness\ of\ the\ repair\ doubler\ (same\ material)=\ 0.056''$

$$Strength\ MS=\frac{0.056''}{0.047\,''}-1=0.19$$

9.2.2.5.Joint capability check

Calculate Load Capacity Loss

Assuming load loss due to trim in the skin. Material: $0.047\,''\ \ (Material\ Al-T3)$

$$A_{lost}=(1'')*(0.047'')=0.047\ in^2$$

$$P_{lost}=A_{lost}*F_{tu}=\ (0.047\ in^2)*(61000\ psi)=\ 2867\ lb$$

Total joint load is carried by 16 fasteners on the repair parts as follows:

9.2.2.6.Total joint load cap

$$P_{JT}\ =Shear\ Strength\ for\ MS21141J-0502=\ 1980\ lb$$

$$P_{JT\ (Tot.)}=2*P_{JT}\ =2*(1980\ lb)=3960\ lb$$

$$M.S.for\ Joint\ Load\ Capacity\ =\frac{P_{JT\ (Tot.)}}{P_{lost}}-1=\frac{3960}{2867}-1=0.38$$

$$0.38>\ 0\quad ALLOWABLE$$

The joint has enough load capacity to handle lost load. This repair is structurally satisfactory.

This completes strength substantiation of this repair.

We can choose MS21141-0502 according to Boeing calculation. In the other hand, we can prefer MS21141-0502 or MS90354-0502 according to Airbus calculation. In conclusion, MS90354-0502 is selected because it is allowable as a result of Airbus & Boeing calculations.

9.2.3. Predictions on the fatigue life of riveted lap joints

$$K_t=\gamma*K_{t,pin}+(1-\gamma)*K_{t,hole,tension}+k*K_{t,hole,bending}\qquad [7]$$

Bending Factor:

$$k = \frac{S_{bending}}{S_{tension}} = 3 \qquad [7]$$

$\boldsymbol{K_{t,hole,tension}}$:

$$d = \frac{5}{32} = 0.15625''$$

$$pitch = 0.75\,''$$

$$\frac{d}{b} = \frac{0.15625''}{0.75''} = 0.2$$

$$K_{t,hole,tension} = 2.5$$

$\boldsymbol{K_{t,hole,bending}}$:

$$d = \frac{5}{32} = 0.15625''$$

$$pitch = 0.75''$$

$$\frac{d}{b} = \frac{0.15625''}{0.75''} = 0.2$$

$$h = 0.047''$$

$$\frac{d}{h} = \frac{0.15625''}{0.047''} = 3.32 \rightarrow \infty$$

$$K_{t,hole,bending} = 1.58$$

$\boldsymbol{K_{t,pin}}$:

$$d = \frac{5}{32} = 0.15625''$$

$$pitch = 0.75''$$

$$\frac{d}{w} = \frac{0.15625''}{0.75''} = 0.2$$

$$\frac{h}{w} = \frac{0.047''}{0.75''} = 0.0627$$

$$K_{t,pin} = 5.5$$

$$K_t = \gamma * K_{t,pin} + (1-\gamma) * K_{t,hole,tension} + k * K_{t,hole,bending} \qquad [7]$$

$\boldsymbol{\gamma}$: $the\ percentage\ of\ the\ load\ transmitted$

$\gamma = 1$ for 1 row fastener

The three stress concentration factors, $K_{t,pin}$, $K_{t,hole,tension}$ and $K_{t,hole,bending}$ depend on the joint geometry (rivet diameter / rivet pitch) . $K_{t,pin}$, $K_{t,hole,tension}$ and $K_{t,hole,bending}$ are found from Figure 9.13, 9.14 & 9.15, respectively.

$$K_t = 1 * 5.5 + 3 * 1.58$$
$$K_t = 10.24$$

9.2.4. Fatigue life calculation

$$\sigma_{hoop} = \frac{\Delta P * R}{t} = \frac{8.6 * 77.76''}{0.047''} = 14.23\ ksi \quad (Gross\ Stress)$$

$$\sigma_{net} = \sigma_{gross} * \frac{pitch}{d - pitch}$$

$$\sigma_{hoop}(net) = 14.23 * \left(\frac{0.75}{0.75 - \frac{5}{32}} \right)$$

$$\sigma_{hoop}(net) = 17.97\ ksi$$

<u>Peak Stress:</u>

$$\sigma_{peak} = \sigma_{net} * k_{t\,(net)} = (17.97\ ksi) * 10.24$$
$$\sigma_{peak} = 184\ ksi$$

<u>MMPDS:</u>

S – N curve for notched specimen for $K_t = 5$

$$S_{peak} = 184\ ksi$$
$$S_{peak_{corrected}} = \frac{S_{eq}}{K_t} = \frac{184\ ksi}{5} = 36.8\ ksi$$

$$logN_f = 8.9 - 3.73 * log(S_{eq} - 3.9)$$

$$S_{eq} = S_{max} * (1 - R)^{0.56}$$

$$S_{eq} = S_{peak_{corrected}} = 36.8\ ksi$$

$$logN_f = 8.9 - 3.73 * log(36.8 - 3.9)$$

$$N_f = 1741 \quad Cycles$$

9.3. Case 3

Dimensions used for repair design are shown in Figure 9.22 (Cutout Size = $1.5'' x\ 1.5''$).

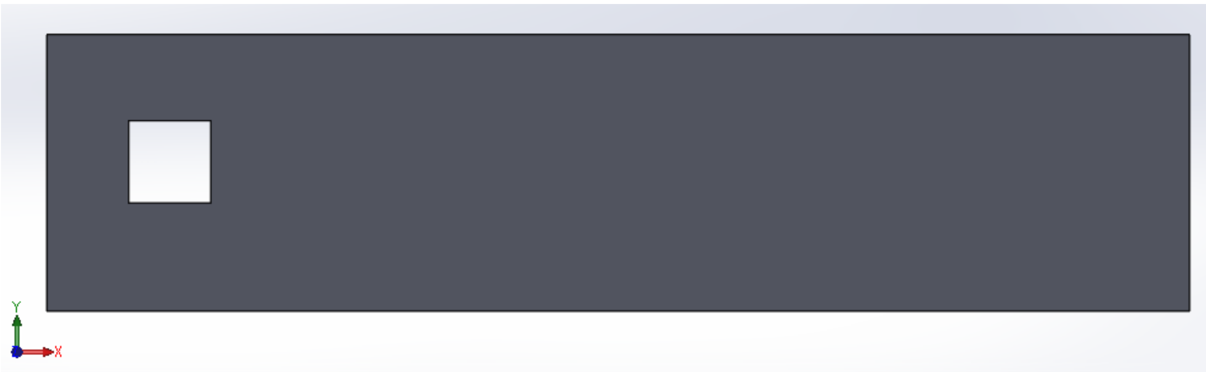


Figure 9.22 Skin Cut-out ($1.5'' x\ 1.5''$)

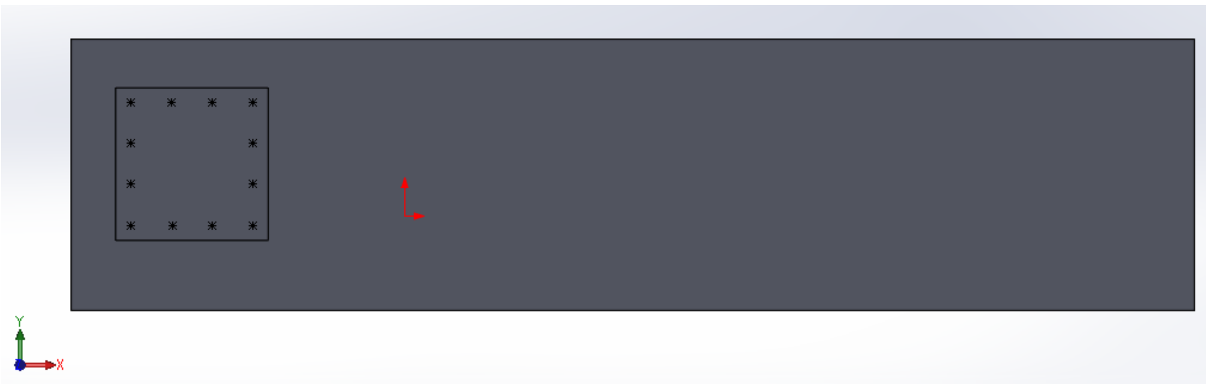


Figure 9.23 Assembly Doubler ($1.5'' x\ 1.5''$)

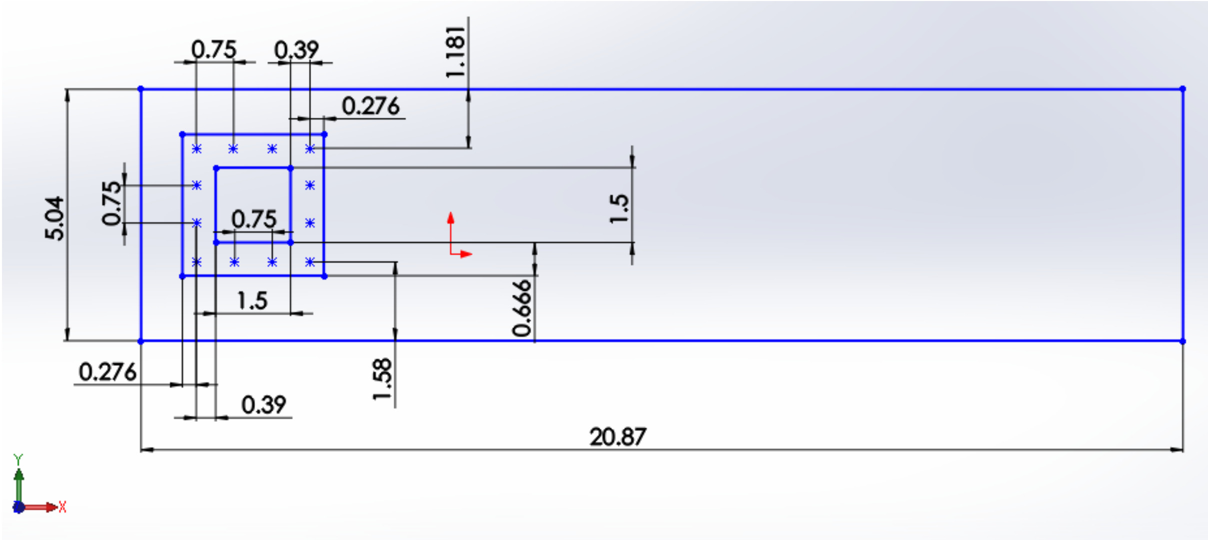


Figure 9.24 Dimensions of New Repair Design (1.5″x 1.5″)

$Fastener\ Pitch\ =\ 0.75''$

$Edge\ Margin\ =\ 2*d\ =\ 0.3125''\ <\ 0.39''$

9.3.1. Airbus calculation

9.3.1.1.Material capability (tension check)

- Original Part:

Clad 2024 T-3 – thickness 1.2 $mm\ =\ 0.047''$

- Doubler:

Clad 2024 T-3 – thickness 1.4 $mm\ =\ 0.056''$

Cutout Original Part:

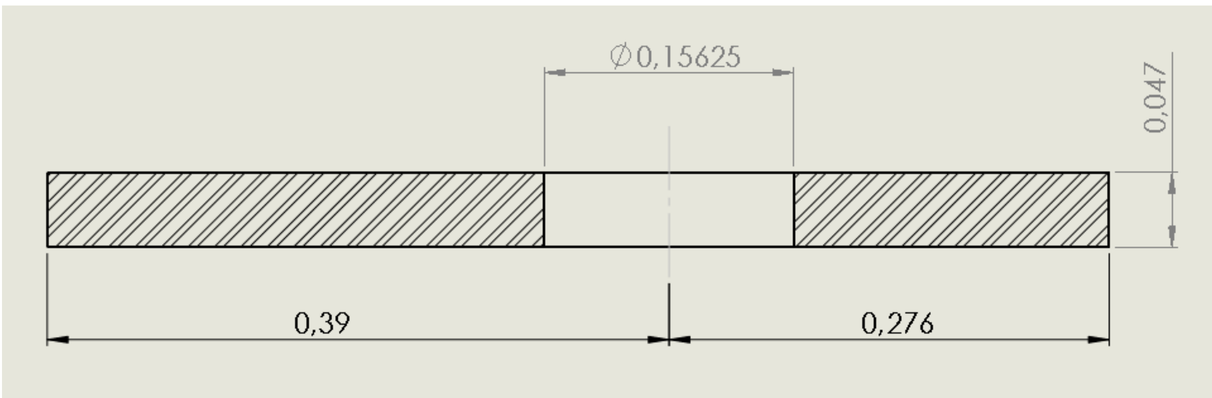


Figure 9.25 Sketch of Original Part Section (inc)

$$Section\ Area\ =\ Length\ *\ Thickness$$

$$Section\ =\ \left(0.666''-\frac{5}{32}\right)*(0.047'')=0.023958\ in^2$$

$$F_{tu}=Allowable\ Tensile\ Stress=61000\ psi$$

$$P_{tens\ max.} = Section * F_{tu}$$

$$P_{tens\ max.} = (0.023958\ in^2) * (61000\ psi) = 1461.45\ lb$$

$$F_{su} = Allowable\ Ultimate\ Stress\ in\ Pure\ Shear$$

$$P_{shear\ max.} = Section * F_{su}$$

$$P_{shear\ max.} = (0.023958\ \ in^2) * (38000\ psi) = 910.41\ lb$$

Doubler Repair:

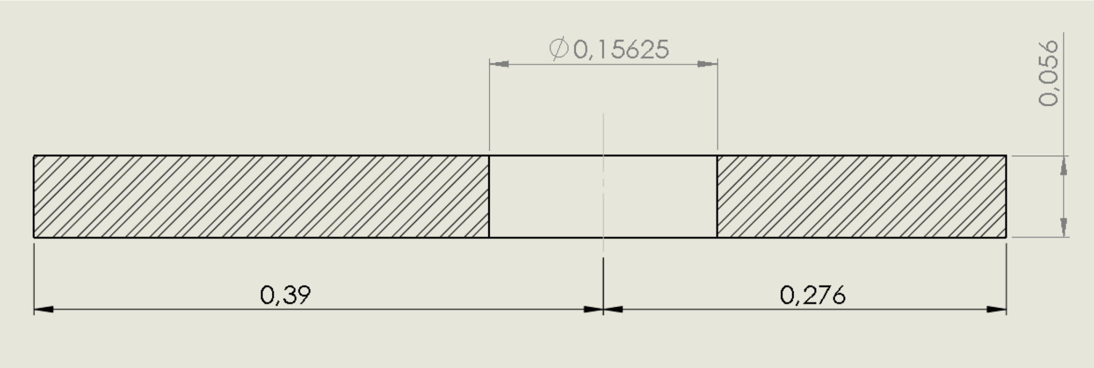


Figure 9.26 Sketch of Doubler Section (inc)

$$Section\ =\ length\ *\ thickness$$

$$Section\ =\ \left(0.666'' - \frac{5}{32}\right) * (0.056'') = 0.028546\ in^2$$

$$P_{tens\ max.} = Section * F_{tu}$$

$$P_{tens\ max.} = (0.028546\ in^2) * (61000\ psi) = 1741.306\ lb$$

$$P_{shear\ max.} = Section * F_{su}$$

$$P_{shear\ max.} = (0.028546\ in^2) * (38000\ psi) = 1084.748\ lb$$

- **Check of tensile reserve factor:**

$$Reserve\ Factor = \frac{P_{tens\ max.\ \ doubler}}{P_{tens\ max.\ \ \ orig}} = \frac{1741.306}{1461.45} = 1.19 > 1$$

- **Check of shear reserve factor:**

$$Reserve\ Factor = \frac{P_{shear\ max.\ \ doubler}}{P_{shear\ max.\ \ \ orig}} = \frac{1084.748}{910.41} = 1.19 > 1$$

- **Combination tensile and shear check:**

The maximum (principle) stress is limited to F_{tu}. The calculation in tensile covers any combined stress in tensile and shear.

- **Compression check:**

The original section is restored by the doubler section with increased thickness and equal or better material characteristics (F_{cy} and E_c).

9.3.1.2.Determination of fastener number

$$R = \sqrt{P_{tens\ max.orig.}{}^2 + P_{shear\ max\ orig}{}^2}$$

$$R = \sqrt{1461.45^2 + 910.41^2} = 1721.83\ lb$$

<u>*Capacity of 1 Row of 1 Fasteners:*</u>

$Shear\ Strength\ for\ MS21141 - 0502 = \ 1980\ lb$

$$1 * (1980\ lb) \ = \ 1980\ lb$$

<u>*Number of Fastener Rows:*</u>

$$Rows \geq \frac{1721.83}{1980} = 0.87$$

$$1 \geq 0.87 \quad ALLOWABLE$$

If $MS21141 - 0502\ bears\ the\ total\ load, MS21141 - 0602$ can bear because its shear strength is more than other.

9.3.2. Boeing calculation

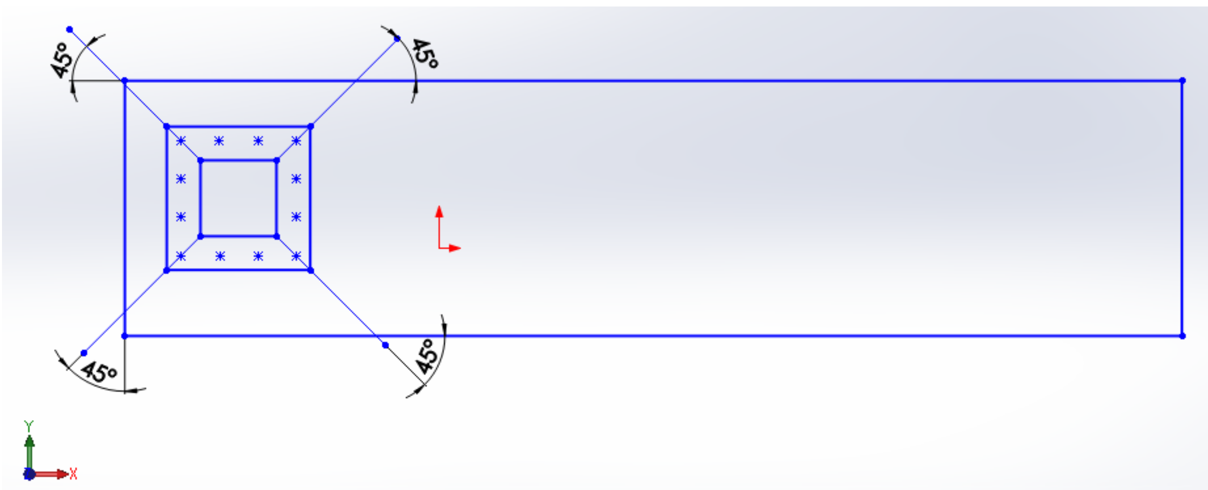


Figure 9.27 1 Row Fasteners for $1'' \ x\ 1''$ Cutout

9.3.2.1. Static strength restoration

$$Thickness\ of\ the\ skin\ =\ 0.047\,''\ \ (Material\ Al-T3)$$

$$Thickness\ of\ the\ repair\ doubler\ (same\ material)\ =\ 0.056''$$

$$Strength\ MS = \frac{0.056''}{0.047\,''} - 1 = 0.19 > 0$$

9.3.2.2. Joint capability check

Calculate Load Capacity Loss

Assuming load loss due to trim in the skin. Material: $0.047\,''\ \ (Material\ Al-T3)$

$$A_{lost} = (1.5'') * (0.047'') = 0.0705\ in^2$$

$$P_{lost} = A_{lost} * F_{tu} =\ (0.0705\ in^2) * (61000\ psi) =\ 4300.5\ lb$$

Total joint load is carried by 12 fasteners on the repair parts as follows:

9.3.2.3. Total joint load cap

$$P_{JT}\ = Shear\ Strength\ for\ MS21141 - 0602 =\ 2925\ lb$$

$$P_{JT\ (Tot.)} = 2 * P_{JT}\ = 2 * (2925\ lb) = 5850\ lb$$

$$M.S. for\ Joint\ Load\ Capacity\ = \frac{P_{JT\ (Tot.)}}{P_{lost}} - 1 = \frac{5850}{4300.5} - 1$$

$$= 0.36\quad ALLOWABLE$$

The joint has enough load capacity to handle lost load. This repair is structurally satisfactory. This completes strength substantiation of this repair. MS21141-0602 has strength according to 2 calculations.

9.3.3. Predictions on the fatigue life of riveted lap joints

9.3.4. Fatigue life calculation

$Pitch\ =\ 0.75\,''$ pitches are the same in the case 1, 2 & 3, so their life

$$N_f = 1741\ Cycles$$

9.4. Case 4

Dimensions used for repair design are shown in Figure 9.30 (Cutout Size = $3'' x\ 1.5''$).

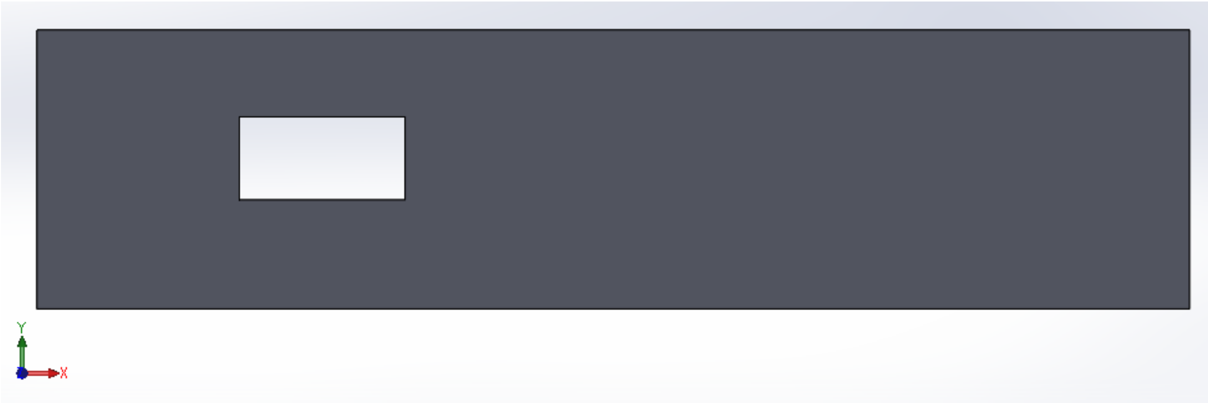


Figure 9.28 Skin Cutout ($3'' x\ 1.5''$)

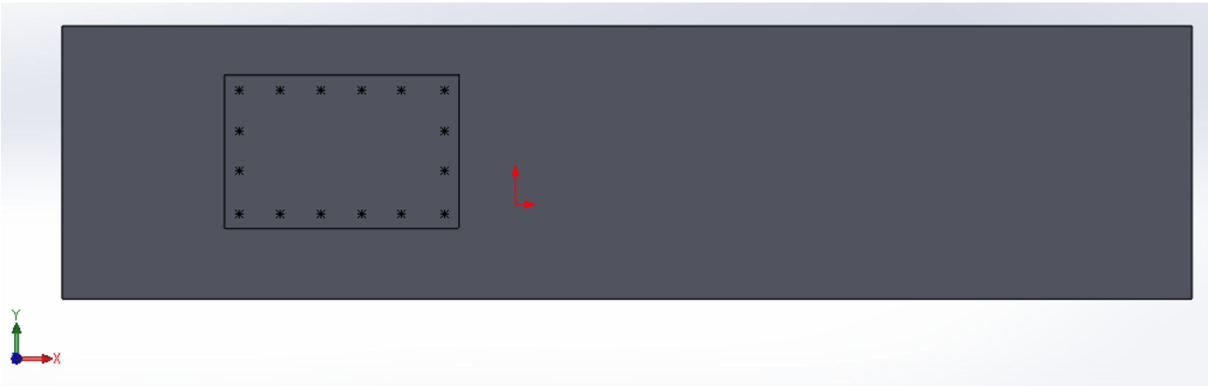


Figure 9.29 Assembly Doubler & 1 Row Fasteners for $3''\ x\ 1.5''$ Cutout

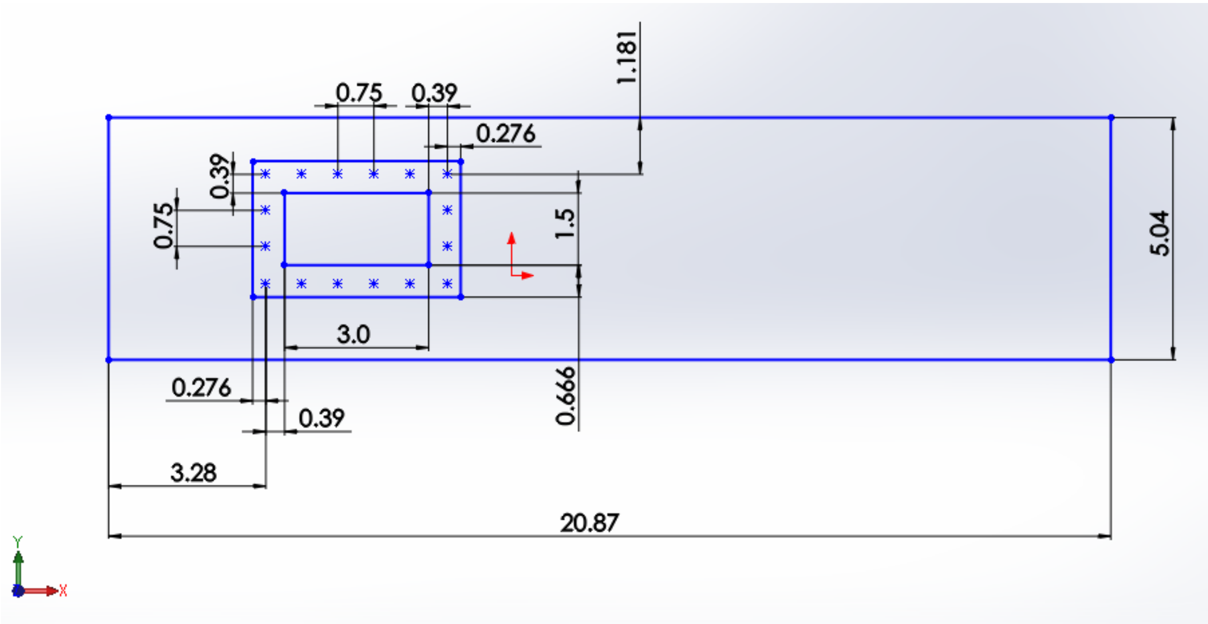


Figure 9.30 Dimensions of the New Repair Design

In The Longitudinal Direction

$Fastener\ Pitch\ =\ 0.75\,'' > 4 * d\ =\ 0.625\,''$ (allowable)

9.4.1. Airbus calculation

9.4.1.1.Material capability (tension check)

- Original Part:

Clad 2024 T-3 – thickness 1.2 $mm\ =\ 0.047''$

- Doubler:

Clad 2024 T-3 – thickness 1.4 $mm\ =\ 0.056''$

Cutout Original Part:

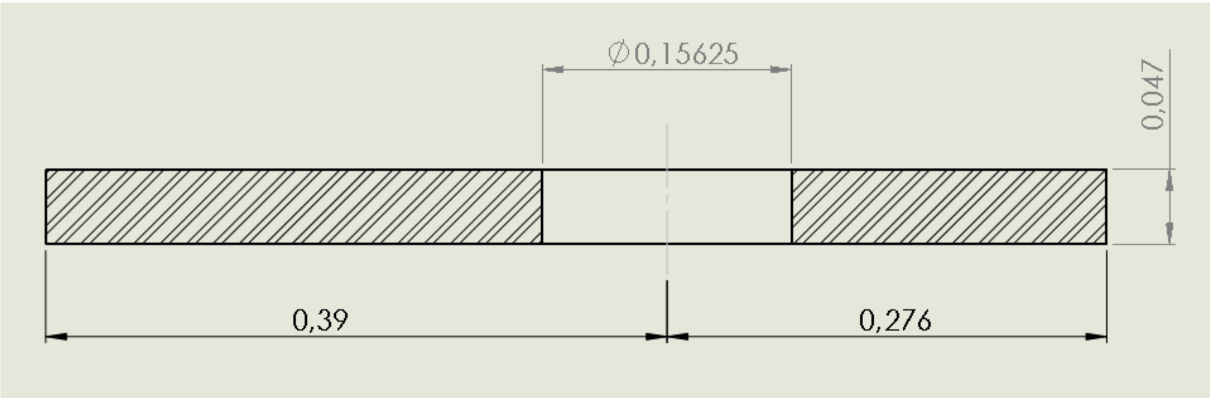


Figure 9.31 Sketch of Original Part Section (inc)

$$Section\ Area\ =\ Length\ *\ \mathrm{Thickness}$$

$$Section\ =\ \left(0.666'' - \frac{5}{32}\right) * (0.047\,'') = 0.023958\ in^2$$

$$F_{tu} = Allowable\ Tensile\ Stress = 61000\ psi$$

$$P_{tens\ max.} = Section * F_{tu}$$

$$P_{tens\ max.} = (0.023958\ in^2) * (61000\ psi) = 1461.45\ lb$$

$$F_{su}\ = Allowable\ Ultimate\ Stress\ in\ Pure\ Shear$$

$$P_{shear\ max.} = Section * F_{su}$$

$$P_{shear\ max.} = (0.023958\ \ in^2) * (38000\ psi) = 910.41\ lb$$

Doubler Repair:

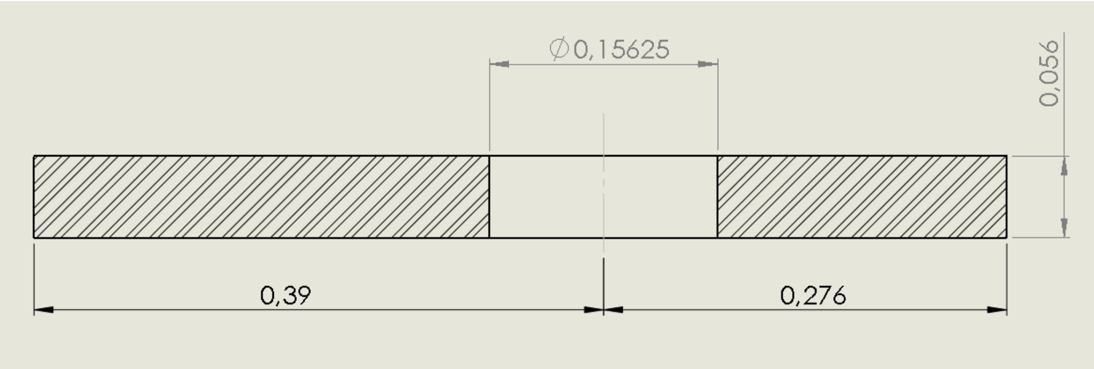


Figure 9.32 Sketch of Doubler Section (inc)

$$Section = length * thickness$$

$$Section = \left(0.666'' - \frac{5}{32}\right) * (0.056'') = 0.028546\ in^2$$

$$P_{tens\ max.} = Section * F_{tu}$$

$$P_{tens\ max.} = (0.028546\ in^2) * (61000\ psi) = 1741.306\ lb$$

$$P_{shear\ max.} = Section * F_{su}$$

$$P_{shear\ max.} = (0.028546\ in^2) * (38000\ psi) = 1084.748\ lb$$

- **Check of tensile reserve factor:**

$$Reserve\ Factor = \frac{P_{tens\ max.\ \ doubler}}{P_{tens\ max.\ \ orig}} = \frac{1741.306}{1461.45} = 1.19 > 1$$

- **Check of shear reserve factor:**

$$Reserve\ Factor = \frac{P_{shear\ max.\ \ doubler}}{P_{shear\ max.\ \ orig}} = \frac{1084.748}{910.41} = 1.19 > 1$$

- **Combination tensile and shear check:**

The maximum (principle) stress is limited to F_{tu}. The calculation in tensile covers any combined stress in tensile and shear.

- **Compression check:**

The original section is restored by the doubler section with increased thickness and equal or better material characteristics (F_{cy} and E_c).

9.4.1.2.Determination of fastener number

$$R = \sqrt{{P_{tens\ max.orig.}}^2 + {P_{shear\ max\ orig}}^2}$$

$$R = \sqrt{1461.45^2 + 910.41^2} = 1721.83\ lb$$

<u>Capacity of 1 Row of 1 Fasteners:</u>

$Shear\ Strength\ for\ MS21141 - 0502 = \ 1980\ lb$

$$1 * (1980\ lb)\ = \ 1980\ lb$$

<u>Number of Fastener Rows:</u>

$$Rows \geq \frac{1721.83}{1980} = 0.87$$

$$1 \geq 0.87 \quad ALLOWABLE$$

<u>Capacity of 1 Row of 1 Fasteners:</u>

$Shear\ Strength\ for\ MS90354 - 0502 = \ 2340\ lb$

$$1 * (2340\ lb)\ = 2340\ lb$$

<u>Number of Fastener Rows:</u>

$$Rows \geq \frac{1721.83}{2340} = 0.736\ \ and\ \ \ 1 \geq 0.736 \quad ALLOWABLE$$

9.4.2. Boeing calculation

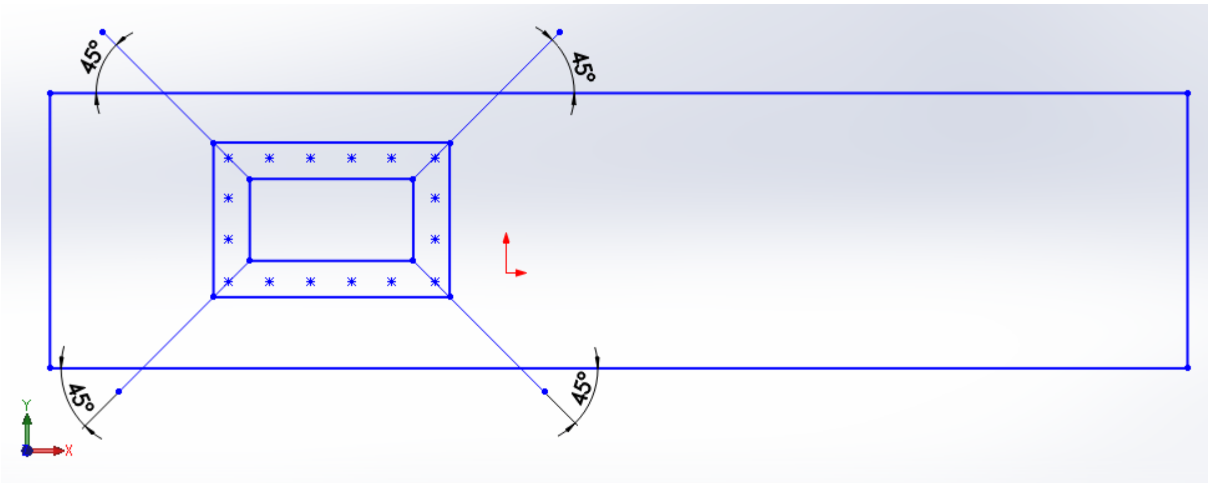


Figure 9.33 1 Row Fasteners for $1'' \ x\ 1''$ Cutout

9.4.2.1.Static strength restoration (longitudinal)

$$Thickness\ of\ the\ skin\ =\ 0.047\,''\ \ (Material\ Al - T3)$$

$$Thickness\ of\ the\ repair\ doubler\ (same\ material) =\ 0.056''$$

$$Strength\ MS = \frac{0.056''}{0.047\,''} - 1 = 0.19 > 0$$

9.4.2.2.Joint capability check

Calculate Load Capacity Loss

Assuming load loss due to trim in the skin. Material: $0.047\,''\ \ (Material\ Al - T3)$

$$A_{lost} = (1.5'') * (0.047'') = 0.0705\ in^2$$

$$P_{lost} = A_{lost} * F_{tu} =\ (0.0705\ in^2) * (61000\ psi) =\ 4300.5\ lb$$

Total joint load is carried by 12 fasteners on the repair parts as follows:

9.4.2.3.Total joint load cap

$$P_{JT}\ = Shear\ Strength\ for\ MS90354 - 0502 =\ 2340\ lb$$

$$\mathrm{P}_{JT\ (Tot.)} = 2 * P_{JT}\ = 2 * (2340\ lb) = 4680\ lb$$

$$M.S.for\ Joint\ Load\ Capacity\ = \frac{P_{JT\ (Tot.)}}{P_{lost}} - 1 = \frac{4680}{4300.5} - 1$$

$$= 0.088 \quad ALLOWABLE$$

The joint has enough load capacity to handle lost load. This repair is structurally satisfactory. This completes strength substantiation of this repair.

9.4.2.4.Static strength restoration (transverse)

$$Thickness\ of\ the\ skin\ =\ 0.047\,''\ \ (Material\ Al - T3)$$

$$Thickness\ of\ the\ repair\ doubler\ (same\ material) =\ 0.056''$$

$$Strength\ MS = \frac{0.056''}{0.047\,''} - 1 = 0.19 > 0$$

9.4.2.5.Joint capability check

Calculate Load Capacity Loss

Assuming load loss due to trim in the skin. Material: $0.047\,''\ \ (Material\ Al - T3)$

$$A_{lost} = (3'') * (0.047'') = 0.141\ in^2$$

$$P_{lost} = A_{lost} * F_{tu} =\ (0.141\ in^2) * (61000\ psi) =\ 8601\ lb$$

Total joint load is carried by 12 fasteners on the repair parts as follows:

9.4.2.6.Total joint load cap

$P_{JT} = Shear\ Strength$ for MS90354 − 0502 = 2340 lb

$$\mathrm{P_{JT\ (Tot.)}} = 4 * \mathrm{P_{JT}} = 4 * (2340\ \mathrm{lb}) = 9360\ \mathrm{lb}$$

$$\mathrm{M.S.\ for\ Joint\ Load\ Capacity} = \frac{\mathrm{P_{JT\ (Tot.)}}}{\mathrm{P_{lost}}} - 1 = \frac{9360}{8601} - 1 = 0.088$$

$$0.088 > 0 \quad ALLOWABLE$$

The joint has enough load capacity to handle lost load. This repair is structurally satisfactory.

This completes strength substantiation of this repair.

9.4.3. Predictions on the fatigue life of riveted lap joints

Pitch $= 0.75''$ pitches are the same in the case 1, 2 & 3, so their life

$$\mathrm{N_f} = 1741\ \mathrm{Cycles}$$

10. COMPAIRE OF THE FLIGHT CYCLES

Large commercial airplanes are designed for a minimum of 20 years of economic service life with a high level of reliability. 20 years of Design Service Goal for A321 is 48.000 FC.

Cutout Dimension	Types of Fasteners	Flight Cycles
$1'' x\ 1''$	MS21141 – 0502	2122 FC
$3'' x\ 1''$	MS90354 – 0502	1741 FC
$1.5''\ x\ 1.5''$	MS21141 – 0602	1741 FC
$3''\ x\ 1.5''$	MS21141 – 0502	1741 FC

It is important that the size of repair area for flight cycles of aircraft. If a fuselage has a big cutout, its flight cycles will be less.

11. COMPAIRE OF THE COSTS OF FASTENERS

11.1.Available Design

In the available design, Structural Repiar Manuel (SRM) proposes NAS 1097 DD5 for related repair. 1 libre 1097 DD5 costs 9 $.

11.2.The New Design

In the new design, we offer to repair design with blind bolts; these are MS90354-0502 and MS21141J-0502. A piece of rivets costs 2 $.

MS90354-0502 is 2 $ (piece of rivet)

MS21141J-0502 is 2 $ (piece of rivet) [6]

When we compare fasteners costs with the ground cost, we prefer the new repair design. The cost of the two products is almost same.

12. RESULTS

Standard repairs are generally given in repair manuals. However, in many cases in-service inspections show damage that is not covered by standard repair manuals and a special repair has to be designed. For such cases detailed static and damage tolerance analyses have to be carried out. Examples of cracked fuselage in a commercial aircraft is given in this study. Standard repair manuals generally do not cover a repair for the damage.

The fatigue limit of joints can be very low due to severe stress concentrations, fretting corrosion and secondary bending. In spite of a high static strength of a joint, the fatigue limit can be low. Prediction of the fatigue life, fatigue strength and fatigue limit is a complex problem for joints.

This study is aimed to ability to repair. In this regard, it investigates the cut-out repairs with doubler on aircraft fuselage. Detailed calculation procedure of a skin repair is presented. After then, two different specific skin repair hand calculations are performed; these are Airbus and Boeing calculations. When these results are compared, we obtain the same results. Hence, THY Technic acquires the ways of repair design for the different size and different locations of repair scenarios on aircraft fuselages.

THY Technic provides increasing of customer satisfaction, a significant competitive advantage and confidence increases. Furthermore, it provides decreasing of staying time on the another airport (e.g. Adana, Bitlis), operating costs, ground costs and hotel & other costs of passengers and workers.

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CURRICULUM VITAE

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B.Sc.:

- ✓ Istanbul Technical University, Mechanical Engineering, Machine Design Department
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Professional Experience and Rewards:

I am working in Turkish Technic Inc., R&D Center as a design engineer now. I have prepared various projects in both Turkish Technic Inc. and the university. In the university, first of all, the Green Shopping Mall project was awarded the first prize by Eskişehir Chamber of Mechanical Engineers in May of 2011. This project was published in “Mühendis & Makina” Journal in May of 2013 (640th Edition). Second of all, the unmanned, tactic and amphibious armored vehicle project was awarded the third prize by Eskişehir Chamber of Mechanical Engineers in May of 2011. This project was published in “Savunma & Havacılık” Journal in November of 2011 (145th Edition). Last but not at all least, I prepared a project about the rear rim of ARIBA-V which is the ITU Solar Car. I designed and analyzed it in terms of structure.

In addition, ‘Unmanned Special Purpose Amfibic Stealth Tactical Vehicle’ which is published as a poster presentation in ESINKAP (11.05.2011) and Project Fair (25.05.2011) while I was a student in Eskişehir Osmangazi University.

I can use Abaqus, Ansys, Catia, Solidworks, Autocad, Solidedge, Matlab-Simulink, Mathcad, Adams-View, Adams-Car, MSC SimXpert / Motion, Nastran SimXpert / Structures, Microsoft Office. I also used these programs in my projects.

My duties are researching and developing ideas for new products and production systems, besides using computer-aided design to create detailed designs and specifications.