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Finite Element Analysis of Multi-Fastened Bolted Joint Connecting Composite Components in Aircraft Structures

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ABSTRACT
In this study contact analysis has been carried out using 3-Dimensional finite element analysis on single lap joint in composite structure with multi-fastener configuration using ABAQUS®. Composite joints with two fasteners and three fasteners are studied for comparative study. The distribution of contact pressure, stress /strain along material axis and radial axis are studied in detail. The shear force distribution among all fasteners is estimated using contact analysis. The failure index of each ply is calculated by using Yamada Sun's failure criteria. The failure load of joint is estimated using the maximum failure load in the laminate.

It is found that in Multi fastened joints the contact pressure variation is non uniform. The maximum stress strain values coming along the material axis. The stress strain distribution in all lamina is observed in local region. The effect of shear modulus (G_{23}) and Poisson's ratio (μ_{23}) does not have any impact on the stress and strain distribution that is used in prediction failure load of the joint.

INTRODUCTION

Composite Joints

Composite have very little capacity to redistribute loads. There are six basic factors to be considered in the design of a composite joint [3].

- The loads which must be transferred.
- The region within which this must be accomplished.
- The geometry of the members to be joined.
- The environment within which the joint must operate.
- The weight/cost efficiency.
- The reliability of joint.

The most efficient composite joints are scarf and stepped lap joints. Double lap, single lap joints are less efficient. The general design requirements for joints are (a)fitting factor (b) overall joint efficiencies (c)eccentricities (d) supported joint (e)joint rigidity (f)mixed fasteners and fittings (g)mixed splice materials(h)fastened and bonded joint (i)permanent set (j)splices adjacent to continuous members.(k)fastener spacing and edge distance(l) counter sunk fastener(m)adjacent skin buckling.

Failure in Composite Joints

Composite joints fail in a number of different macroscopic modes. The most frequent ones are illustrated in 2. The net-section and cleavage modes are abrupt, with a well fined failure load, whereas bearing and shear-out usually are more ductile.

Shear out-Failure:

Caused by shear stresses and occurs along shear out planes on hole edge, typical failure mode when end distance is short.

Tension (net-section) Failure:

Caused by tangential tensile or compressive stresses at the edge of the hole. For uni-axial loading conditions, failure occurs when bypass/bearing stress ratio is high.

Bearing Failure:

Occurs in area adjacent to contact area due to compressive stresses, likely when bypass/bearing

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stress ratio is low, strongly affected by through-thickness clamping force.

Bolt pull-through Failure:

Due to low through-thickness strength of composite material.

Bolt shear failure:

Cause by high shear stresses in the bolt.

LITERATURE SURVEY

The stress distribution around a hole in multi fastened composite laminate system is depends on stacking sequence, bolt hole clearance, bolt load, contact behavior, orthotropic material property etc. The mechanical fastening joints in composite structure should be designed such a way that it fails only in bearing failure mode as it gives prior warning by yielding the hole, before failure of the joint. Finite element analysis is the best numerical method to analyze composite joints because the analysis results are showing good agreement with the experimental results, thereby save time and effort of testing can be eliminated.

Problem Statement

This thesis is aimed at the load distribution among the fasteners (protruding head) in a multifastened composite joint in aircraft structures, by using 3-D analysis of the

- 1-bolted single lap joint
- 2-bolted single lap joint
- 3-bolted single lap joint

The joint in the inter spar rib connecting top skin and bottom skin of the wing structure was taken for the studies. Due to symmetry, only one side of the joint is considered for the analysis.

Objective

The objective of this thesis is the 3-Dimensional finite element analysis of multifastened composite laminate and investigates the following:

- Load distribution among the fasteners.
- Critical failure load.
- Deflection Of the laminate
- Stress Strain Distribution in each ply orientation.
- Comparing the maximum stress /strain in each ply with the theoretical maximum stress/strain theory.
- Variation of maximum stress/strain angle in each ply.
- Effect of Poisson's ratio and shear modulus along the thickness direction

METHODOLOGY

The methodology of the present study was explained in this chapter.The geometry was modelled in Hypermesh®. It is exported to ABAQUS® for analysis. In this model the input parameters like orthotropic material properties, lay up sequence contactparameters, boundaryconditions loads were assigned. After the analysis the post processing was done in ABAQUS®. The results were extracted according to the objective of the present study. The methodology in the form of flowchart was as shown in **Ошибка! Источник ссылки не найден.**

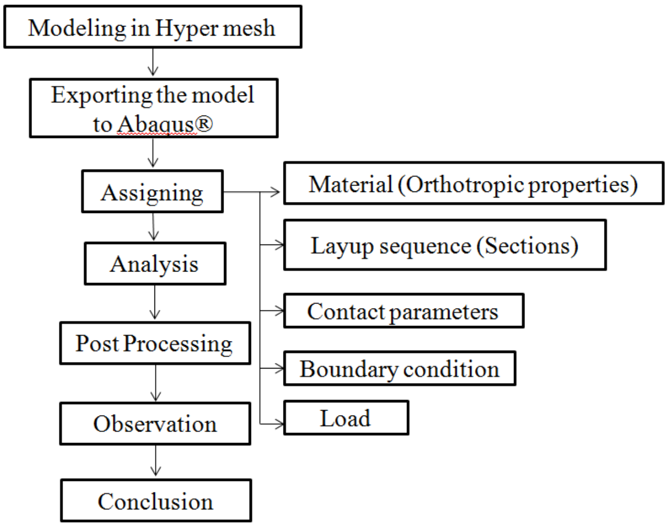

Fig 3.1Flow chart for methodology

FINITE ELEMENT ANALYSIS

An understanding of the structure is important to study the behavior of the bearing phenomena. The aim of computational study is to build a model which can be

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used in the future course to study a series of different combination of parameters without going for experimental study for each case. There are some models developed in the past some authors using different software packages like ABAQUS®, LS dyna, Ansys etc.

A nonlinear finite element analysis was carried out to understand the behavior of three dimensional model of multi fastened composite single lap joint by using ABAQUS®.

Small Sliding versus Finite Sliding

ABAQUS® does provide three schemes in defining relative surface motions, namely small, finite and infinitesimal sliding. In infinitesimal sliding, the scheme assumes that both the relative motion of the surfaces and the absolute motion of the contacting bodies are small. Similarity and differences between small and finite sliding are given as follows.

- Both formulations allow two bodies to undergo large motions. However limitation in the small sliding is that it assumes that there will be a relatively small amount of sliding of one surface along the other.
- The slave node can transfer load to any nodes on the master surface in the finite sliding while it can only transfer load to a limited number of nodes on the master surface for the small sliding.
- In small sliding analysis every slave node interacts with its own tangent plane on the master surface and consequently slave nodes are not monitored for possible contact along the entire master surface. Thus the small sliding is less expensive computationally than the finite sliding contact.

Small sliding scheme will be adopted for analyses due to its computational advantage over the finite sliding. In bolted joints the contacting surface does not slide more than a small fraction of a typical element dimension, so in this case small sliding is suitable.

Geometric Details

The geometrical detail of the single lap joint with1-Bolted joint, 2-Bolted joint, 3-Bolted joint with single lap joint.

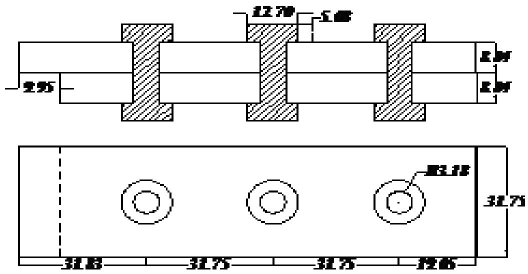

Fig 4.1 3-Bolt Single lap Joint

Composite Laminate Details

- No of plies in each laminate:52
- Thickness of each laminate: 0.17mm
- Assumed thickness of laminae for the analysis: 0.34mm (two adjacent layers are laminate are with same orientation. In this study they have been looked as a single lamina. By assuming so, the number degree of freedom has reduced by half. Therefore the problem can be solved at the faster rate)
- Total thickness of each laminate: 8.84 mm
- Layup sequence(45,0,-45,0,90,0,45,0,90,0,-45,0,90)Sym

Composite Laminate Details

The layup sequence of the composite laminate is as shown in Ошибка! Источник ссылки не найден.. The stacking sequence considered is symmetry about mid plane of laminate. The top and bottom laminates are having same stacking sequence.

Layer No	Stacking sequence
1 *	+45 (Top)
2	0
3	-45
4	0
5	90
6	0
7	+45
8	0
9	90
10	0
11	-45
12	0
13	90
14	90
15	0
16	-45
17	0
18	90
19	0
20	+45
21	0
22	90
23	0
24	-45
25	0
26 **	+45

* Top lamina of top laminate

** Bottom lamina of a top laminate

Table 4.1 Layup Sequence

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Material Properties

The properties of material used for both composite and fasteners are given in **Ошибка! Источник ссылки не найден.**.

S. No			Composite laminate	Fastener
	Name of the property	Units	T800*	Titanium
[1]	*[2]*	*[3]*	*[4]*	*[5]*
01	E1	(MPa)	150000	111000
02	E2	(MPa)	9000	
03	E3	(MPa)	9000	
04	µ12		0.28	0.3
05	µ13		0.28	
06	µ23		0.36	
07	G12	(MPa)	4000	
08	G13	(MPa)	4000	
09	G23	(MPa)	4000	
10	Density	(Kg/mm³)	1.63E-06	4.48E-06

(*3960-2 graphite-epoxy pre-preg)

Table 4. 2 Material Properties of the laminate and fastener.

FE Modeling

The three dimensional model is developed using Hypermesh® as a pre-processing software. These elements are imported in ABAQUS® CAE as orphan mesh. Then the element types C3D8R are assigned to the elements. The details of quality checks made are given in **Ошибка! Источник ссылки не найден.**. It is seen that all parameters of element are meeting the require limits.

Name	Value	Limit
Warpage	0.02	<15⁰
Aspect ratio	5.01	5:1
Skew angle	44.06	<60⁰
Min length	0.2703mm	
Max length	1.7mm	
Jacobian	0.70	0-1.00(>0.6)

Table 4.3 Element Qualities Check

The finite element models for 1-bolted to 3-bolted joint are shown in **Ошибка! Источник ссылки не найден.**3, Figure 4.4 and Figure 4.5 respectively. The meshing details of fastener are also shown in **Ошибка! Источник ссылки не найден.**and it is the same for all fasteners used in three models.

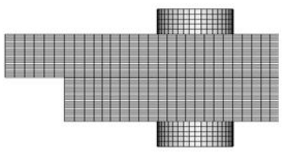

Fig 4.31-bolted single lap joint

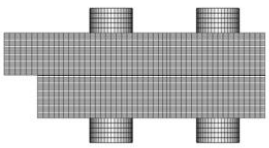

Fig 4.4 2-bolted single lap joint

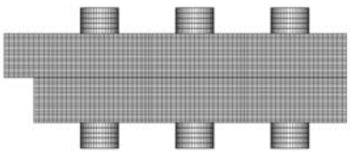

Fig 4.53-bolted single lap joint

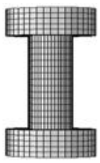

Fig 4.6 Fastener

Loading

The loading that has been applied on all models is the same which is obtained from the actual composite wing structure. The detailed calculation of loading

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applied on FE models are shown in the

S.No	Description	Value	Unit
1	Applied flow of stress (obtained from actual wing model)	910.41	N/mm
2	Width of the laminate	31.75	mm
3	Total force that should be applied on the laminate (910.41 x 31.75)	28905.52	N
4	Equivalent pressure that can be applied on the laminate (28905.52/(8.84 x31.75))	102.99	M Pa

Table 4. 4 Calculation of the applied pressure load

. The loading is applied on top laminate which is trying to pull the top laminate away from the boundary condition as shown in **Ошибка! Источник ссылки не найден.**.

S.No	Description	Value	Unit
1	Applied flow of stress (obtained from actual wing model)	910.41	N/mm
2	Width of the laminate	31.75	mm
3	Total force that should be applied on the laminate (910.41 x 31.75)	28905.52	N
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Table 4. 4 Calculation of the applied pressure load

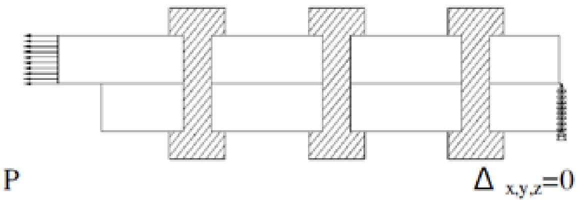

Fig 4.7Loading and boundary condition

Boundary Condition

The nodes on bottom laminates are constrained against all translation as the constraint of rotation are not needed for 3D analysis as shown in **Ошибка! Источник ссылки не найден.**. The boundary condition is applied at the end to simulate the actual behaviour joint in the actual wing structure. The boundary conditions are same for all three models.

Contact Details

In FE analysis the contact conditions are special cases of discontinuous constrains, allowing forces to be transmitted from one part of the model to the other. The procedure for contact analysis is described below:

1. Created contact surface (master and slave) between the bolt and hole, between the laminates. Created normal from master surface.
2. Defined the contact interactions. (The parameters considered for the contact analysis are described in section 4-3.)
3. General contact is defined for the whole structure.
4. Contact pair is defined for the contact in between the fastener and laminate, between the top and bottom laminate.Contact interaction Scheme (as shown in **Ошибка! Источник ссылки не найден.**)
5. Contact controls, Automatic closure, Unsymmetrical solver are given for the early convergence.

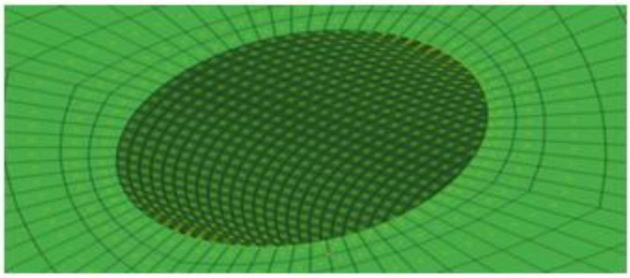

Fig 4.9Contact interaction schemes

RESULTS AND DISCUSSION

The load distribution in the laminate at each fastener location has been arrived at using contact normal force and contact frictional force values around fattener holes in top and bottom laminates of all models. It has been identified the fastener that is subjected to maximum shear force among all other fastener for two and three bolted joint. Other output data is extracted in the laminate only at this fastener location as the failure of laminate lies at this fastener location. The contact pressure distribution around holes is extracted. The stress and strain distribution in each lamina along the material axis system has been extracted. That stress and strain distribution along radial axis is extracted by creating a separate cylindrical coordinate system at the center of the hole.

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	Top laminate		Bottom laminate	
	CNF1(N)	CSF1(N)	CNF1(N)	CSF1(N)
[1]	*[2]*	*[3]*	*[4]*	*[5]*
Bolt-1	2.55E+04	2.16E+03	-3.03E+04	4.30E+03
B/w laminate		1.24E+03		-1.69E+03
Total	2.89E+04		-2.77E+04	
Applied load	2.89E+04			
Difference %	0.00		-4%	

Table 5.1 Load distribution for 1-Bolted single lap joint

	Top laminate		Bottom laminate		Total load	%distribution
	CNF1 (N)	CSF1(N)	CNF1(N)	CSF1(N)		
[1]	*[2]*	*[3]*	*[4]*	*[5]*	*[6]*	*[7]*
B/w laminates	0.00E+00	1.81E+02	0.00E+00	-1.65E+02		
Bolt 1	1.51E+04	1.69E+03	-1.54E+04	-1.87E+03	1.68E+04	58.19
Bolt 2	1.09E+04	8.51E+02	-1.12E+04	-5.21E+02	1.18E+04	40.74
Applied load	2.89E+04					
Sum	2.88E+04		-2.92E+04			
Differences (%)	0.34		-1.00			

Table 5.2 Load distribution for 2-bolted single lap joint (Bolt-1)

	Top		Bottom		Total load	% of distribution
Bolt No(Figure)	CNF1(N)	CSF1(N)	CNF1(N)	CSF1(N)	(N)	
[1]	*[2]*	*[3]*	*[4]*	*[5]*	*[6]*	*[7]*
Bolt-1	1.32E+04	1.27E+03	-1.40E+04	-1.65E+03	1.44E+04	49.89
Bolt-2	7.29E+03	4.62E+02	-7.52E+03	-2.90E+02	7.75E+03	26.83
Bolt-3	5.78E+03	4.72E+02	-5.90E+03	-3.91E+02	6.25E+03	21.61
B/w the laminate	0.00E+00			3.13E+02		
Sub total	2.62E+04	2.20E+03	-2.75E+04	-2.02E+03		
Total	2.84E+04		-2.95E+04			
Applied load	2.89E+04					
Difference	0.02		-0.02			

Deflection of Joint

The clearance of 0.01 mm called between the hole and bolt also causing bending of fastener, however, the bending tendency depends on the material properties of both laminate and fastener. It has been observed that the top and bottom laminate comes in contact with each other on the side where the external load is applied, whereas they try to open up on the side where the rigid boundary conditions are applied. This behavior is attributed to the fact that the laminates are subjected to tensile force.

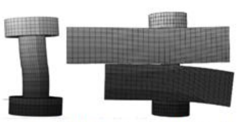

Fig 5.1Deflection of 1-bolted joint

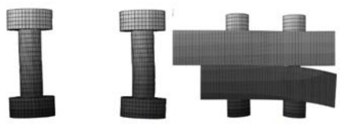

Fig 5.2Deflection of 2-bolted joint

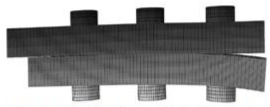

Fig 5.3 Deflection of3-bolted joint

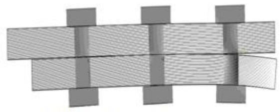

Fig 5.4 Overall behavior of 3-bolted joint

Failure Prediction in Composite Laminate

Prediction of failure of multi-fastened joint in composite laminate is prime objective of this study after carrying out contact analysis. The maximum stress and strain distribution is understood through contact analysis. The failure of joint can be estimated by using the maximum stress values in each ply of laminate and allowable stress values applying Yamada Sun's failure criterion (Eq-5.1) that had been modified to suit for unidirectional pre-pregs composite materials. Yamada Sun's applied the failure theory to a lamina in plane stress (Eq-5.1).The present study has been carried out using only axial load applied on the joint no shear force is applied. Because of which, the following equation further simplified into (σ_{11}/ X) form which indicate failure index (FI) values in a given ply. If the value is less than 1.00 the ply is considered to be safe, greater than 1.00 is considered to be failed. The failure index values are calculated for 3-bolted joint at bolt-1 which is subjected to maximum shear force.

The detailed calculations on FI values performed in this connection are given in **Ошибка! Источник ссылки не найден.**1 to4.The failure index values in each ply are calculated using allowable tensile and compressive stress values. The allowable tensile value is more than compressive stress however reason not discussed in this study as it is out of scope of the study.

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$$\sqrt{\left(\frac{\sigma_{11}}{X}\right)^2 + \left(\frac{\tau_{12}}{S}\right)^2} < 1 \qquad \text{Eq (5.1)}$$

Where

σ_{11} - Tensile stress from applied load,

τ_{12} - Shear stress from applied load,

X- Allowable strength of the lamina in the fiber direction,

S- Allowable shear strength of the lamina.

Sl.No	Orientation	Max. Radial stress	Max. Material stress	Max. Allowable tensile stress (test)	Angle of Maximum Radial stress	Angle of Max. Material stress	FI (Yamada Sun's)
[1]	[2]	[3]	[4]	[5]	[6]	[7]	[8]
1	45	28.45	6.54	1200	25	225	0.01
2	0	362.94	50.49	1200	25	90	0.04
3	-45	191.55	21.03	1200	25	360	0.02
4	0	363.75	58.03	1200	25	90	0.05
5	90	25.53	14.36	1200	25	350	0.01
6	0	329.91	168.03	1200	25	180	0.14
7	45	75.44	180.86	1200	55	150	0.15
8	0	325.50	259.73	1200	25	180	0.22
9	90	25.55	16.20	1200	25	160	0.01
10	0	351.33	342.73	1200	25	5	0.29
11	-45	191.69	333.29	1200	25	35	0.28
12	0	340.99	433.08	1200	25	5	0.36
13	90	33.34	46.75	1200	25	110	0.04
14	90	30.50	100.26	1200	110	110	0.08
15	0	364.49	582.11	1200	25	5	0.49
16	-45	218.83	665.72	1200	145	50	0.56
17	0	383.71	684.06	1200	25	5	0.57
18	90	64.66	261.17	1200	110	90	0.22
19	0	383.31	818.95	1200	25	180	0.68
20	45	228.46	1010.46	1200	55	135	0.84
21	0	398.51	965.05	1200	25	180	0.81
22	90	63.17	508.51	1200	110	90	0.42
23	0	468.47	1120.48	1200	25	5	0.93
24	-45	273.92	1308.15	1200	25	50	1.09*
25	0	519.89	1293.06	1200	25	180	1.08*
26	45	47.61	1123.58	1200	170	140	0.93

* FI > 1.00, therefore these two plies are failed when subjected to tensile stress

Table 5.1 Failure index values in each ply -Max tensile stress for top laminate

Sl.No	Orientation	Max. Radial stress	Max. Material stress	Max. Allowable tensile stress (test)	Angle of Maximum Radial stress	Angle of Maximum Material stress	FI (Yamada Sun's)
[1]	[2]	[3]	[4]	[5]	[6]	[7]	[8]
1	45	96.37	2275.20	1200	360	320	1.89
2	0	318.58	3984.41	1200	150	185	3.33
3	-45	178.73	2423.01	1200	190	230	2.00
4	0	271.59	3442.86	1200	155	185	2.86
5	90	181.38	757.95	1200	360	270	0.63
6	0	223.50	2895.04	1200	30	360	2.44
7	45	113.72	1659.56	1200	180	315	1.39
8	0	196.41	2267.28	1200	270	360	1.89
9	90	84.78	278.29	1200	360	270	0.23
10	0	219.55	1658.75	1200	270	185	1.39
11	-45	226.05	953.57	1200	325	215	0.79
12	0	260.55	1138.61	1200	270	185	0.95
13	90	19.76	98.64	1200	290	275	0.08
14	90	15.16	45.45	1200	290	290	0.04
15	0	166.97	393.61	1200	200	185	0.33
16	-45	50.82	398.34	1200	290	25	0.33
17	0	77.89	433.99	1200	270	25	0.36
18	90	25.61	114.78	1200	290	275	0.10
19	0	63.87	495.30	1200	270	25	0.41
20	45	52.49	52.25	1200	50	50	0.04
21	0	44.61	562.99	1200	270	25	0.47
22	90	45.60	142.02	1200	160	275	0.12
23	0	47.59	693.05	1200	90	25	0.58
24	-45	154.74	733.75	1200	110	25	0.61
25	0	231.85	840.03	1200	90	25	0.70
26	45	82.55	311.71	1200	40	55	0.26

* Plies with FI > 1.00 failed. The number of plies in bottom laminate failed more than top laminate due to two reasons that are eccentricity and boundary condition applied on bottom laminates.

Table 5.2 Failure index values in each ply -Max tensile stress for bottom laminate

Observations:

The failure index values calculated in above tables show that the bottom skin member that is subjected to compressive load may fail at the load just above P/5.88 (Applied load / Max. FI), when the failure index value is just more than 1.00. The same joint may fail at the load just above (P /3.33) times the originally applied load when subjected to tensile load. However, these observations need to be further confirmed / discussed by carrying out testing on the joint schemes.

CONCLUSION

The conclusions drawn from the present study as mentioned below.

- It has been understood that the behavior of top and bottom laminate in the single lap joint is

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not same as the top laminate is free from constraint, and bottom laminate is constraint from free displacement at the point where boundary conditions are applied.

- The maximum stresses and strains values are noticed along the material axis in the composites, as the material stiffness is available only in this direction.
- The contact pressure distribution in the case of multi fastened system is non-uniform due to the effect of shear modulus and Poisson's ratio associated with the number of fastener used in the joint.
- The maximum stress values obtained from analysis are compared with that of allowable stress values to find the failure index values in individual lamina of laminate. It has been observed that many lamina failed as the values of failure index found to be greater than 1.00.
- In stress and strain distribution in all lamina is observed in local region which may trigger the sudden or brittle failure of the joint.
- The effect of shear modulus (G_{23}) and Poisson's ratio (μ_{23}) does not have any impact on the stress and strain distribution that is used in prediction failure load of the joint.

FUTURE SCOPE

- Further studies on the extended topic can be continued in the area prediction of failure modes using same ABAQUS tools.
- The bearing bypass studies can be initiated using same models and estimate the joint failure load for clear fit and loose fit with different clearance values practically occurring in the shop floor.
- Studies can be repeated by applying bolt pre tension load for studying the effect of pre tension load in each composite plies.
- Investigate the behaviour of multi fastened joints subjected to high dynamic loading using three dimensional analyses.
- Analytical studies to see the effect of counter sunk fasteners on the bolt strength can be initiated.

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