BRNO UNIVERSITY OF TECHNOLOGY
VYSOKÉ UČENÍ TECHNICKÉ V BRNĚ

FACULTY OF MECHANICAL ENGINEERING
FAKULTA STROJNÍHO INŽENÝRSTVÍ

INSTITUTE OF AEROSPACE ENGINEERING
LETECKÝ ÚSTAV

JOINING OF CFRP AND METAL PARTS
SPOJE KOMPOZITNÍCH A KOVOVÝCH ČÁSTÍ

BACHELOR'S THESIS
BAKALÁŘSKÁ PRÁCE

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Zadání bakalářské práce

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Ředitel ústavu Vám v souladu se zákonem č.111/1998 o vysokých školách a se Studijním a zkušebním řádem VUT v Brně určuje následující téma bakalářské práce:

Spoje kompozitních a kovových částí

Stručná charakteristika problematiky úkolu:

Proveďte rozbor mechanických spojů mezi kompozitní a kovovou nosnou konstrukcí. Rozbor směřujte na aplikaci spoje mezi kovovým okem závěsu hlavního nosníku křídla a pásnicí hlavního nosníku křídla. Zvažte namáhání konstrukčního detailu, typickou konstrukci a použité materiály. Zvažte více typů spoje a navrhněte konstrukční a technologický demonstrátor spoje pro experimentální zkoušky.

Cíle bakalářské práce:

1. Rozbor požadavků kladených na lepené spoje.
2. Popis požadavků pro spoj závěsu nosníku s pásnicí nosníku křídla.
3. Návrh jedné a více variant spoje.
4. Návrh vzorku pro prototypové zkoušky.

Seznam literatury:

ASM Handbook Volume 21: Composites. Editors: D.B. Miracle and S.L. Donaldson. ASM International, 2001.

Niu, M. C. Y. (1992). Composite Airframe Structures: Practical Design, Information and Data. Conmilit Press Ltd, Hong Kong.

Department of Defense Handbook. Composite Materials Handbook Volume 3: Polymer Matrix Composites Materials Usage, Design, and Analysis (MIL-HDBK-17-3F), 2002.

Termín odevzdání bakalářské práce je stanoven časovým plánem akademického roku 2015/16

V Brně, dne

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doc. Ing. Jaroslav Juračka, Ph.D. ředitel ústavu	doc. Ing. Jaroslav Katolický, Ph.D. děkan fakulty

Summary

This thesis describes basic principles of joining advanced composites together and also joining advanced composites to metal counterparts. The main goal of this thesis is to come up with suitable solution of the joint between metal hinge and CFRP wing spar for flying car AeroMobil 3,0. For calculation of stress-strain analysis a special analytical method is used, which was performed by means of computer software MATLAB and Excel. Final design of the joint is performed by aid of computer software Autodesk Inventor. Submitted solution is not practically tested, hence it has to be considered as a first approximation of future design.

Keywords

adhesive bonding, mechanical fastening, CFRP, advanced composite materials, metal hinge of wing, wing spare

DOLINSKÝ, J. *Joining of CFRP and metal parts*. Brno: Brno University of Technology, Faculty of Mechanical Engineering, 2016. 61 p. Supervisor Volodymyr Symonov, M.Sc.

Declaration

I declare that I have written the bachelor's thesis *Joining of CFRP and metal parts* on my own according to advice of my bachelor's thesis supervisor Volodymyr Symonov MSc., and using the sources listed in references.

Brno 22. 5. 2016

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Jiří Dolinský

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Contents

Introduction

Nowadays, aeronautical engineers face an interesting issue. Modern technology is able to manufacture very light and simultaneously high-strength and high-stiff composite materials. With the help of these materials, they are already able to produce many components for aircrafts. Since that these components usually exhibit much better mechanical properties than components commonly used before, it is a huge promise for the future of aeronautical industry. Findings mentioned above also caused that much emphasis is being directed toward replacing metal components in weight sensitive structures. Unfortunately, development of technologies related with advanced composites is very complicated and expensive, thus engineers often must look for an alternative solution.

One of the main problems of structures made of composites is to perform stress-strain analysis in the joints. It is quite a complex problem which requires a complex approach including at least stress analysis methodology, methodology for estimation of special composite material's properties and taking into account production technology aspects. There are some very precise approaches to this issues, but all of them are too complex and bulky for engineering application. This is mainly caused by heterogeneity of composites, which evinces anisotropy[1] and many other unusual properties, which have to be included in the mathematical model.

This thesis aims to sum up all so far known information about requirements applied to mechanical fastened and adhesive bonded joints. It also aims to describe analytical approach to stress-strain analysis of a composite-metal joint, which can be used to estimate approximate stress in the bonded parts and adhesives. Last of all, suitable design of joint between metal hinge and CFRP wing spar for flying car AeroMobil 3,0 is going to be designed and it also will be accompanied by stress analysis.

[1] Anisotropic material does not have the same properties in all directions [20].

1 Aerospace applications of composite materials

FIBER-REINFORCED COMPOSITES have become an increasingly attractive alternative to metal for many aircraft components [1]. The reason of that is mainly weight saving, which is achieved when composite materials are used. Another advantages of FRP[2] are: better strength and stiffness, corrosion resistance, improved resistance to fatigue, better thermal insulation or thermal conductivity (depends on what is needed), etc. However, usually it is impossible to achieve all of mentioned advantages within one structure, moreover, usually it is not required. In fact, some of the properties are in conflict with one another, e.g., thermal insulation versus thermal conductivity. The objective is merely to create a material that has only the characteristics needed to perform the design task [2]

The composite material system being considered consist of parallel, high strength fibers, supported in a relatively ductile matrix material. The fibers act as load carriers while matrix serves principally as a load transfer medium between fibers [3]. The composite materials used in the aircraft industry are generally reinforcing fibers or filaments embedded in a resin matrix. The most common fibers are carbon, aramid, and fiberglass, used alone or in hybrid combinations. Carbon fiber is replacing fiberglass as the most widely used reinforcement. The resin matrix is usually an epoxy-based system requiring curing temperatures between RT[3] and 180 °C [1].

Plies, which are single layers of parallel fibers surrounded by the matrix material, are stacked at various orientations into one part called laminate. This is the procedure which enables the designer to achieve the desired strength and stiffness properties and to increase the structural efficiency of a given amount of material [3]. Unfortunately, strength and stiffness properties of composite laminate are highly directional and designer always have to include this fact in his calculations.

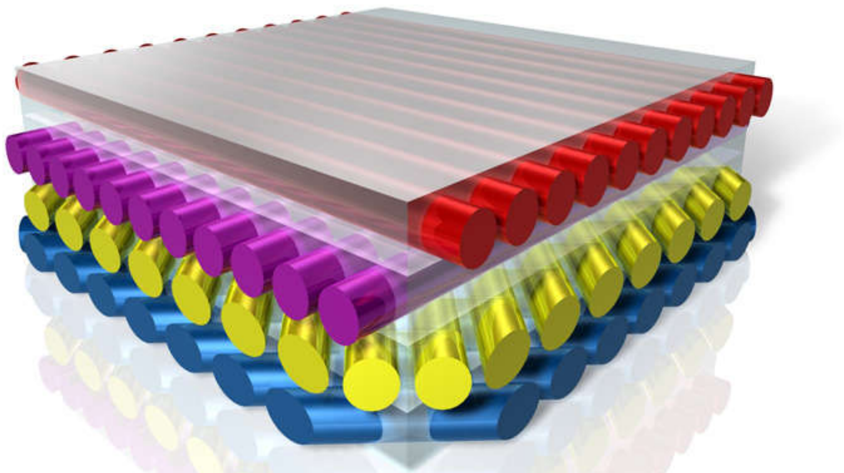

Fig. 1-1Illustration of possible composite structure.

In early days of composites applications, these materials were used just for a few components such as vertical tails, horizontal tails, doors or fairings. Over time, the

[2] FRP – fiber-reinforced plastic
[3] RT – room temperature

application of composites was extended to almost the entire airplane. An example might be a Eurofighter Typhoon shown in Fig. 1-2.

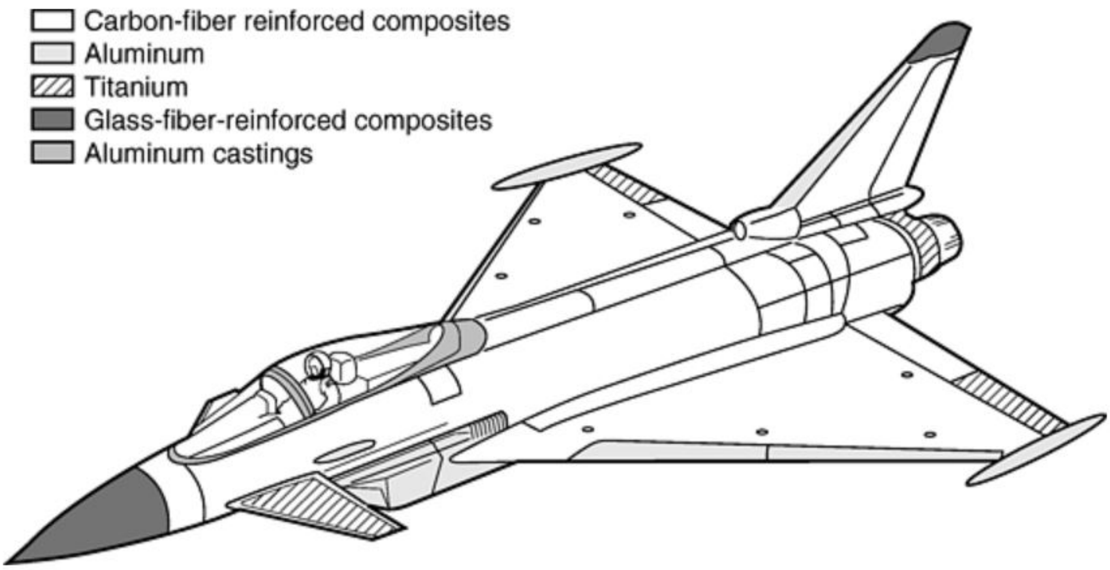

Fig. 1-2 Eurofighter Typhoon

Another example of composite application might be the best-selling jet commercial airliner Boeing 737 and its spoiler designed by NASA, shown in Fig. 1-3.

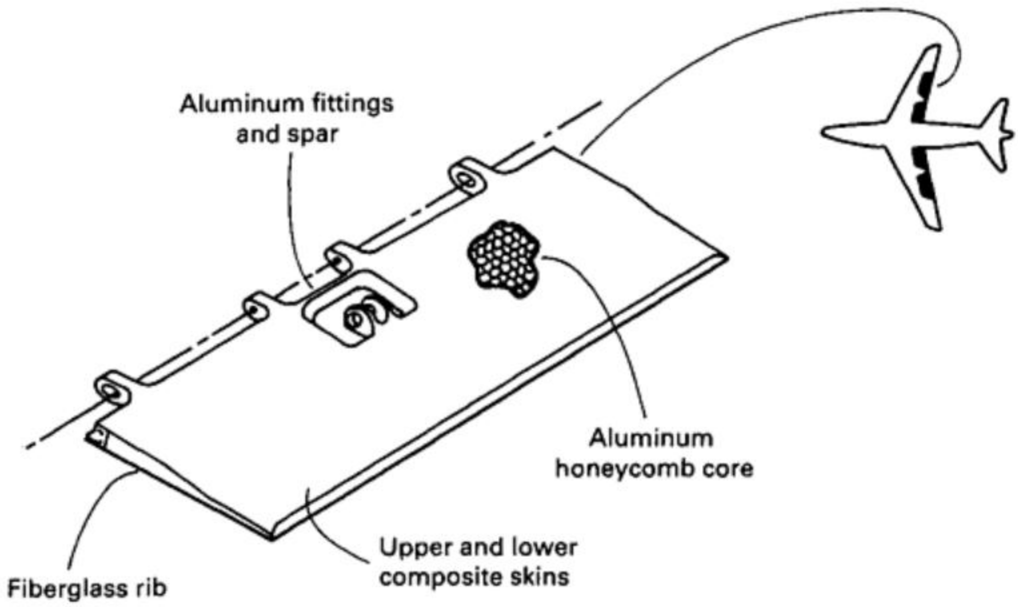

Fig. 1-3 Boing 737/NASA composite spoiler

Examples mentioned above are just two of the thousands others. The rest of this thesis is going to be focused directly on joining of composite components with their composite or metal counter parts, because this is still serious problem of structures made of advanced composites.

2 Mechanical fastening

2.1 Introduction

Mechanical fastening is the most common method for assembling of airframe structures. It stands to reason that first attempt of joining composites was mechanical fastening as well, but the design of an efficient bolted joint of composite materials is much more complex than that of metals, for three major reasons:

- Laminate composite exhibit unique failure modes not found in metals
- The material properties of composites can be allowed to vary along the length of the joint [3]
- Material is relatively brittle, which results in high stress concentration at hole edges

Design of mechanical joints of composite materials require to take into account the next considerations. Composite materials do not respond to fastener in the same way as metals, thus it is not possible to design fasteners that are generally applicable to all composites. Even though composites are very strong, they can be also very delicate if they are not treated properly. It is caused by their characteristics. As a consequence, this means the selection of the correct fasteners is tremendously important.

An element of a typical mechanical joint is show in the following figure, which also defines all of the key dimensions.

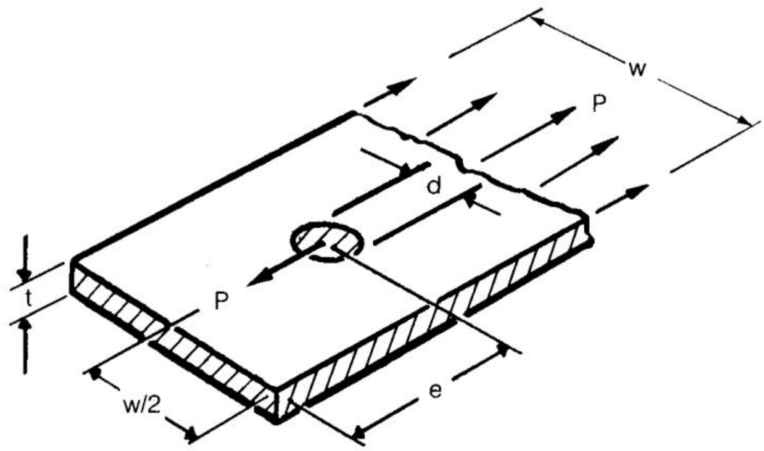

Fig. 2-1 Typical Mechanical joint element

Where:

d – fastener diameter, e – edge distance, t – laminate thickness, w – width, P – load

Although many assembly problems have been solved with adhesive-bonding techniques, there are a lot of cases where only mechanical joints are capable of meeting design requirements. For example, parts requiring replacement or removal for ease of fabrication or repair [4].

2.2 Failure modes

The principal failure modes of bolted joints are: bearing failure of the material as in the elongated bolt hole of Fig. 2-4 c), tension failure of the material in the reduced cross section through the hole (net tension), shear-out or cleavage failure of the material (actually transverse tension failure of the material), bolt failures (mainly shear failures) and combination of these failures. One of the way to increase the bearing strength of a joint is to use metal inserts as in the shimmed joint of Fig. 2-2 b). Another way is to thicken a section of the composite laminate as in the reinforced-edge joint in Fig. 2-2 a).

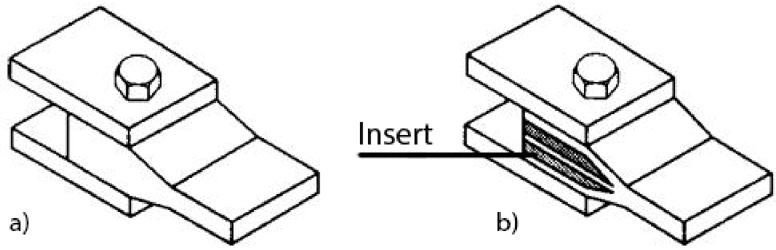

Fig. 2-2 Reinforced-Edge joint and Shimmed joint

Net-tension failures can be avoided or delayed by increased joint flexibility to spread the load transfer over several lines of bolts. Composite materials are generally more brittle than conventional metals, so loads are not easily redistributed around a stress concentration such as a bolt hole. Simultaneously, shear-lag effects caused by discontinuous fibers lead to difficult design problem around bolt holes. A possible solution is to put a relatively ductile composite material such as S-glass epoxy in a strip of several times the bolt diameter in line with the bolt rows. This approach is called the softening strip concept[4] [2].

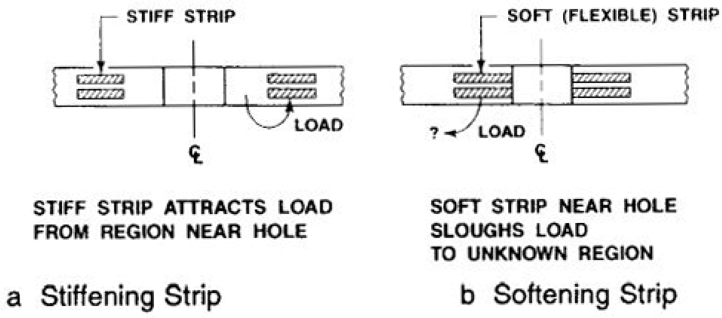

Fig. 2-3 Concepts For Stress Concentration Reduction around Holes

The following figure presents mentioned failure modes, which can occur in mechanically fastened composite parts:

[4] The overall stress concentration effect around holes in composite laminates can be reduced in two different manners that are unique to composite materials and have no analog in metal structures practice. The first way is called the Stiffening Strip Concept that consist of placing strops of a stiffer composite material in a region away from the hole to attract load (away from the hole boundary). There, we know where the load will be taken in laminate. In contrast, in the second way, the Softening Strip Concept, a strip of composite material that is less stiff (softer) is placed right beside each hole to slough the load that would ordinarily be concentrated near the hole to some other region of laminate. However, we do not know where that load will be carried, just where it will not be carried. Both concepts are commonly used in design practice [2].

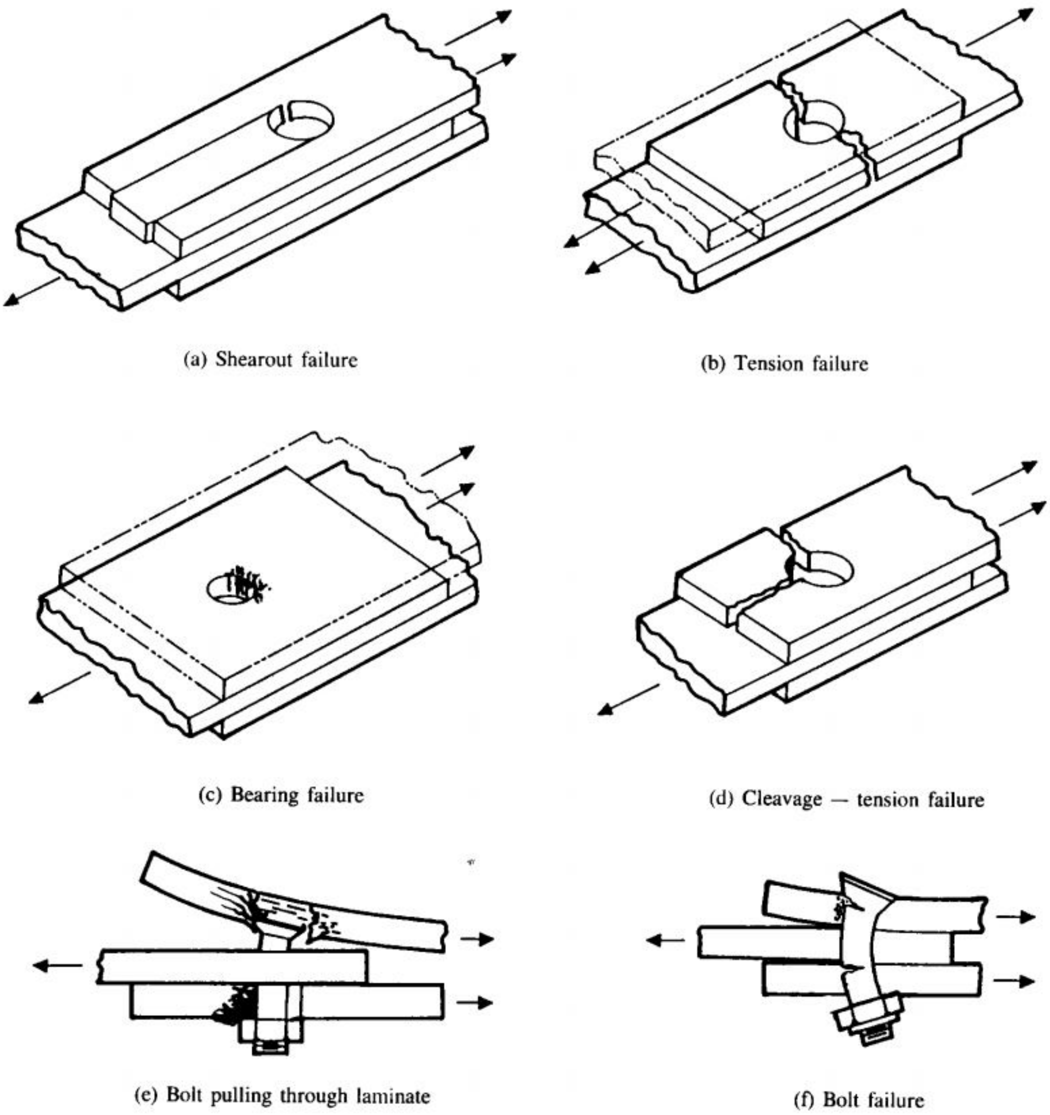

Fig. 2-4 Failure modes of advanced composites mechanical joints

Perhaps the most important thing to understand the mechanically fastened joints of fibrous composite structures is that the eventual failure of the composite occurs long after the laminate has stopped behaving like the one-phase homogeneous engineering material for which it is usually modeled. While the fibers and the resin matrix are both essentially linear until failure, the microcracks and delamination around bolt holes in composite laminates cause substantial internal load redistributions that are not accounted for in conventional mathematical models of bolted or riveted composite joints. Thus, there appears to be substantial nonlinear behavior associated with the normal rivet or bolt sizes used in composite structures. However, while there are indeed softened zones at the microlevel, as shown in Fig. 2-5, this softening is unlike the yielding associated with ductile metal alloys in similar circumstances.

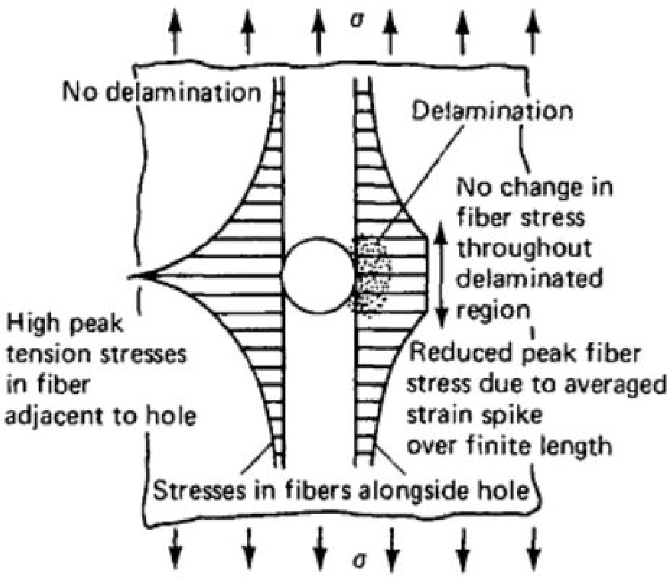

Fig. 2-5 Stress concentration relief at small hole in fiber-reinforced composites

The literature on bolted composite joints contains another basic approach to this nonlinearity problem, usually referred to as the characteristic-length or characteristic-offset approach. The origins of this approach are the point-stress and average-stress failure criteria. With that method, the linearly elastic analysis is presumed to be valid outside some empirically determined softened zone adjacent to the hole. The basic drawback to that approach, which can, of course, always be shown to be capable of explaining any test results one at a time, is that the so-called characteristic dimension varies considerably with bearing stress, and that failure is being predicted at some place other than where it is known to occur. Nevertheless, both methods of analysis will continue to be used until it is possible to cover all joint geometries and bearing stress intensities with a single theory. At present, each approach covers some situations not covered by the other. Both have been used successfully in hardware applications, and both have led to increased understanding of stress concentrations around bolt holes in composite structures [1].

2.3 Delamination

The delamination is a separation of the layers of material one from another in a laminate. This may be local or may cover a large area of the laminate. It may occur at any time during the curing or subsequent life of the laminate and may arise from a wide variety of causes.

The formation and growth of the delamination is generally related to the LSS[5] and matrix characteristics. The delamination can have varying effects on tensile strength performance, depending on the delamination location and the specific property of interest. The most studies performed to date have considered specimens with significant free edge surface area where the interlaminar stresses are known to concentrate. Although all structures have some free edges, it is important to realize the limits of analysis and tests performed with specimen geometries. For example, the

[5] LSS – laminate stacking sequence

magnitude of the interlaminar tensile stresses, which are crucial to the edge delamination, approach zero for plate width to thickness ratios of 30 and greater.

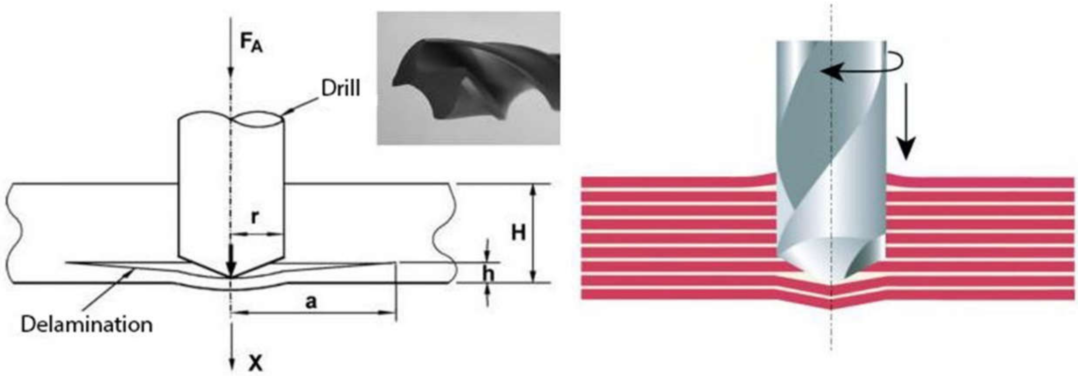

Fig. 2-6 Formation of delamination during drilling of the hole.

The reduced laminate stiffness due to edge delamination can affect the measured tensile strength in two distinct ways [5].If all plies remain loaded after delamination, the ultimate laminate strain has been found to be equal to the critical strain of the primary load bearing plies. In these cases, laminate strength drops in proportion to the apparent axial modulus. However, if off-axis plies cease to carry loads because they have been isolated by an interconnected network of matrix cracks and delamination, a local strain concentration can form. When this occurs, the global laminate strain for failure can be less than the critical strain of primary load bearing plies [6].

2.4 Recommended design practices for mechanically fastened joints

- Stress concentrations exert a domination influence on the magnitude of the allowable design tensile stress. Generally, only 20 – 50% of the basic laminate ultimate tensile strength is developed in mechanical joint
- Mechanically fastened joints should be designed so that the critical failure mode is in bearing, rather than shear out or net tension, so that catastrophic failure is prevented. This will require an edge distance to fastener diameter ratio (e/d) and a side distance to fastener diameter (s/d) relatively greater than those for conventional metallic materials. At relatively low e/d and s/d ratios, failure of the joint occurs in shear out at the ends, or in tension at the net section. Considerable concentration of stress develops at the hole, and the average stresses at the net section at failure are a fraction of the basic tensile strength of the laminate
- Multiple rows of fasteners are recommended for unsymmetrical joints, such as single shear lap joints, to minimize bending included by eccentric loading
- Local reinforcing of unsymmetrical joints by arbitrarily increasing laminate thickness should generally be avoided because the resulting eccentricity can give rise to greater bending stress which counteracts or negates the increase in material area
- Since stress concentrations and eccentricity effects cannot be calculated with a consistent degree of accuracy, it is advisable to verify all critical joints designs by testing a representative sample joint

Fig. 2-7 presents minimal recommended fastener spacing and edge distance as a function of the diameter. It is necessary to note that these values are just indicative and depends on many factors such as tensile strength of the fibers or their orientation.

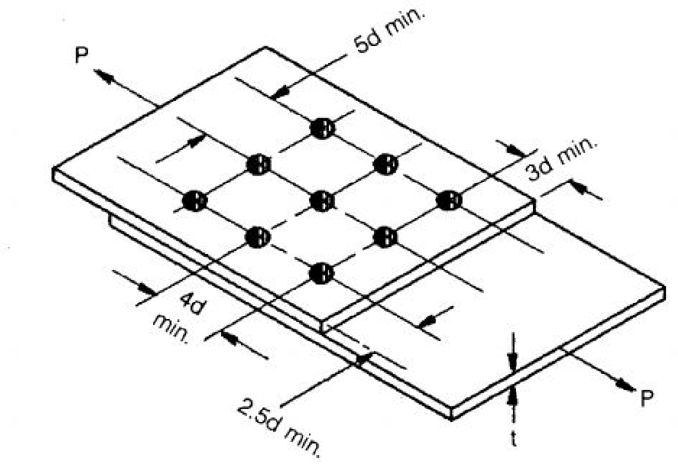

Fig. 2-7 Minimal Fastener Spacing and Edge Distance

In general, the best fastened joints of fibrous composites still cause a strength loss of about half the basic material strength [4].

3 Adhesive bonding

3.1 Introduction

Bonding of discrete panels and joining of separately fabricated components are standard practices for composite structures [4]. The main reason of using this method is that adhesive joints are capable of high structural efficiency and constitute a resource for structural weight saving because of the potential for elimination of stress concentrations which cannot be achieved with mechanically fastened joints [7].

Adhesively bonded joints can be strong in shear but are inevitably weak in peel, so the objective of good design practice is to arrange the joint to transfer the applied load in shear and to minimize any direct or induced peel stresses[6]. The thinner members can be joined effectively by simple, uniformly thick overlaps, while thicker members require the more complex stepped-lap joints [1].

In spite of weakness in peel adhesive bonding is often considered as the best way to permanently join composites to each other. Adhesives have many advantages over mechanical fastening for composite assembly. They can often be of the same family of materials as the matrix, which ensuring compatibility. They resist corrosion, seal as well as join, and weigh less than fastened joining. They can create a strong joint, with stress distributed over large areas, and chiefly don't require drilling. In following sub-section, we are going to focus on commonly used types of adhesive bonded joints [4].

Unfortunately, because of lack of reliable inspections methods and a requirement for close dimensional tolerances in fabrication, aircraft designers have generally avoided bonded joints in primary structures. One of the first application were bonded step lap joints used in attachments for the F-14 and F-15 horizontal stabilizers as well as the F-18 root fitting [7]. Nowadays, the best examples of application of adhesive bonding joints are Boeing B787 Dreamliner and Airbus A350WXB, which are mainly made of CFRP.

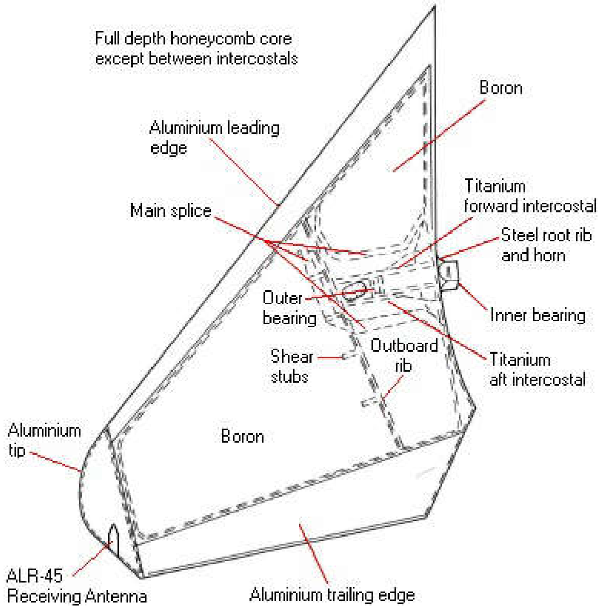

Fig. 3-1 The F-14 Composite Horizontal Stabilizer

[6] Peel stress – Transverse normal stress [7]

3.2 Adhesively bonded joint types

Laminated panels are usually bonded by a cocuring[7] operation or by using adhesives. Some typical examples of that are shown in Fig. 3-2.

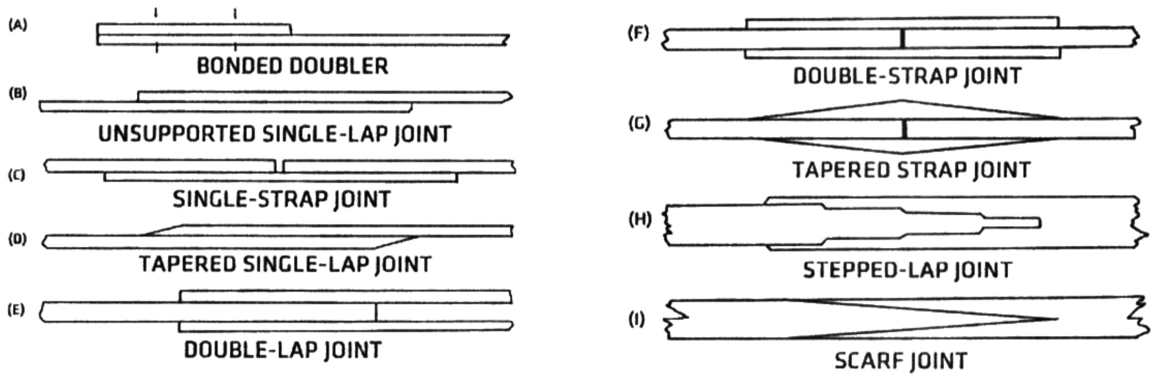

Fig. 3-2 Types of Bonded Joints

Basically, the thinner members can be joined effectively by simple, uniformly thick overlaps, while thicker members require the more complex stepped-lap joints [1]. This statement will be more explained below.

The strength of the adhesive bond is independent of the overlap, but the overlap has to be set at a sufficiently high value that the adhesive in the middle of overlap will not creep under the worst of circumstances. Short overlaps could permit failure by creep-rupture; longer overlaps do not add to the strength or durability of the bonded joint [4].

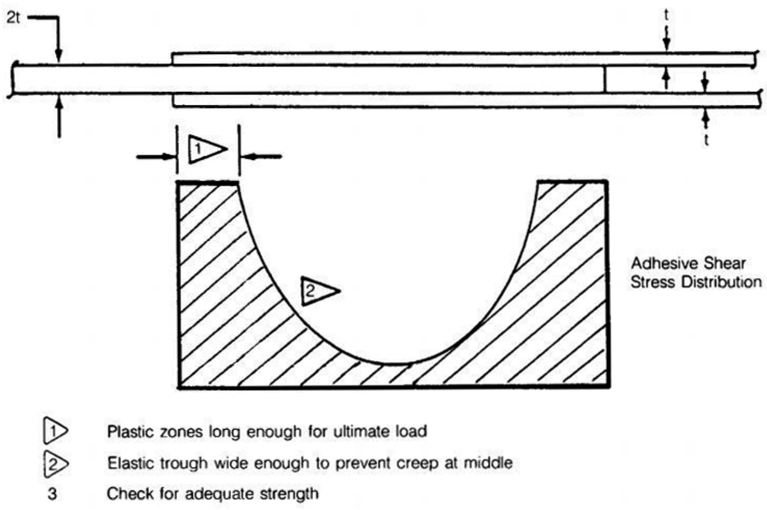

Fig. 3-3 Illustration of Elastic Trough of Double-lap Joint

The load is actually transferred through two end zones with a lightly-loaded elastic trough in-between, as shown in Fig. 3-3. The extent of these zone is defined largely by the adhesive plasticity. Increasing the total overlap of uniform thickness adherends, for all but very short overlaps, merely moves the effective end zones further apart without changing the load transferred or the maximum adhesive stresses and

[7] Cocuring is defined as the curing of laminate with stiffeners and attachments all in one operation [4].

strains developed. Scarf and stepped-lap joints differ a little from uniform lap joints, since, although they also contain limited plastic zones of adhesive, the load transferred by the elastic zone increases indefinitely with longer overlaps. The dominant characteristic of scarf joints is that, regardless of all other factors, the ratio of the average bond shear stress to the peak bond shear stress equals the lower ratio of the adherend extensional stiffness. This is why such joints retain their effectiveness for thicker sections than can be joined efficiently by uniform lap joints. [8].

The lightly loaded trough in the diagram should not be considered as a joint inefficiency to be eliminated by improved design. Its presence is vital to ensuring an adequate resistance to failure of the joint by creep-rupture. The total overlap length must be sufficient to ensure that the adhesive shear stress in the middle of the overlap is essentially zero or at least so low that creep cannot occur.

A designer also should be aware that adhesives with the highest strength do not always produce the highest joint strength. It is caused by effect of ductility, which improves the load distribution by reducing stress peaking at the end of joint. The relative comparison between a brittle adhesives and ductile adhesives is shown in Fig. 3-4 [4].

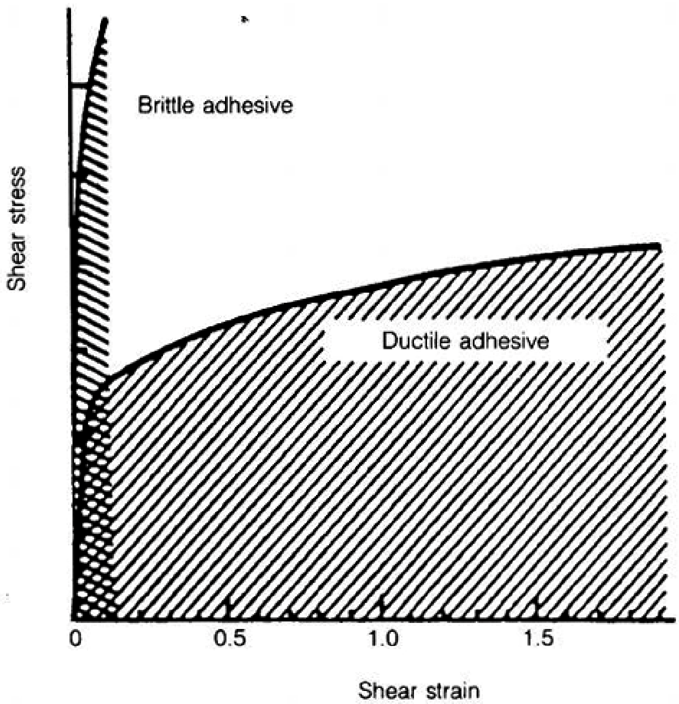

Fig. 3-4 Comparison of shear behavior between brittle and ductile adhesives

3.3 Adherend Failures and Bond Failures

Adhesive joints in general are characterized by high stress concentrations in the adhesive layer and the primary function of these joints is to transfer load by shear. Shear stresses arise, because of unequal axis straining of the adherends. However, in bonded joints also arise peel stress, which is caused by eccentricity in the load path. As was mentioned above, considerable ductility is associated with shear response of typical adhesives, which is beneficial in minimizing the effect of shear stress on joint strength. Response to peel stresses tends to be much more brittle than that to shear stresses, and reduction of peel stresses is desirable for achieving good joint performance [4; 7].

From the standpoint of reliability, it is vital to avoid letting the adhesive layer be the weak link in the joint; this means that, whenever possible, the joint should be designed to ensure that the adherends fail before the bond layer. This is because failure in the adherends is fiber controlled, while failure in the adhesives is resin dominated, and thus subject to effects of voids and other defects, thickness variations, environmental effects, processing variations, deficiencies in surface preparation and other factors that are not always adequately controlled. This is significant challenge, since adhesives are inherently much weaker than the composite or metallic elements being joined. However, the objective can be accomplished by recognizing the limitations of the joint geometry being considered and placing appropriate restrictions on the thickness dimensions of the joint for each geometry.

Fig. 3-5 illustrates a development of joint types which represent increasing strength capability from lowest to the highest in the figure.

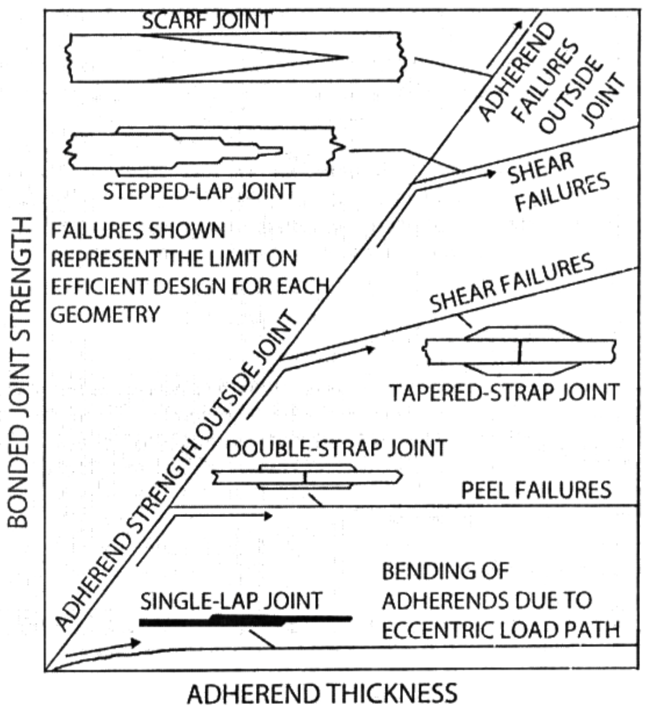

Fig. 3-5 Impact of joint geometry on strength of bond.

The thickness of adherend can be increased in order to gain higher ultimate load. Nevertheless, when the adherend is relatively thin, it guarantees that adherend will reach their allowable stress before the failure could occur in adhesive bond. As the adherend thicknesses increase, the bond stresses become relatively larger until a point is reached at which bond failure occurs at a lower load than that for which adherends fail. To avoid this issue, maximum thickness of adherend has to be restricted to an appropriate range relatively to the layer thickness of adhesive. These considerations result in specific range of adherend thickness for each of the joint types and when this range is exceeded, it is necessary to change the joint configuration to one of higher efficiency rather than to increasing the adherend thickness indefinitely [7].

3.4 Influence of the joint geometry on the strength

3.4.1 Single-lap joint and Double-lap joint

At the very beginning it is necessary to state that every step change of thickness of adherend causes stress concentration, as is shown in Fig. 3-6.

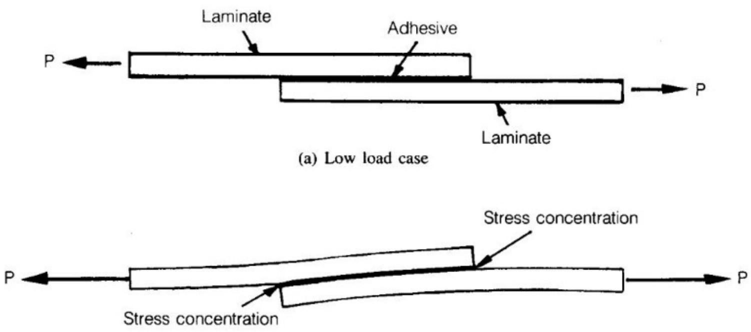

Fig. 3-6 Single-lap joint Vs. Load case

The low interlaminar tension strength of composite laminates limits the thickness of the adherends which can be bonded together efficiently by lap joints. The inner laminate splits apart locally due to peel stresses, thereby destroying the shear transfer capacity between the inner and outer plies. This overloads the outer filaments, which break in tension, and the failure can progress through the material. The effect of this failure mode is to restrict the thickness of composite which can be bonded efficiently using standard double-lap and single-lap joints [8]. This failure procedure is shown in following Fig. 3-7, where Fig. 3-7 a) shows delamination of plies, Fig. 3-7 b) shows fiber breakage and finally Fig. 3-7 c) shows failure initiated by peel stress.

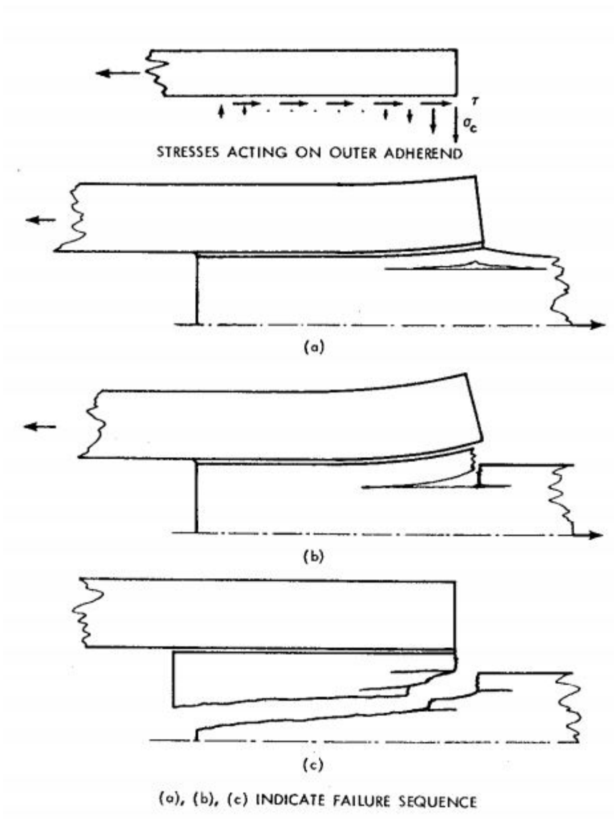

Fig. 3-7 Peel stress failure of thick composite joints

Single and double lap joints with uniform adherends (Fig. 3-2 – Joints (B), (E) and (F)) are the least efficient joint type and are suitable primarily for thin structures with low running loads[8] [7].

In case of single-lap joint the maximum stresses within outside the unsupported single-lap joint are influenced greatly by the value of the bending moment induced, just outside the overlap, by the eccentricity in the load path [8]. However, use of this joint relatively effective at location where a transverse member exists to react bending moment, as shown in Fig. 3-8. Single lap joints also require greater overlap lengths to alleviate adherend bending stresses associated with the eccentricities in the load path.

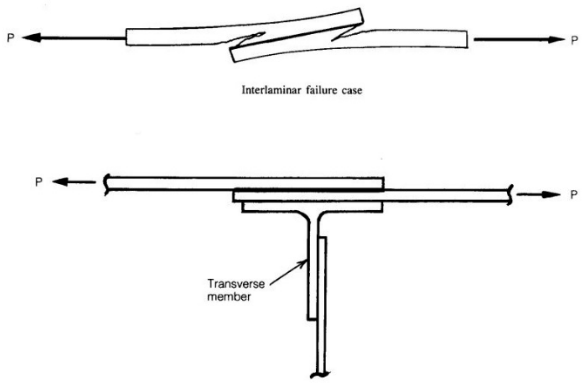

Fig. 3-8 Single-lap joint with Transverse Member as Supporting Structure

In case of double-lap joint, it is possible to use a little bit thicker adherends compared to a single-lap joint. Actually, the shear load transfer is accomplished through a narrow effective zone at the ends of the overlap. The shear stress distribution in the middle of the adhesive carries the same small load until the limit length is reached. If the overlap length is less than the limit length, then shear stress depends on the joint length. This joint is also highly tolerant of manufacturing imperfections throughout its structural efficient range. As shown in Fig. 3-9, the strength of double-lap joint is affected adversely by the presence of laminate stiffness unbalance of joint[9] [4].

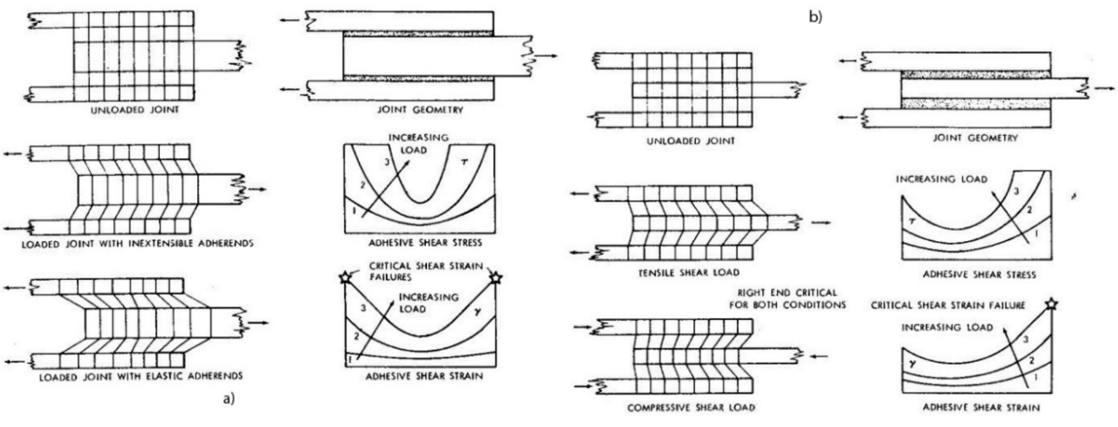

Fig. 3-9 a) Balanced and b) Unbalanced double-lap joint

---

8 Load per unit width, i.e., stress times element thickness

9 A joint is balanced if the extensional stiffness of the adherends carrying the load in the two directions are equal. An example of unbalanced joint is shown in Fig. 3-9 b), where all three adherends have similar stiffness so the single one in the center carrying the load to the right has only about half the stiffness of the outer pair transmitting load to the left [4].

3.4.2 Tapered joint

The simple design modifications that reduce the peel stresses to insignificance is to make the tips of the adherends thin and flexible [1]. This method originated from consideration that for very thin adherends, the induced adhesive peel stresses are so small as to be negligible and vice versa as the thickness of the adherends is increased, the peel stresses become significant until they actually detract from the shear strength of the adhesive bond [9]. Another method of improving joint properties is thickening of adhesive layer towards the end of joint. This is also beneficial, but such local thickening of the adhesive layer must be used with caution with high-flow heat-cured adhesives, which could create voids by capillary action [1]. Both methods are shown in Fig. 3-10.

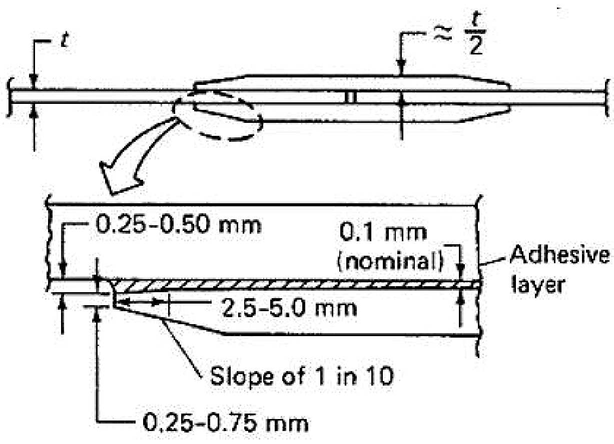

Fig. 3-10 Tapering of edges of splice plates to relieve adhesive peel stresses

No tapering is needed at ends of the overlap where adherends butt together because the transverse normal stress at that location is compressive and rather small. Likewise, for double strap joints under compressive loading, there is no concern with peel stresses at either location since the transverse extensional stresses that do develop in the adhesive are compressive in nature rather than tensile; indeed, where the gap occurs the inner adherends bear directly on each other and no stress concentrations are present there for the compression loading case [7].

3.4.3 Scarf Joint

The shear stress in the adhesive of a scarf joint, which is subjected to membrane applied loads, is virtually constant. This means that a sufficiently small scarf angle can be selected to provide 100% joint efficiency (the joint adhesive strength is equal to the laminate failure load) [4].

In spite of the theoretical potential for complete elimination of stress concentration, actual joint efficiencies are less than 100% because the manufacturing constraints often require angles greater than optimal, which is caused by some minimum thickness corresponding to one or two ply thicknesses that has to be incorporated at the thin end of the scarfed adherend leading to the occurrence of stress concentrations in these sections. Theoretically, any required load capacity could be achieved in the scarf joint by making the joint thick enough and also long enough. Nevertheless, real scarf joints are usually less durable because of a tendency toward creep failure associated with a uniform distribution of shear stress along the length of the joint unless emphasis is not being directed toward avoiding stresses into the nonlinear range of adhesive (refer to chapter 3.2).

Result of these considerations is that the scarf joint tends to be used just for repairs of very thin structures. In addition, unbalanced scarf joints do not achieve the uniform shear stress condition and are somewhat less structurally efficient. This is caused by rapid increase of load near the thin end of the adherend [7].

3.4.4 Stepped-lap joint

Composite laminates that are too thick and hence too strong to be joined by simple uniform lap-splice bonded joints can be bonded together successfully by stepped-lap joints of the type shown in Fig. 3-2 h) and like the scarf joint, the stepped lap joint offers considerably greater efficiencies for thick laminates than is possible with single or double-lap joints. This join type basically represents a practical solution to the challenge of bonding thick members and also of bonding metal to composite. Stepped-lap joint shares some of the characteristics of both the double-lap joint and scarf joint. The joint strength can be increased by increasing number of steps and approximating more closely the scarf joint design, which leads to another advantage of this joint – stepped-lap joint utilizes layered structure of composite laminates. The layers are arranged into required shape and then cocured into one part. However, once the maximum number of steps possible has been attained by equaling the number of plies in the laminate, no further strength increase can be developed, but the strength may be also increased by tapering of the outer adherends of the laminates in the overlap (refer to chapter 3.4.2) [1; 4; 7].

The critical location in this joint design is the tail end of the thinnest step as shown in Fig. 3-11. In this location very often occurs fatigue failure [4].

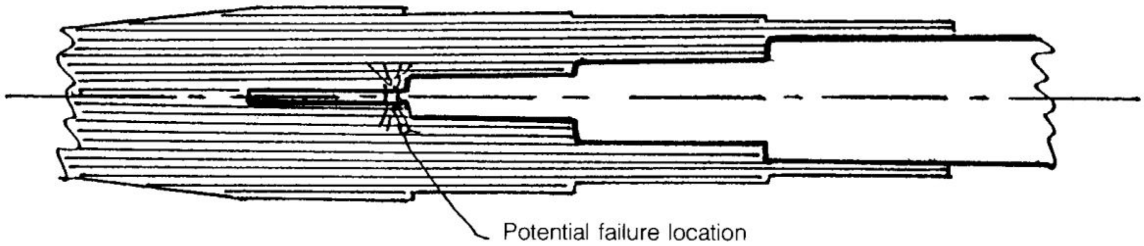

Fig. 3-11 Potential Failure Location of Stepped-lap joint

3.5 Surface Treatment

Surface treatment of joining parts is one of the most important requirement applied to adhesive bonding and it is even more important when different materials are joined, e. g. such as metal and composite.

3.5.1 Surface treatment for bonding composite to composite

By using surface treatment, the following improvements can be reached, which results in better bond performance:

- elimination of 'weak boundary layers' at the bonding surfaces such as contaminants, oxidized layers, low-molecular-weight species and loose, friable, surfaces
- improved wetting of low-energy surfaces
- chemical modification such as the introduction of polar chemical groups or coupling agents onto the surface which are then available for bonding
- increase in surface roughness giving rise to improved mechanical interlocking or increased bondable surface area.

The intention of the surface treatment is to modify the chemistry or morphology of a thin surface layer without affecting the bulk properties. Some of the special characteristics of composite materials need to be considered when treating their surfaces for adhesive bonding, because there is a danger that some treatments may cause delamination defects just below the surface or damage to the relatively brittle fibers, which may result in poorer mechanical properties of the composite. There is also problem with contamination of composite during surface treatment, which can also make mechanical properties worse [10]. There are two commonly used methods of surface treatment: Mechanical abrasion and Peel ply.

Mechanical abrasion use sanding or grit blasting to prepare small areas for secondary bonding or for field repairs. The choice is generally determined by available production facilities and cost. Sanding is usually the most appropriate surface treatment for field repairs, because it is able to remove any organic finishes and also prepare the surface. Nevertheless, this method must be used with caution to prevent damage of the laminate.

Peel ply can be applied during the original layup of the laminate. This method involves using an additional ply which is put on laminate and removed just prior bonding, which gives the bond a mechanical grip and provides surface consistency. Pre-impregnated peel ply has proven to be the most satisfactory surface treatment for bonding of large parts during manufacturing. Peel plies are made of kevlar, teflon, nylon or aramid [4; 11].

3.5.2 Surface treatment for bonding composite to metals

Composites are usually bonded to titanium or steel. Titanium is usually preferred because it exhibits better adhesion [4].

Preparing the surface of a metallic sample involves multiple steps, all of which are not always taken. It is not possible to obtain a strong adhesive bond without cleaning (and abrading) the metal surface. Metals have high energy surfaces and absorb oils and other contamination that may be present in vapor form or fine mist in the ambient. The following steps should be taken to prepare metal surfaces for bonding, although steps 4 and 5 are less frequently utilized:

1. cleaning (using a solvent or another chemical),
2. removal of loose materials (mechanical, e.g., grit blasting) also increases contact surface,
3. improvement of corrosion resistance,
4. priming (applying a primer material to the surface),
5. surface hardening (mechanical or chemical to strengthen the surface).

The best method to clean a metal surface is by vapor degreasing with an organic solvent such as trichloroethane. The next step is sandblasting to increase the adhesive contact surface area by roughening the metal surface. Chemical etching strips away weakly bonded oxides and forms an oxide that is strongly bonded to the bulk material. An alternative is priming of the part surface to improve the wettability of the surface and protect it from oxidation. Also the special primers are used in order to improve the adhesion of disparate materials. Moreover, special high-elastic glues can be used as primers in order to eliminate the stress concentrations.

A given process is considered desirable only if the entire production process can accommodate it. Cost must be considered and balanced against the requirement for reliability, maintainability, and critical roles of the joint [12].

Titanium

Mechanical, chemical, electrochemical and energetic surface treatments are used to enhance the surface of titanium alloys prior to bonding. Durability studies of titanium reveal that surface preparations that produce no roughness (macro or micro) evince the poorest bond durability. Those that produce significant macro-roughness but little micro-roughness evinces moderate to good durability. Finally, those that produce significant micro-roughness evince the best durability. Surface treatment of titanium is required when joint is exposed to hot-moist environmental conditions or harsh environmental conditions. In these conditions, joint without adequate pretreatment exhibit a little durability.

Titanium surface treatments include traditional methods such as chromic acid anodizing[10] and sodium hydroxide anodizing as well as laser treatment. Typical composite surface treatments include traditional abrasion/solvent cleaning techniques (refer to chapter 3.5.1) for thermoset composites, while thermoplastic composites require surface chemistry and surface topographical changes to ensure strong and durable bond strengths [11].

[10] Anodizing is an electrolytic passivation process used to increase the thickness of the natural oxide layer on the surface of metal parts. The layer developed by anodizing do not oxidize, thus protects material.

4 The issues associated with joining of CFRP to metals

When joining CFRP to metals, it brings new problems and measures, which were not discussed so far. It is evident that for the mechanically fastened joints different measures are applied to avoid failures caused by interaction between CFRP and metals than for the adhesive bonded joints, but unfortunately one issue is common for both. It is galvanic corrosion.

4.1 Galvanic corrosion

Since the fasteners are usually made of metals as any joints with the fasteners are the metal-CFRP joints, regardless materials of the parts. This means the galvanic corrosion can occur.

In general, corrosion is defined as a degradation of metal by an electrochemical reaction with its environment [13]. Galvanic corrosion can occur when two metals are connected together. Then the less noble metal tends to corrode at an accelerated rate, compared with the uncoupled condition while the more noble metal will corrode more slowly. Fig. 4-2 shows galvanic series[11] in seawater. This arrangement of metals in a galvanic series based on observations in seawater is frequently used as a first approximation of the probable direction of the galvanic effects in other environments [14]. As is shown, graphite[12], which is used to reinforce the polymer matrix, is very noble material, thus it is vital to make fasteners of comparably noble material. From the point of view of the galvanic corrosion, the best choice would be gold or platinum, but regarding to their cost and weight it is necessary to find better possibility. In most of cases, the best option is titanium, which exhibits exceptional strength-to-weight ratio, elevated temperature performance and galvanic corrosion resistance. These properties make the titanium one of the most suitable material for aerospace application [15].

Another possibility may be stainless steels, which possess unusual resistance to corrosion in atmospheric environment and are produced to cover a wide range of mechanical and physical properties for particular applications [15]. Unfortunately, as it is shown in Fig. 4-2, their nobility compared with the graphite is very poor, thus it is not much appropriate for joining to CFRP without any insulating layer, same as the aluminum.

In case of metal-composite adhesive joints, the solution is rather simple. Electrical isolation of carbon fibers and metal can be performed by using organic or glass fibers plies. However, in case of mechanically fastened joints, it is not enough due to the fact that the fasteners still have contact to CFRP in the thread. Thus it is essential to use sealer coatings to cover all the surfaces of the fasteners which have contact with CFRP. [16].

[11] Galvanic series arranges the materials given their corrosion potential. Differences in corrosion potentials of dissimilar metals can be measured in specific environments by measuring of the current that is generated by the galvanic action of these materials when exposed in a given environment.
[12] Carbon fibers can be also called graphite fibers. In engineering practice, these two expressions are taken as the same thing.

In case of adhesive bonded joints, the adhesive can be used as the insulating layer instead of organic fibers plies and the problem seems to be solved, but there is a remarkable case of corrosion failure found in the Airbus A320 CFM56-5b intakes. This failure occurs in a dissimilar material joint and is strongly related to the adhesive loss of integrity at very low temperatures. The ductility reduction of adhesive at low temperatures promotes the microcracks formation within adhesive layer of the bonded joint, which in turn sparks the corrosion process. These microcracks allow the infiltration of dust, moisture, contaminants and salt water in the interface between adhesive and adherends, which promotes the creation of an electrolyte between the joint adherends and consequently their galvanic corrosion. These microcracks are usually found in the bonded joints so it is necessary to take this possible problem into consideration and use adhesives with better mechanical properties when it is essential. Another solution is to use some organic fibers plies or sealer coating with similar or better adhesion as the CFRP counterpart exhibits [17].

Noble or Cathodic	Platinum
	Gold
	Graphite
	Titanium
	Silver
	Chlorimet 3
	Hastelloy C
	18-8 Mo Stainless Steel (Passive)
	18-8 Stainless Steel (Passive)
	Chromium Steel >11% Cr (Passive)
	Inconel (Passive)
	Nickel (Passive)
	Silver Solder
	Monel
	Bronzes
	Copper
	Brasses
	Chlorimet 2
	Hastelloy B
	Inconel (Active)
	Nickel (Active)
	Tin
	Lead
	Lead-tin Solders
	18-8 Mo Stainless Steel (Active)
	18-8 Stainless Steel (Active)
	Ni-resist
	Chromium Steel >11% Cr (Active)
	Cast Iron
	Steel or Iron
	2024 Aluminum
	Cadmium
Active or Anodic	Commercially Pure Aluminium
	Zinc
	Magnesium and Its Alloys

Fig. 4-1 Galvanic series of metals and alloys in seawater

4.2 Recommended design practices for CFRP-metal spliced joints

For adhesive bonding of CFRP to metal step joints are preferred, due to their better design flexibility, more consistent results and lower cost for thicker laminate joints compared with the scarf joints. One of these is shown in Fig. 4-3.

Fig. 4-2 Example of CFRP-titanium spliced joint

The first property of material, which have to be take into consideration is the thermal expansion, due to the fact that diverse thermal expansion could cause increase of stress in joint. From this point of view, the best material is titanium. Steel or aluminum can also be used, but just under special circumstances.

Then a suitable LSS have to be set. Basic recommendations say that where it is possible, 0° plies in primary load direction should be placed adjacent to the bond-line; ±45° plies are also acceptable, but 90° plies should never be placed adjacent to the bond-line unless it is also primary load direction. Then, on the first and the last step of a bonded joint, ±45° plies should be used to reduce peak interlaminar shear stresses at the end of steps. In addition, too many of 0° plies should not be used at the end of any step surface to avoid peak stress.

The last consideration should be directed toward the metal thickness at the end step. The metal should be thick enough at this location in order to prevent its failure. [4].

5 Stress-strain analysis of the joints

For stress-strain analysis of the joint being designed the analytical method developed by Y. S. Karpov was used [18]. The method is based on discretization of joint to enough small elements along the joint axis and perpendicular to it. Afterwards, stresses are calculated in each element. The calculation was performed by means of Excel and MATLAB. User have to enter input data to Excel file. Then Excel creates system of matrixes, which are solved by script written in MATLAB, hence it is really important to understand how MATLAB and Excel cooperates.

After entering the input data to the Excel file, user have to close Excel to enable to MATLAB take data from it. Then, everything is prepared for MATLAB, which takes data from Excel and solves matrixes. The required solutions are automatically written back to the Excel file, which calculates the rest, stresses in the parts and adhesive or fasteners. From these solutions it also calculates failure indexes, which are listed in clearly arranged worksheet. In this thesis, just stress-strain analysis of adhesive bonded joints will be explained, but the same method with some differences can be used for mechanically fastened joints as well.

5.1 Loading modes

In the following section, the analytical method for assessment of the stress-strain state of the adhesive bonded joints will be briefly explained. First of all, the loading mode has to be determined. There are two models of surface adhesive joint, which are shown in Fig. 5-1. Model 1 should be used when the shear modulus of the adhesive G_{adh} is smaller than the interlaminar shear modulus G_{13}, where 1 is the direction along the fibers and 3 is the direction perpendicular to the plane of the joint. Otherwise, model 2 should be used.

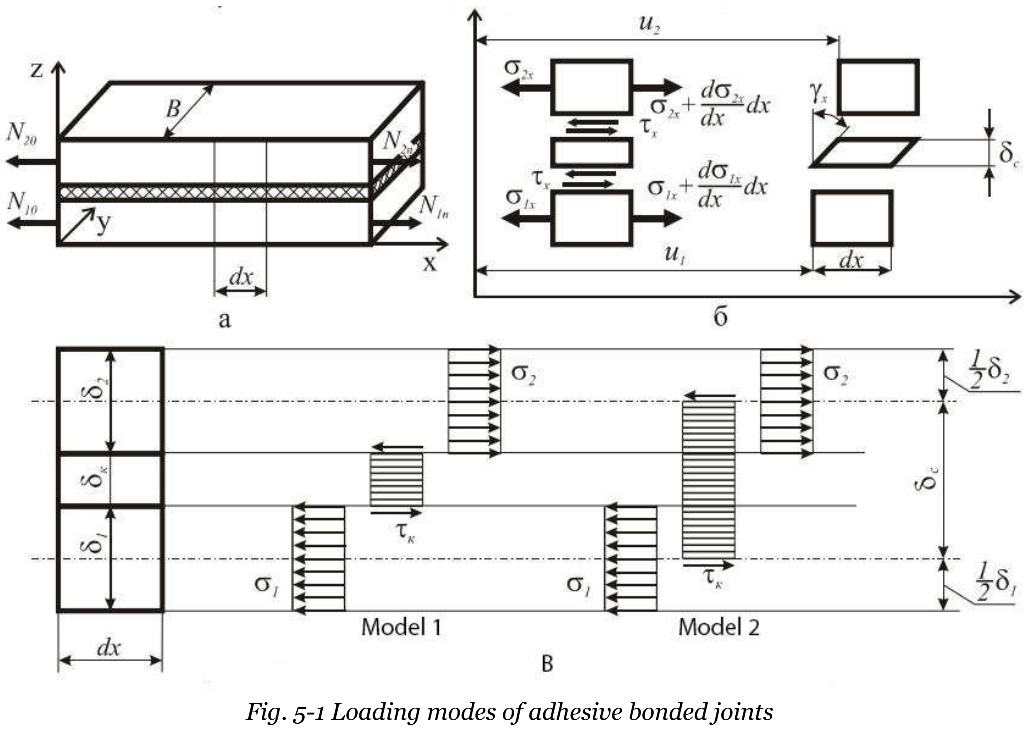

Fig. 5-1 Loading modes of adhesive bonded joints

After selection of the appropriate model, the joint has to be split into elements (discretization), which also includes definition of spacing between elements t_x. Spacing between elements is of course determined from the length of the joint and the number of elements. For determination of the length it is appropriate to meet another condition, which is related with the strength of the joint and defines the minimum length of a lap joint, even though, this condition is proposed for the joint between parts from the same material and with the same thickness in each element. For our purpose it can serves as a preliminary assessment. The most important parameter for this condition is the ratio between length l, which is selected at the beginning of calculation, and maximum thickness δ_0 of the joined parts.

If $\frac{\delta_0}{l} < 0{,}1$ then,

$$\frac{2N}{Bl} \leq \sigma_{KB} \; or \; \frac{N}{Bl} \leq \tau_{KB}$$

If $\frac{\delta_0}{l} \geq 0{,}1$ then,

$$\frac{N}{Bl}\sqrt{\frac{4+\frac{\delta_0^2}{l^2}}{1+\frac{\delta_0^2}{l^2}}} \leq \sigma_{KB}$$

Where $\sigma_{KB} = 2 * \tau_{KB}$ and τ_{KB} is the maximum allowable shear stress of adhesive, B is the width of the joint. If the condition is not met, the ratio between the length and the thickness have to be changed.

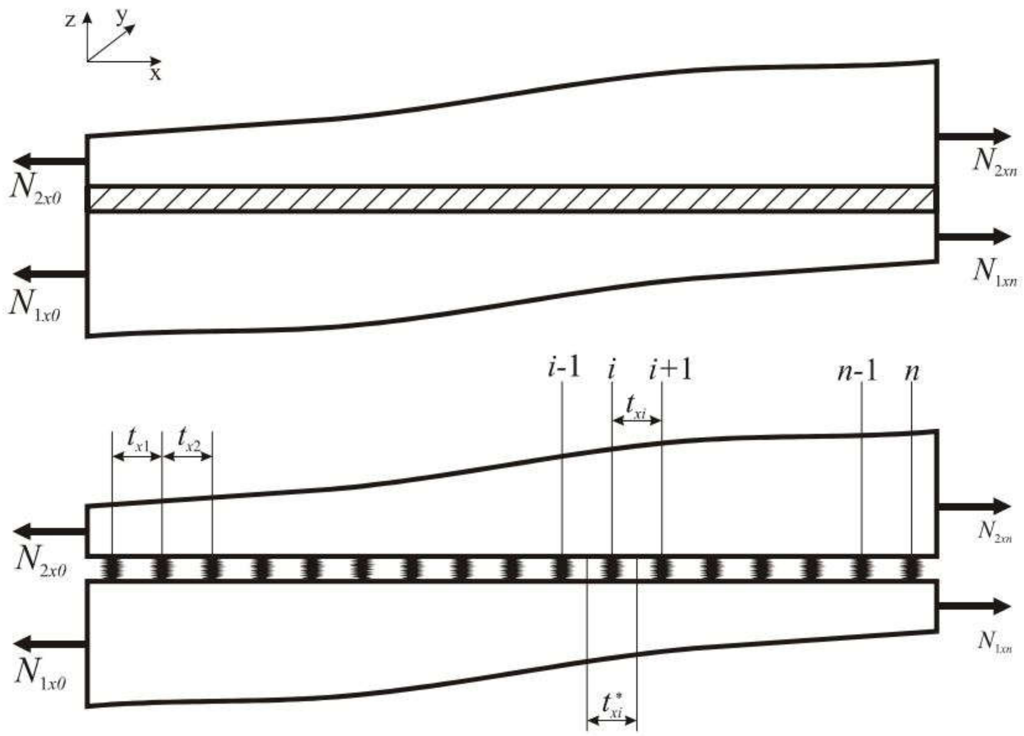

Fig. 5-2 Discretization of the joint

After the discretization, the mechanical compliances of both parts and of the adhesive are determined. Calculation is performed by applying formulas described below. This calculation is valid only for case of balanced joint, which is loaded as is shown in Fig. 5-3.

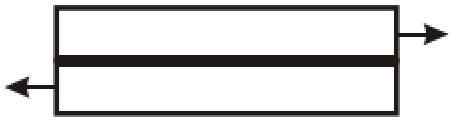

Fig. 5-3 Type of load

5.2 Calculation of compliances in direction of loading

For the bonded parts hold that:

$$if\ i \leq n-1: \pi_{1,x_i} = \frac{2}{B \cdot E_1 \cdot \left(\delta_{1,i} + \delta_{1,i+1}\right)}$$

$$if\ i > n-1: \pi_{1,x_i} = 0$$

Where 1 labels that compliance is calculated for the first part[13], x is the longitudinal axis designation, E_1 is the Young's modulus in loading direction, n is the total amount of elements and i is the element number where the compliance is calculated, as is shown in Fig. 5-2.

The compliance calculation of the adhesive in the loading direction depends on the adhesive joint model.

For the model 1 holds that:

$$if\ i \leq n-1: \pi_{a,x_i} = \frac{\delta_{adh}}{B \cdot t_x \cdot G_{adh}}$$

$$if\ i > n-1: \pi_{a,x_i} = 0$$

For the model 2 holds that:

$$if\ i \leq n-1: \pi_{a,x_i} = \frac{1}{B \cdot t_x} \cdot \left(\frac{\delta_{1_i} + \delta_{1_{i+1}}}{4 \cdot G_{13}} + \frac{\delta_{2_i} + \delta_{2_{i+1}}}{4 \cdot G_{13}} + \frac{\delta_{adh}}{G_{adh}}\right)$$

$$if\ i > n-1: \pi_{a,x_i} = 0$$

Where δ_{1_i} is the thickness of the first part and δ_{2_i} is the thickness of the counterpart in the i-th element and δ_{adh} is the thickness of the adhesive layer.

By means of these compliances, shear forces in the adhesive and normal forces in the parts are determined.

---

[13] Due to the fact that two parts are bonded together, compliance must be determined for both parts. Formulas are the same. Only difference is that each index 1 is replaced by index 2, which is designation of the second part (counterpart).

5.3 Calculation of compliances in the transvers direction

Deformation in the longitudinal direction causes initiation of stress in the transverse direction, hence calculation of the stress in the transverse direction is sometimes also required even though joint is not loaded in that direction. For that purpose, it is necessary to determinate compliances in transverse direction as well.

For bonded parts hold that:

$$if\ i \leq n-1: \pi_{1,y_i} = \frac{1}{B \cdot E_2 \cdot \delta_{1,i}}$$

$$if\ i > n-1: \pi_{1,y_i} = 0$$

Where y is the transverse axis designation and E_2 is the Young's modulus in the transverse direction.

The compliance calculation in the transverse direction is also dependent on the chosen adhesive joint model. In our case, thickness in transverse direction is constant, thus it is possible to simplify formulas into following forms.

For the model 1 holds:

$$if\ i \leq n-1: \pi_{a,y_i} = \frac{\delta_{adh}}{B \cdot t_x \cdot G_{adh}}$$

$$if\ i > n-1: \pi_{a,y_i} = 0$$

For the model 2 holds that:

$$if\ i \leq n-1: \pi_{a,y_i} = \frac{1}{B \cdot t_x} \cdot \left(\frac{\delta_{1_i}}{2 \cdot G_{13}} + \frac{\delta_{2_i}}{2 \cdot G_{13}} + \frac{\delta_{adh}}{G_{adh}}\right)$$

$$if\ i > n-1: \pi_{a,y_i} = 0$$

Calculation of transverse stress will not be more explained in this thesis, because in our case, deformation in that direction is not significant enough to induce hazardous stress. This was verified in calculation which is enclosed in the appendix A.

5.4 Calculation of longitudinal forces and stresses applied in the joint

Shear forces in the adhesive are determined from following equation:

$$\begin{aligned}(\pi_{1,x_i} + \pi_{2,x_i}) \sum_{k=1}^{i-1} Q_{xk} + Q_{xi}\left(\pi_{1,x_i} + \pi_{2,x_i} + \frac{1}{t_{x,i}}\pi_{a,x_i}\right) - Q_{x,i}\frac{1}{t_{x,i}}\pi_{a,x_{i+1}} \\ = \pi_{1,x_i}N_{1x0} - \pi_{2,x_i}N_{2x0} + \Delta T(\alpha_{1,xi} - \alpha_{2xi});\ i = 1, \dots, n-1\end{aligned}$$

and this equation is complemented by the summing equation:

$$\sum_{k=1}^{n} Q_{xk} = N_{1x0} - N_{1xn} = N_{2xn} - N_{2x0}$$

Where Q_{xi} is the shear forces in the i-th element. N_{1x0}, N_{2x0}, N_{1xn} and N_{2xn} are the internal longitudinal forces occurred in the bonded parts, as is shown in Fig. 5-2. ΔT is the temperature change and $\alpha_{1,xi}$, α_{2xi} are the coefficients of linear temperature expansion.

Calculation of shear stresses in the adhesive:

$$\tau_{xy_j} = \frac{Q_{x_j}}{t_x \cdot B}; j = 1, \dots, n$$

For the longitudinal forces in each element hold that:

$$N_{1xi} - N_{1x0} = -\sum_{k=1}^{i} Q_{xk}$$

$$N_{2xi} - N_{2x0} = \sum_{k=1}^{i} Q_{xk}$$

$$i = 1, \dots, n-1$$

Calculation of the normal stresses in the bonded parts:

$$\sigma_{1,x_i} = \frac{N_{1,x_i}}{\delta_{1_i} \cdot B}; i = 1, \dots, n-1$$

$$\sigma_{2,x_i} = \frac{N_{2,x_i}}{\delta_2 \cdot B}; i = 1, \dots, n-1$$

Forces and stresses are determined for each element and then failure indexes are calculated.

5.5 Calculation of shear compliances and forces for a shear loaded joint

The joints are often subjected to the shear load hence the calculation of shear stresses is also needed. The method of determination of shear stresses will be shortly explained below.

For the shear compliances hold that:

$$\pi_{1,xy_i} = \frac{1}{\delta_{1i} \cdot G_{1xy}}$$

$$\pi_{2,xy_i} = \frac{1}{\delta_{2i} \cdot G_{2xy}}$$

Where G_{1xy} and G_{2xy} are the shear moduli of bonded parts.

For shear forces hold that:

$$Q_{yi} = t_{y,j}\left(q_{1,j} - q_{1,j-1}\right), Q_{yi} = t_{yj}\left(q_{2,j-1} - q_{2,j}\right)$$

$$j = 1, \dots, n$$

$$q_{1,i-1}t_{y,i} + q_{1,i}\left[t_{x,i}\left(\pi_{1,xy_i} + \pi_{2,xy_i}\right) - t_{y,i}\left(\pi_{a,y_i} + \pi_{a,y_{i+1}}\right)\right] + q_{1,i+1}t_{y,i}\,\pi_{a,y_{i+1}} = t_{x,i}\,\pi_{2,xy_i}\left(q_{1,0} - q_{2,0}\right)$$

$$i = 1, \dots, n-1$$

Where $t_{y,i}$ is spacing in transverse direction, q_{1i} and q_{2i} are shear flows in bonded parts, Q_{yi} is shear force in the i-th element determined from shear flows.

Following equation can be used for check of correctness of solution:

$$q_{1,0} + q_{2,0} = q_{1,i} + q_{2,i} = q_{1,n} + q_{2,n}$$

From shear forces Q_{yi} shear stresses and also shear failure indexes can be calculated.

In conclusion, it is important to state that method described in this whole chapter is simplified for the case shown in Fig. 5-3. It is not general method which could be used for any kind of joints.

6 Design of the given joint for the AeroMobil 3.0

The main goal of this thesis was to design the structural design of the joint between metal hinge and CFRP wing spar. During designing of the joint much emphasis was being directed toward the resulting weight. At the very beginning, two types of joints were considered: a mechanically fastened joint and an adhesive bonded joint.

Stress-strain analysis was performed for both types of the joints. On the basis of results, adhesive bonded joint was found as a better option for our case. This was mainly caused by very narrow and thin flange. Manner of placing of bolts with enough big diameter for maintaining safety and simultaneously preserving lower weight, than adhesive bonded joint exhibits for that case, was not found for these dimensions of flange. Moreover, the process of composites mechanical fastening is very time consuming and more complicated than the adhesive joining. Special technics and tools are required for this process.

Resultant joint was performed as a stepped-lap joint with nine steps and is shown in following figure.

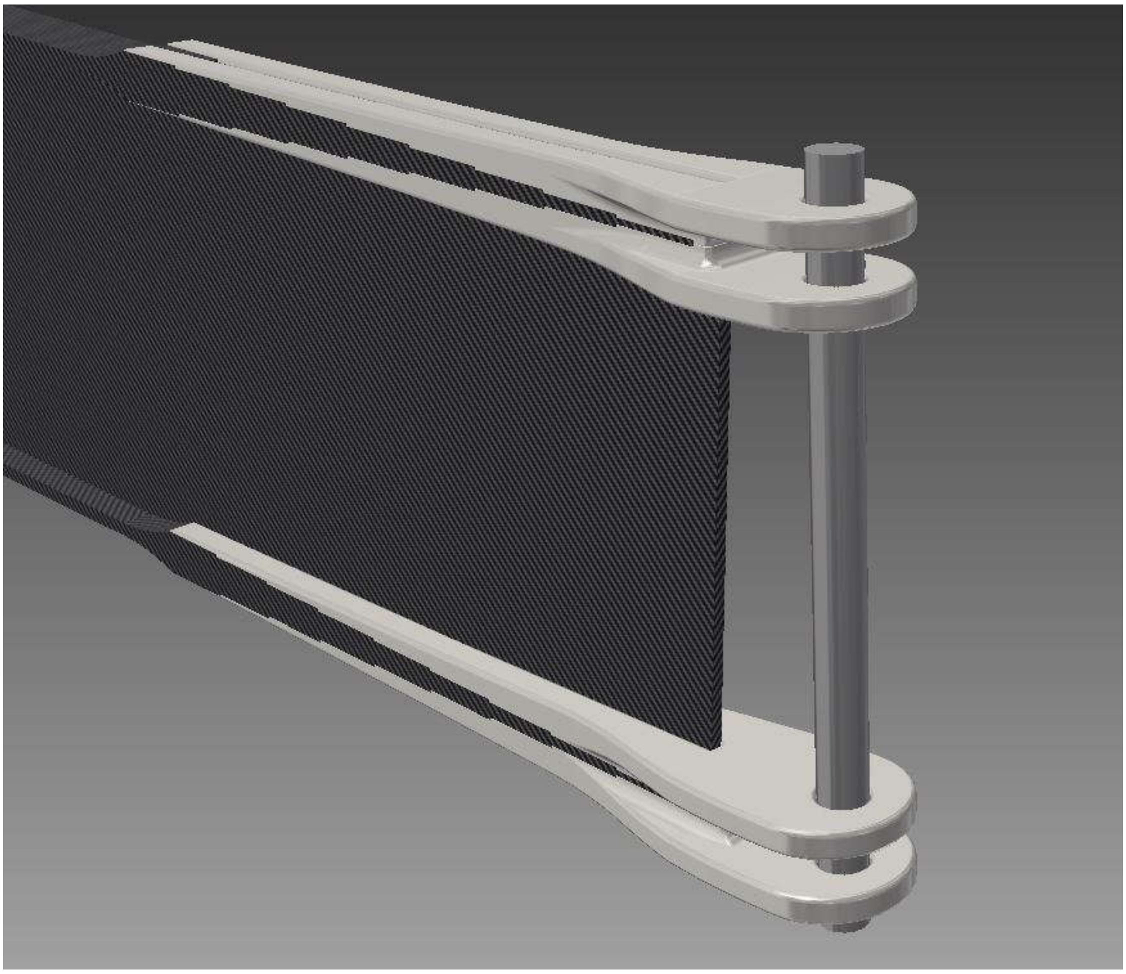

Fig. 6-1 Structural design of the joint between metal hinge and CFRP wing spar for AeroMobil 3,0

The material of hinge was chosen titanium, because of its great mechanical properties and relatively low density. Another advantage of titanium is good galvanic corrosion resistance (refer to chapter 4.1).

Titanium hinges are attached to wing spar belts by two stepped plates as is shown in Fig. 6-1. Both joints are divided into two parts, upper and lower section. The reason of that is simplification of assembling process. Titanium is hard to-machine hence during designing great emphasis must be placed on manufacturability of components. Individual parts are shown in following figures.

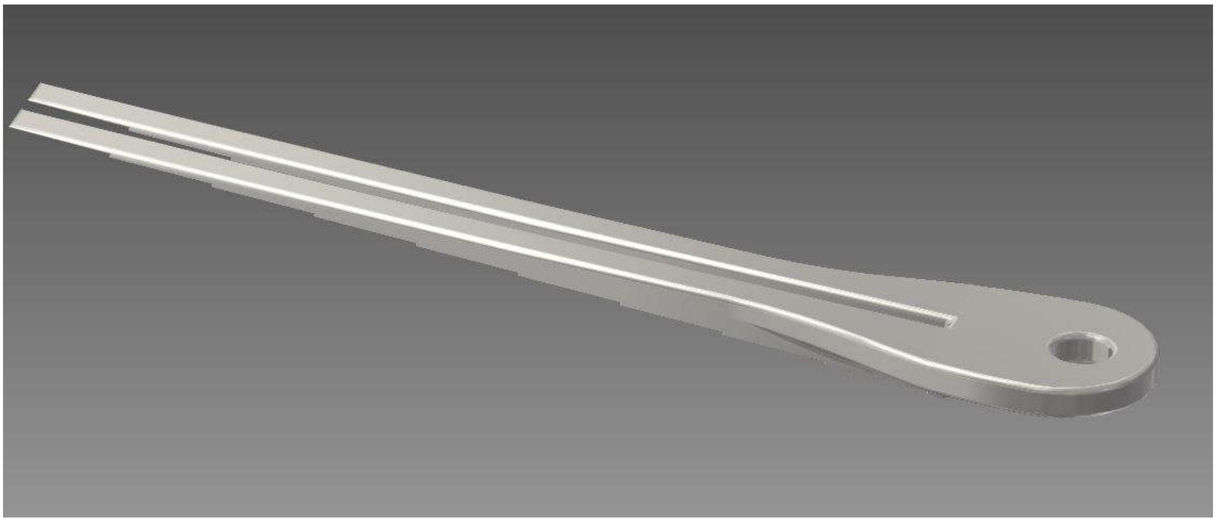

Fig. 6-2 Upper hinge, upper part

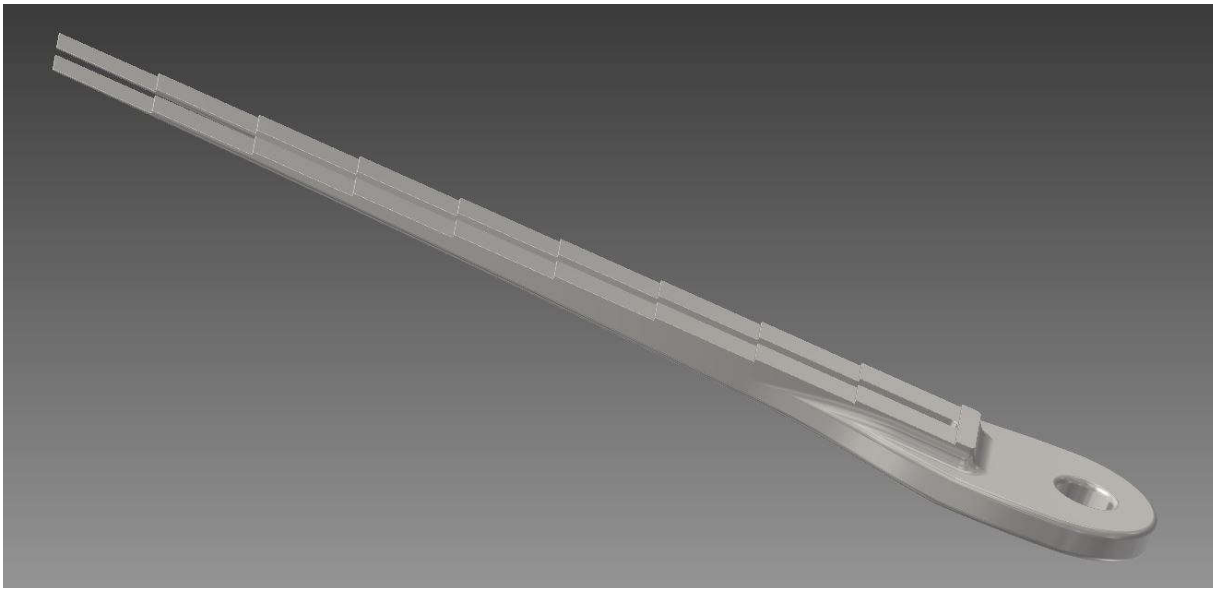

Fig. 6-3 Upper hinge, lower part

The upper hinge and the lower hinge do not have the same dimensions. The Lower hinge is longer and slightly heavier than the upper one. This is caused by misalignment of the axle passing through the loop.

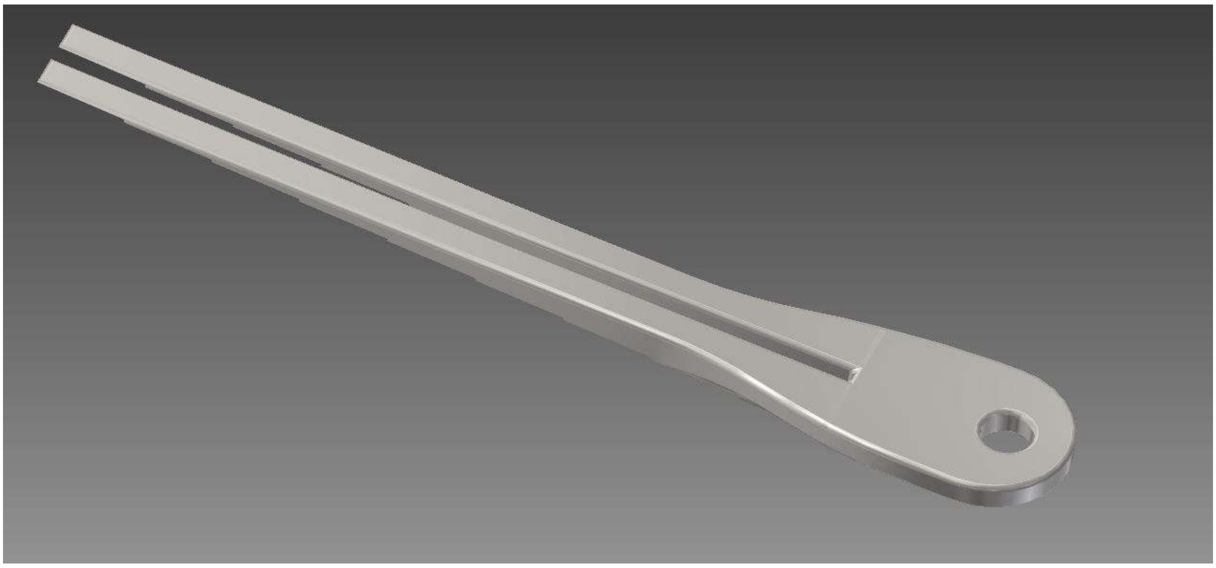

Fig. 6-4 Lower hinge, upper part

Fig. 6-5 Lower hinge, lower part

The parts are bonded to the wing spar from both sides of the spar belts. The thickest places are expected where the upper and the lower parts are in contact. Demonstration of the joint assembling is shown in animation, which is enclosed in the Appendix A.

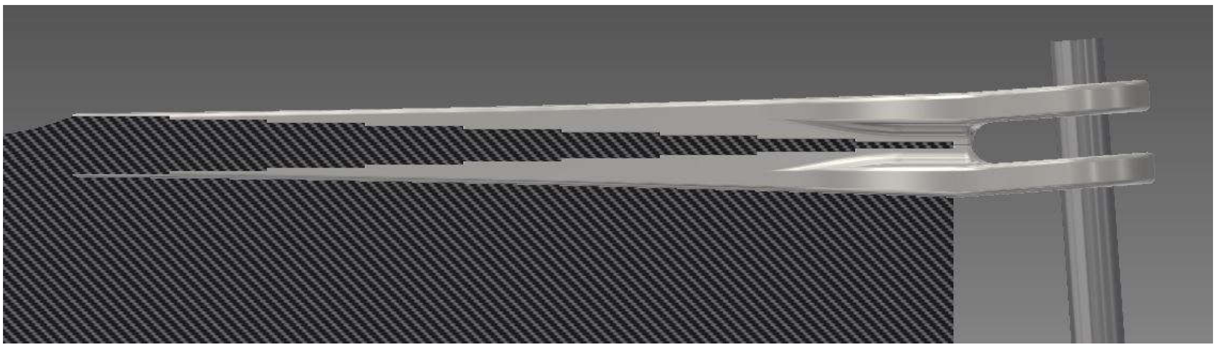

Fig. 6-6 Assembled upper hinge

As can be seen in Fig. 6-1 and 6-6, the CFRP tip of the wing spar flange had to be also adjusted, because of very small thickness. In the joint, thickness of the flange was increased what allowed manufacturing of required steps. This is better shown in following figure.

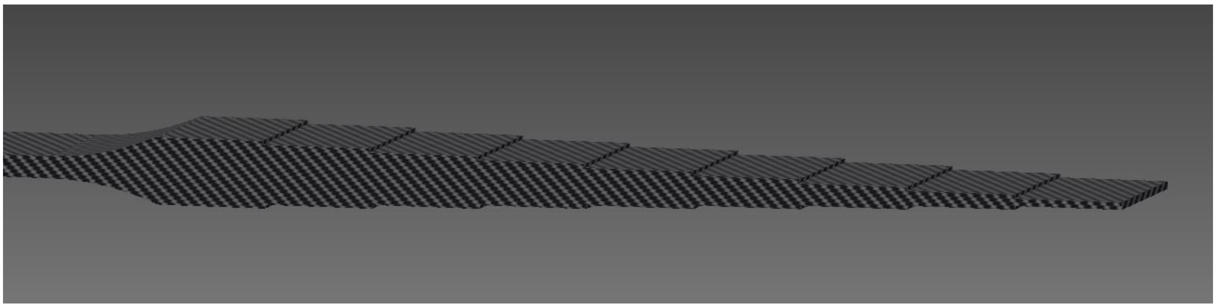

Fig. 6-7 Thickness change of the flange

The main advantage of this structural design is substantial weight saving. The designed metal hinge is approx. 12% lighter than the original one (see Tab. 6-1). Significant weight saving will be also reached due to reduction of thickness of the spar wall. The flange thickness increasing is negligible in comparison to the spar wall thickness increasing of the original design. Stress-strain analysis of this structural design is enclosed in appendix and includes also failure indexes in bonded parts and adhesive. Load of the joint was taken from the report [19], which was provided by AeroMobil s.r.o. The weight saving was estimated by the comparison of the computer models of the original joint (which is enclosed in Appendix A) and of the designed one as is shown in following table.

Proposed design	Approximate weight	Length of the joint	Width of the joint
Upper hinge	$0{,}800\ kg$	$279\ mm$	$4 \cdot 12{,}5\ mm$
Lower hinge	$0{,}889\ kg$	$279\ mm$	$4 \cdot 12{,}5\ mm$
Original design	Approximate weight	Length of the joint	Width of the joint
Upper hinge	$0{,}929\ kg$	$306\ mm$	$2 \cdot 32\ mm$
Lower hinge	$0{,}987\ kg$	$311\ mm$	$2 \cdot 32\ mm$
Comparison	Percentage weight saving	Length difference	
	12%	$27\ mm$ $32\ mm$	

Tab. 6-1 Comparison between the original design of the hinge and the proposed design

The thickness of the flange was increased from 6 mm to 18 mm and then reduced by 1 mm in each step.

Following table shows the most important failure indexes.

Failure indexes	
Normal stress in bonded parts	$k_{CFRP} \cong 4{,}32$
	$k_{titanium} \cong 1{,}52$
Shear stress in adhesive	$k_{adhesive} \cong 1{,}46$

Tab. 6-2 Summary of the failure indexes

Letoxit KFL 120 made by 5M – Smart Technologies was used as the adhesive for the calculation.

Design of the sample for static test is shown in Appendix B.

Conclusion

The main goal of this thesis was to propose suitable design of the joint between metal hinge and CFRP wing spar of the flying car AeroMobil 3,0. Secondary goal was to sum up basic principles of joining advanced composites to metals. This topic was processed in the first part of the thesis. The second part of the thesis was dedicated to designing of the new structural design of the joint between the metal hinge and the CFRP wing spar of AeroMobil 3.0. On the basis of calculation, it was decided that the best joining method for this joint is adhesive bonding and as the result, the significant weight saving was reached.

The proposed design can be taken as preliminary because the temperature loading was not taken into account and the optimization of the joint was not done. In order to validate the provided design, FE analysis of the designed joint can be performed and finally real static tests of the joint demonstrator can be done.

However, the provided design could serve as a first approximation of the future design.

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Symbols and abbreviations

CFRP	Carbon-Fiber-Reinforced-Polymer
FRP	Fiber-Reinforced-Polymer
LSS	Laminate Stacking Sequence
d	Diameter
e	Edge distance
t	Laminate thickness
w	Width
P	Load
N	Internal longitudinal force occurred in the jointed parts
B	Width of the joint
l	Length of the joint
G_{adh}	Shear modulus of adhesive
G_{13}	Shear modulus in the plane 13, where 1 is direction along the fibers and 3 is direction perpendicular to plane of joint
σ_{KB}	Adhesive normal stress allowable
τ_{KB}	Shear stress allowable of the adhesive
δ_1, δ_2	Thickness of bonded parts
$\pi_{1,x}, \pi_{2,x}$	Compliances of bonded parts in longitudinal direction
E_1	Young's modulus in longitudinal direction of the parts material
$\pi_{a,x}$	Compliance of the adhesive in longitudinal direction
δ_{adh}	Thickness of the adhesive
t_x	Spacing in longitudinal direction between elements centroids
$\pi_{1,y}, \pi_{2,y}$	Compliances of parts in transverse direction
E_2	Young's modulus of the parts material in transverse direction
$\pi_{a,y}$	Compliance of the adhesive in transverse direction

Q_x	Shear force applied to adhesive
ΔT	Temperature change
$\alpha_{1,x}, \alpha_{2,x}$	Linear thermal expansion coefficients of bonded parts
$N_{1,x}, N_{2,x}$	Internal longitudinal forces occurred in the bonded parts
τ_{xy}	Shear stress in the adhesive
$\sigma_{1,x}, \sigma_{2,x}$	Normal stress in bonded parts
$\pi_{1,xy}, \pi_{2,xy}$	Shear compliances of bonded parts
Q_y	Shear force in transverse direction occurred in adhesive
q_1, q_2	Shear flow
t_y	Spacing in transversal direction

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Appendix A

Stress strain analysis and 3D model of the designed joint

As an appendix A is enclosed CD, which contains following files:

Stress-strain analysis of designed joint

- Excel file - calculation_adhesive.xlsx
- MATLAB file - calculation_matlab_adhesive.m

3D model of designed joint

- Animation of assemblage - assemblage.avi
- Inventor files
    - 3D model of the joint - joint.stp
    - 3D model of the lower hinge, lower part - lower_metal_hinge_lower.stp
    - 3D model of the lower hinge, upper part - lower_metal_hinge_upper.stp
    - 3D model of the upper hinge, lower part - upper_metal_hinge_lower.stp
    - 3D model of the upper hinge, upper part - upper_metal_hinge_upper.stp
    - 3D model of the sample for the static test - sample_static_test.stp

Appendix B

Sample for the static test

In the following figures is shown the sample for the static test. All bolts have 5 mm diameter and for better strength, the use of the adhesive between jigs and wing spar is also recommended. For static test, the flange has to be broadened, due to the lack of place for the bolts.