Bolted Joining and Repair of Composite Stiffeners

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***Abstract*—This document reviews the failure modes of bolted joints and also tries to give more importance to the view of re-designing the joint by varying many parameters individually and test how these influence the failures. It also checks the optimization of each component in the joint. In this, the structures damaged (at different levels) joined by bolts are analyzed (by both FEA and Theoretically) and compared with the load carrying capacity of undamaged structures and effects to prove that, the particular joint has substituted for the strength losses due to damage. Using all the above data, the components are re-designed in such a way that they are to be perfectly optimized to take loads that generally act on daily basis. This will reduce the weight of components without any compromise on required load taking capacity. For this purpose, there are design curves available from which a designer can easily choose various details wanted for the required joint (like type of bolt, diameter, thickness of structure) according to the data on hand about the loads expected to be taken by each component. So this also serves as a database for comparing & selecting optimized joints for each case to keep the reserve factor of design greater than and near to unity and a research tool restricted only to analyse with axial in-plane loads**

***Index Terms*— Joining technology in composites ; bolted joints ; optimisation of joints; aircraft wing structure analysis**

I. INTRODUCTION

The use of advanced composites in primary and secondary aircraft structures is steadily growing. The increased application of composites in aircraft industry is motivated as these materials offer a number of advantages compared to conventional engineering materials such as steel and aluminum. Although bolts and fasteners may be the smallest elements in design, they do not minimize their importance. Bolt failures have resulted in fatal accidents, crashes, catastrophic ruptures, foreign object damage in gas turbines, leaks and subsequent explosions of hydrocarbons The strength of a bolted connection is governed by the least value of the following:

· Shear strength of the bolts.
· Bearing strength of the bolted members.
· Tensile strength of the bolted members at the weakest section.
· Cleavage strength. [7]

In addition to this, secondary bending moments may also act [5]. The failure load and the transition between the failure modes are affected by several parameters, such as laminate stacking sequence, the geometry of the joint and properties of the fastener. Loads are transferred between the joint elements in the bolted connections by compression on internal faces of the fastener holes with smaller component of shear on outer faces. Stiffeners are used in aircraft structures along with other components with particular load taking capacity. [1, 4] Some typical failure modes of bolted joints are shear out, bearing, cleavage and tension. The type of failure depends on the ratio of the effective width to the diameter of the fastener hole *w/d*, and the ratio of the edge distance to the diameter *e/d.* (Fig 1)

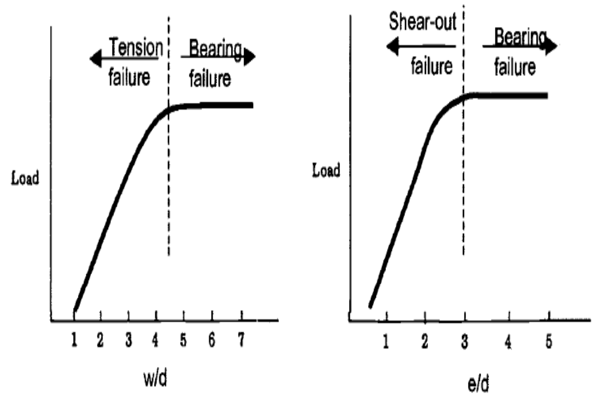

Fig 1: Criteria for Failure Modes

Here, aircraft wing structure model with different levels of damage has been considered as follows:

A101- intact model,
B101-half stringer damage (repaired with 6 bolts)

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B201-half stringer damage (repaired with 8 bolts)
C201-full stringer damage (repaired with 8 bolts)

II. FINITE ELEMENT ANALYSIS (FEA)

Structural laminates can be modeled using the two-dimensional planar plate elements. Beams of various sections can also be assembled from plate elements if plate elements are used to represent the webs and flanges of the beam. [10,9].In this approximation, the stresses acting through the thickness are not modeled.

Real-time cut-section of a wing as a stiffened panel, consisting of stringers & covering wing skin is analyzed for taking loads. As the laminate is thin, it is assumed that the laminate deforms uniformly over the thickness under in-plane loading where, through thickness shear cracks and delamination are out of plane failure modes.

The composite layer stacking sequence differs between the skin, web and flange as they are chosen using the HACCOT industrial standard. These are input in the composite properties card PCOMP separately for each which is used to derive the stiffness constants of the composites. The stringer and skin are connected by RBE2 (rigid body) elements to ensure uniform distribution of loads applied for simulating adhesive joining effects as per the assumptions [6,5] i.e. to satisfy the co-bonded condition

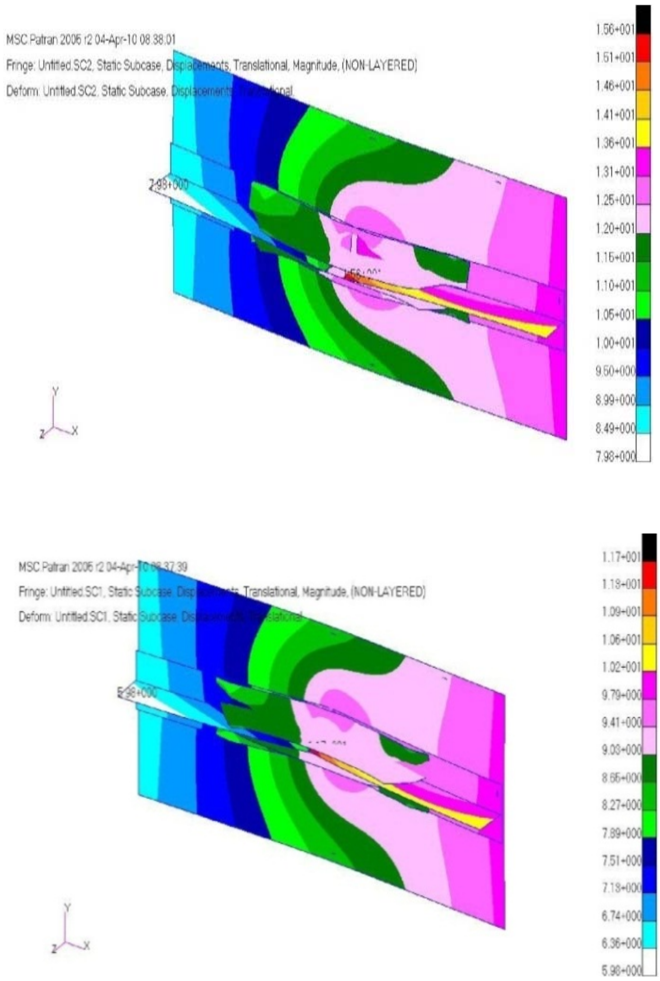

Fig 2: FEA graphics of deformation data output.

The independent nodes are that of skin and the dependent nodes are that of stringer flange. Here MULTIPOINT constraint modeling is used. The bolts are shown by the 1D bush elements and its properties are input using PBUSH card.

The bolt properties are calculated by HUTH criteria and are different for bolts in the web and bolts in the flange section. First, a set of orthotropic properties is defined for the unidirectional or fabric plies to be used in the laminate. A tabular input is then given to define the orientation, thickness, and material for each ply. The properties required for the plate are then calculated and cross checked with that of numerical result.

The wing structure is usually reinforced at regular intervals by the ribs and so here an interval of 800 mm has been considered constraining only translation and rotation in Z direction allowing it to have displacements in longitudinal and lateral directions. At one end the boundary condition is given as totally constrained as it is assumed to be as a fixed end and the other end with an axial loads (Fig 2) either in tension (8000 KN) or compression (6000 KN)

III THEORETICAL ANALYSIS

The design and analysis of any of these bolted joint invariably require analytical tools for the determination of the stress distribution in the vicinity of bolt holes. It is needed to analyze and compare theoretically how the strength and stiffness of the ply varies with the variation in the composition percentages of different angled lamina. Initially 0, 45 & 90 ° lamina were taken for our plies, these being the industrial standard used in wing construction and most fatigue resistant lamination sequence [3].The specific advantage of it is the 0 ° fibers take the lateral aircraft axial loads and also the bending moments [2]. The 45 ° fibers take the torsion, buckling and twisting of wing. For deciding the range of variation of percentages, the 10% rule for laminate characteristics is used.

Variation of stiffness vs. AML (Angle minus Laminates) is plotted (Fig 3) to decide upon the composition of the required percentages of plies. The observations made here are like: As the AML increases the E_{xx} value decreases linearly, so for negative AMLs the E_{xx} is higher. As 0° % increases for the same 45° %, the E_{xx} increases linearly. Also for same 0° % series, as 45° % increases, E_{yy} decreases.

This is observed opposite to E_{xx} character. In any constant 0% series, as AML increases, E_{xx} decreases for variation in 90° % linearly, but with lower slope than 45° % series. From these, optimum value can be taken according to the needs. Eg: 10% of 90° gives max of G_{xy} & E_{xx}, but min of E_{xx}. As plate is thin and there are no out-of-plane loads, then the plate is considered to be under plane stress.

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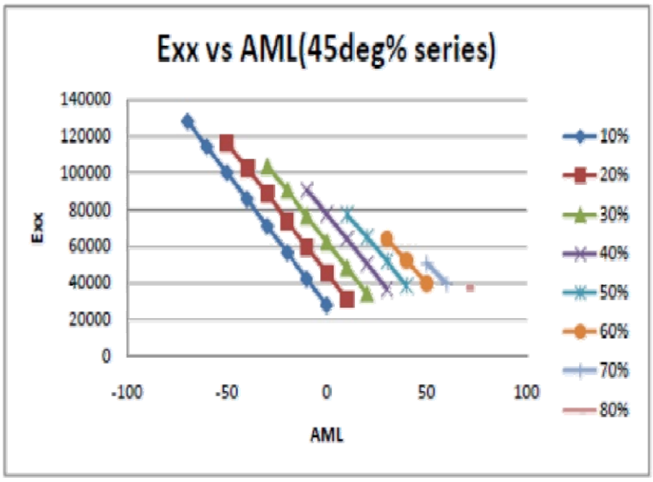

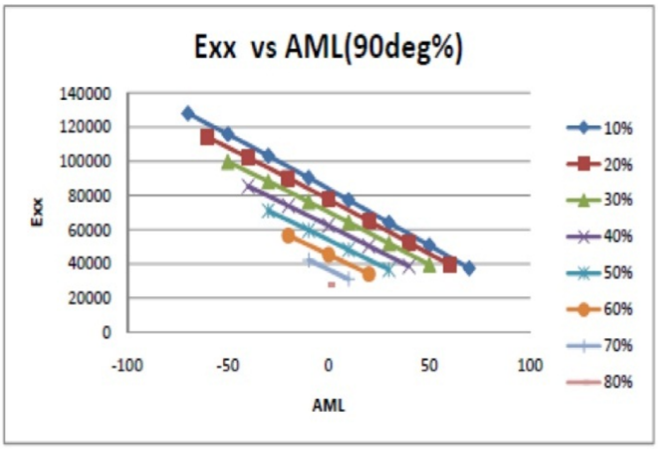

Fig 3: Structural properties (vs.) AML a) 45 ° b) 90 °

For every component, properties have being taken from standard booklets. Skin (T800_M21, 268gsm), stringer and joint straps (T800_M21, 198gsm) have been assumed to be made of high strength carbon fiber, reinforced with epoxy resin. Ti-Al bolt is chosen with 10 % elongation at break. Their properties are given in Table 1.

The load is applied at the point where neutral axis coincides with web, to create pure axial loading. The fastener flexibility is a measure of the influence of fasteners (rivets, bolts, etc.) on the flexibility of joints between the sheets of material from which most modern aircraft are constructed.

Flexibility = f / F

Where,
f- Deflection of joint due to fastening,
F- Load applied.

In fact, the very definition of fastener flexibility is the total joint Flexibility minus the flexibility that would have resulted purely from sheet stiffness. The total force F_{total} at the first fastener row of a joint is split into a bypass force F_{bp} and a load-transfer force F_{lt}, which itself is comprised of the bearing force F_{br} (between the shank of the fastener and the holes of the sheets) and the friction force F_{fr} (between the sections of sheet material). For calculating the bolt stiffness, the formula of Huth [6] has been used.

The classical works by Hart-Smith give a deep insight into the practical joint design considerations in primary aircraft structure components viz., fuselage, wing, horizontal and vertical tails, control surfaces, etc. Hart-Smith also emphasized the need to design the joints first, and then optimize the basic structure in order to avoid the design of a structure which is not practically feasible to assemble, or has a weak design. For composite laminates, these design curves account for stress concentration effects on strength ranging from unloaded fastener holes to loaded fastener holes where all load are transferred in bearing. Strength variations due to lay-up are accounted for by strength carpet plots and they describe joint failure modes of shear-out, net section, and bearing.

These values are inputs to the next calculations, with a formula to calculate the loads that act on each component from the total load applied as

$F_c = F_t \times SR_c$

Where,
F_c - Force acting on component,
F_t - Force (total) applied,
SR_c - Stiffness Ratio of component

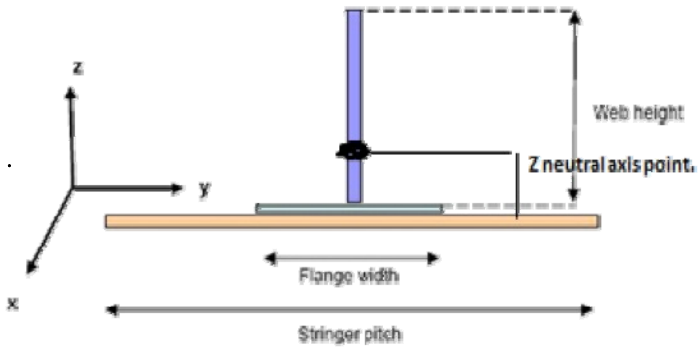

Fig 4: Basic model

Table 1. Material properties

Component	Exx(GPa)	Eyy(GPa)	Gxy(GPa)	Tensile Ult(MPa)
Skin	84.85	43.19	18.58	
Stringer -flange	107.27	33.23	14.18	
Bolt	3	114	44	900

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IV. DESIGN CURVES

With the composition of (0°, 45°, 90°)
Skin lay-up - 35/50/15, 50/40/10, 55/35/10,
Stringer lay-up - 55/35/10, 62/18/10.
Thickness of skin - 5mm, 7.5mm and 10mm
Flange thickness - 5.336mm,
Web thickness - 10.672mm,
Diameter of bolt - 6.35mm,; 7.9375mm; 9.525mm.

The various types of stress with their magnitudes are calculated along with the reserve factors too. For this purpose, the program written in VBA macro in Excel sheet is used. These are given as inputs to the late properties in theoretical analysis. (Fig 4) The bolt parameters and positions are also declared. So this is formulated for analyzing a joint, having a single element between successive bolt nodes. The Forces are applied at each node, at the ends of the components and also the constraints reflecting the boundary condition are given.

Table 2: Strength Allowable

a) Fastener Diameter (mm)	Shear Strength (N)				
6.35	20684				
7.94	32472				
9.53	46706				
b) Laminate Strength	25/50/25	35/50/15	50/40/10	55/35/10	62/28/10
Open hole compression (Mpa)	306	373	475	510	550
Open Hole Tension (Mpa)	562	686	873	937	1026
Bearing Strength (Mpa)	95	116	143	158	173

Plates' stiffness is calculated using the formula: Stiffness= (A*E)/L. Using the plate's stiffness, it has been possible to formulate the matrix needed, as from the formula Used in FEM. This is inverted to get the flexibility matrix.

Reserve Factors (RF) are ratios, non-dimensional numbers to represent the quality of the design, whether it is under-designed, over-designed, or optimized for such joint. This is used as a relative number to judge the load taken by the joint is appropriate or other alternative better designs are possible or not. Optimized joint will have the load taken by component approximately equal to the allowable load, indicating that the component is fully utilized.

$RF= F_{allow}/F_{acting}$

Bearing Loads on the bolts are taken from the forces calculated for bolts already mentioned. These are calculated to find out, if they are within the allowable limits of the bolt bearing stress first and secondly, to get the RF of bolts which tells if it is used to its allowable limits or less. Critical Bearing is the bearing load that acts, when the bypass load is zero. Here the failure will be due to the bolt only. Bypass loads are taken by the plates that they act on. Being a part of the load transferred, they are also verified with the failure loads of composite laminates as in the previous case. Composite components' RF is also affected by this. For comparison as mentioned earlier the maximum allowable values are listed in Table 2, for each combination.

V. RESULTS AND CONCLUSIONS

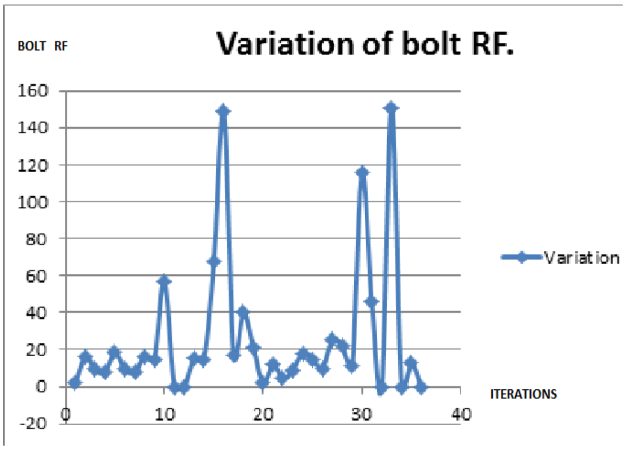
From B101 Model:

Variation in strain: It happens in a cyclic manner, as shown (Fig 5a, b, c). Strain remains constant with skin thickness, even when the bolt diameter changes. It gets reduced as skin-thickness increases, invariantly. As the percentage of 45 ° plies decrease in both the skin and stringer, the strain also reduces.

Variation in Bolt RF: The iteration having taken the second bolt diameter, always has the minimum RF compared to other two iterations having first and third diameter, for no obvious reason, but just optimization. So this works out for for any combination of skin, stringer parameters. As the 0° ply % increases, slight increases can be seen to account for the increase in the bypass load taken by the laminate so in-turn decreases the bearing loads.

Variation of Minimum RF of Laminates: The variation of RF in iterations with the variation in bolt diameter seemed have a negative slope followed by positive slope (second to third bolt diameter). The first stringer layup gives maximum RFs, as opposed to Bolt RFs.

Similarly the plots are obtained for models with different extent of damage (A101, C201 etc). So these show the trend of variation of RF with variation of each parameter discussed before.



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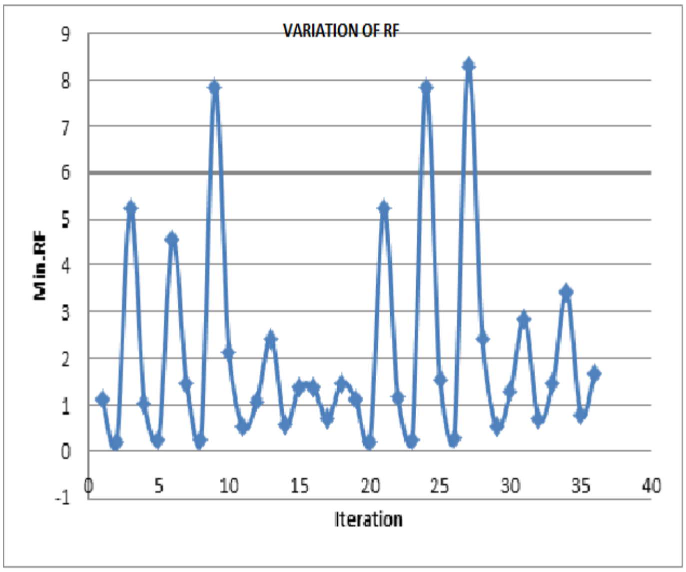

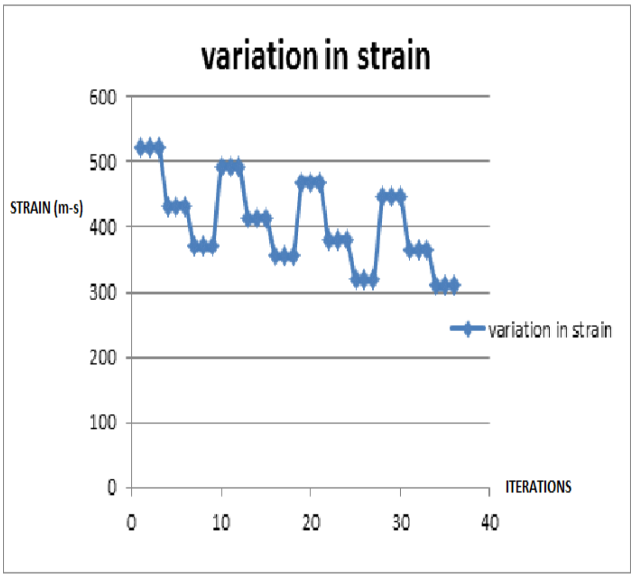

Fig 5 a) Variation of bolt RF b) variation of RF c) variation in strain

Table 3 Results and Comparison

MODEL CONFIGURATION	FEA RESULT (strain in mm)	NUMERICAL RESULT(strain)
B101-Tension	5.23e-3	4.58e-3
B101-compression	-4.28e-3	-3.88e-3
B201-tension	5.25e-3	4.57e-3
B201-compression	-4.66e-3	-3.96e-3
C201-tension	5.11e-3	4.61e-3
C201-compression	-4.78e-3	-4.1e-3

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