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Design And Analysis On Aircraft Wing To Fuselage Lug Attachment

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Abstract : There are two classes of loads which acts on the aircrafts which are surface forces and body forces. Surface forces include aerodynamic and hydrostatic pressure forces while body forces are due to gravitational and inertial effects on the volume of the structure of an aircraft. In these aerodynamic forces are more severe. With the pressure load which occurs due to the production of lift force, bending moment and shear forces will be acting on the wing. These are the most important cases of forces while studying about the Wing to Fuselage lug attachment connections. Rarely an aircraft will fail due to a static overload during its service life. For the continued airworthiness of an aircraft during its entire economic service life, fatigue and damage tolerance design, analysis, testing and service experience correlation play a pivotal role.

In the current paper, an attempt will be made to predict the fatigue life of wing to fuselage lug attachment in airframe. Fatigue life calculation will be carried out for typical service loading condition using constant amplitude Stress-No. of cycles(S-N) data for various stress ratios. The software used are solid edge V19 for modeling lug and MSC Patran and Nastran for detail analysis.

Keywords -Wing to fuselage lug attachment, stress concentration, Fatigue life, S-N data, Factor of safety, Occurrence, Exceedence and Damage tolerance.

I. Introduction

Lugs are the connector type of elements widely used for structural supports in airframe structure. A wing to fuselage attachment fitting is considered for fatigue load analysis. The entire model considered for analysis is made of steel alloy, heat treatment AISI 4340, which is a standard in aerospace industry[3]. The stress life (S-N) based method is used for life estimation of the component under a given flight load spectrum. The total damage of the part is obtained by FE analysis in MSC Nastran software. This damage is used to find the life of the component in hours. The flight spectrum considered for analysis is a standard in aerospace industry. Alternating forces in terms of 'g' are applied on the component. With the applied loading, the stresses obtained in the structure far exceeds the ultimate point strength of the material, hence case 2 of the problem is also considered with a slight modification of the structure.

1.1. Material Specification

Selection of the material depends upon any considerations, which can in general be categorized as cost and structural performance. Cost includes initial material costs and maintenance costs. A combination of various materials is often necessary i.e alloys are used. The key material properties that are pertinent to maintenance cost and structural performance are, Density, Stiffness, Strength, Durability ,Damage tolerance and Corrosion.

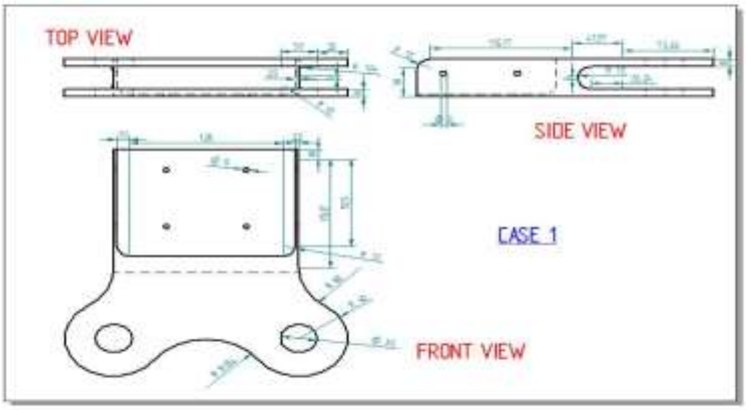
II. Problem Definition

- To design a wing to fuselage lug attachment against fatigue failure.
- Linear static stress analysis of the wing to fuselage lug attachment.
- Calculation of the fatigue life to crack initiation in the wing to fuselage lug attachment.

Problem definition in wing to fuselage lug attachment
⇩
Modeling of wing to fuselage lug attachment in solid edge V19 software
⇩
Meshing of wing to fuselage lug attachment
⇩
Selection of material for wing to fuselage lug attachment
⇩
Applying of boundary conditions for wing to fuselage lug attachment
⇩
Stress analysis for wing to fuselage lug attachment by using MSC patran and nastran software
⇩
Fatigue life estimation of wing to fuselage lug attachment

III. Modeling Procedure

The below figure represents the modeling of Wing to fuselage lug attachment of both the cases in solid edge V19[3]. It represents the Top view, side view and Front view of fitting.



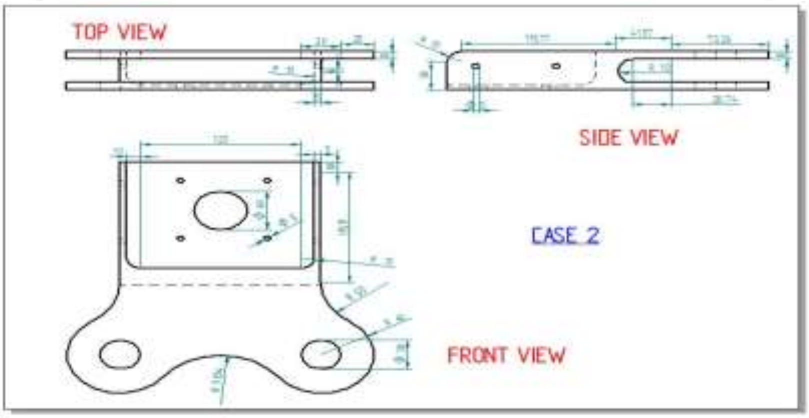


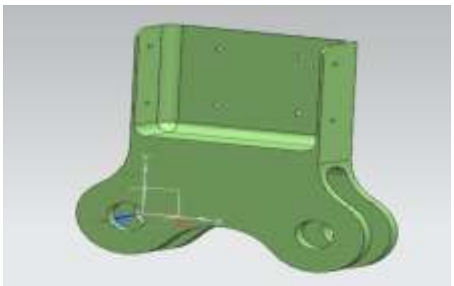
TABLE I. DIFFERENCES IN BOTH THE CASES.

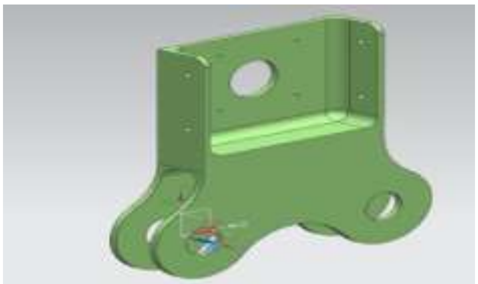
CASE 1	CASE 2
A circular hole at the web is not created	A circular hole at the web is created so as to reduce the weight of the lug
The web thickness taken here is 2mm	The web thickness taken here is 5 mm
Flange thickness taken here is 2.5mm	Flange thickness take here is 5mm

IV. Methodology

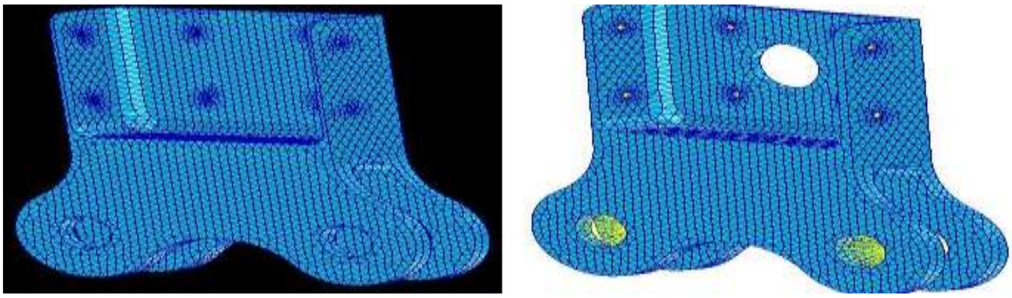
The load calculations are done for wing to fuselage lug attachment. Create a 3D model using solid edge V19 software. Import the 3D model into MSC patran to perform static and dynamic stress analysis. A finite element pre and post processors (such as MSC PATRAN) is a graphic based software package primarily designed to aid in the development of Finite Element Model (Pre processing) and to aid the display and interpretation of analysis results (Post processing)[3]. MSC PATRAN software is a mechanical computer aided engineering tool created for design engineers[3]. Fatigue life calculation is done to estimate the life of the fitting. Optimization of the fitting is performed to increase the life of the lug.

4.1. Importing Lug into MSC Patran





4.2. Meshing the Lug



4.3. Selecting of Materials

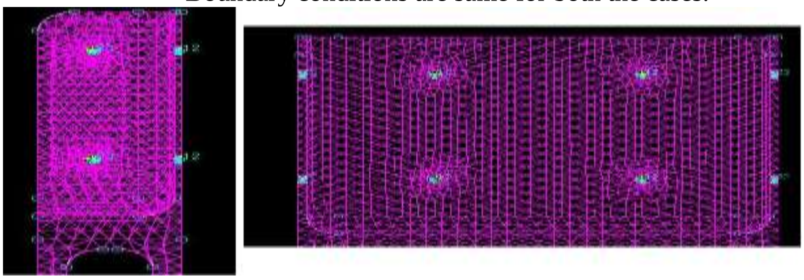
The material used for lug attachment of the structure is Steel Alloy, Heat Treated AISI 4340 with the following properties.

TABLE II: MATERIAL PROPERTIES FOR LUG ATTACHMENT.

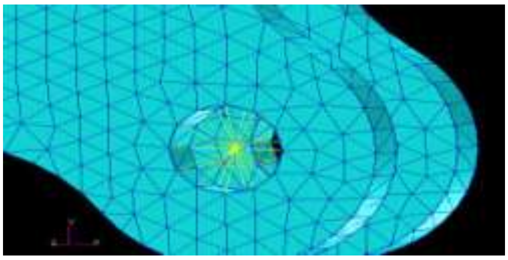
S.NO	PARAMETERS	Steel Alloy,AISI-4340
1	Young's Modulus(N/mm2)	203000
2	Poison's Ratio	0.32
3	Ultimate Tensile Strength (N/mm2)	1835
4	Yield Stress, (N/mm2)	1600.8

4.4. Applying Boundary Conditions

Boundary conditions are same for both the cases.



4.5. Applying Load

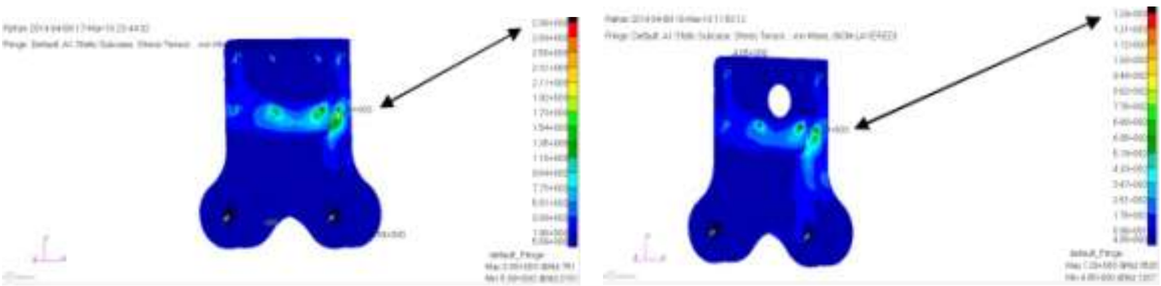


A load of 90584.62N is applied at the fork section of lug of both the cases[3].

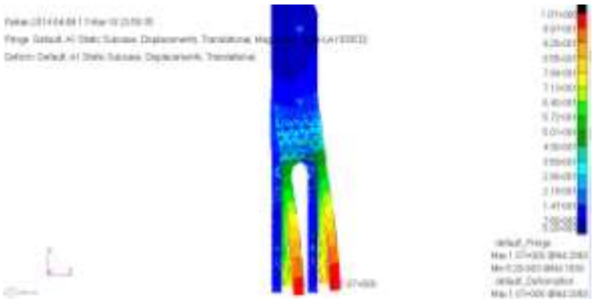
V. Analysis

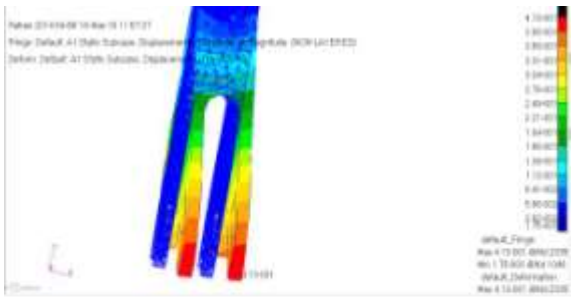
3D model of the lug was developed in solid edge V19 from the paper[3]. The model was then converted into a parasolid to import into MSC patran. Initially load calculations for the wing to fuselage lug attachment are done for 8 cases of 'g'. Static analysis was done by applying the calculated loads for materials Steel Alloy AISI-4340 . From the analysis the displacement and maximum principle stresses are calculated and tabulated.

5.1 Maximum Stress



5.2. Maximum displacement





VI. Fatigue Life

Fatigue is a phenomenon caused by repetitive loads on a structure. It depends on the magnitude and frequency of these loads in combination with the applied materials and structural shape. Structural members are frequently subjected to repetitive loading over a long period of time[2]. Often machine members subjected to such repeated or cyclic stressing are found to have failed even when the actual maximum stresses were below the ultimate strength of the material, and quite frequently at stress values even below the yield strength. The most distinguishing characteristics are that the failure had occurred only after the stresses have been repeated a very large number of times[1]. Hence the failure is called fatigue failure. Formulas to calculate fatigue life are[1],

$$1.\ Resultant\ force(R) = \sqrt{R_x^2 + R_y^2}\ N$$

$$R = \sqrt{R_y^2 + R_z^2}\ N$$

$$Where,\ R_x = force\ in\ X\text{-}direction\ in\ N$$

$$R_y = force\ in\ Y\text{-}direction\ in\ N$$

$$R_z = force\ in\ Z\text{-}direction\ in\ N$$

$$2.\ Shear\ stress(\tau) = \frac{Force}{Area} MPa$$

$$3.\ Factor\ of\ safety(FOS) = Ultimate\ stress/Allowable\ stress$$

$$4.\ Stress\ amplitude(S_a) = \frac{S_{max} - S_{min}}{2}\ MPa$$

$$Where,\ S_{max} = maximum\ stress\ in\ MPa$$

$$S_{min} = minimum\ stress\ in\ MPa$$

$$5.\ Mean\ stress(S_m) = \frac{S_{max} + S_{min}}{2}\ MPa$$

$$6.\ Stress\ equivalent(S_{eq}) = S_{max}(1 - R)\hat{}\,0.52\ MPa$$

$$7.\ No.\ of\ cycles(N) = 10\hat{}(8.35\text{-}3.10\log(S_{eq}\text{-}10.6))$$

$$8.\ Damage(D) = Occurrence(n)/No.\ of\ cycles(N)$$

6.1 Fatigue Life Calculation of both the cases

TABLE III. TABLE SHOWS DAMAGE 0.197MM AND MAIN LIFE 1.0117MM IN CASE 1.

Occurrence (n)	Excedence	g (max)	g (min)	Smax (Mpa)	Smin (Mpa)	Sa	Sm	R	Seq(Ksi)	N(cycles)	Damage	□D	L(Life in hours)	scatter factor	Main life
						(Smax-Smin)/2	(Smax+Smin)/2	Smin/Smx			(D=n/N)		(1/□D)		(hours)
0.01	0.01	8	-2	2880	5.89	1437.05	1442.945	0.00204	417.55	1.8214	0.005	0.197	5.0589	5	1.01179
0.034	0.044	5.5	-1.95	1980	-702	1341	639	0.3545	336.49	3.6262	0.009				
0.076	0.12	5	-1.85	1800	-666	1233	567	-0.37	307.71	4.8298	0.015				
0.17	0.29	4.5	-1.6	1620	-576	1098	522	-0.3555	275.42	6.899	0.024				
0.4	0.69	4	-1.25	1440	-450	945	495	-0.3125	240.74	10.661	0.037				
0.91	1.6	3.5	-0.7	1260	-252	756	504	-0.2	201.05	19.16	0.047				
2.3	3.9	3	-0.15	1080	-54	567	513	-0.05	160.77	40.043	0.057				
7.1	11	2.5	0.3	900	108	396	504	0.12	122.22	∞	0				
27	38	2	0.65	720	234	243	477	0.325	85.182	∞	0				
152	190	1.5	0.75	540	270	135	405	0.5	54.656	∞	0				

TABLE IV. TABLE SHO

Occurrence (n)	Excedence	g(max)	g(min)	Smax (Mpa)	Smin (Mpa)	Sa	Sm	R	Seq (Ksi)	N (cycles)	Damage	□D	L(Life in hours)	scatter factor	Main life
						(Smax-Smin)/2	(Smax+Smin)/2	Smin/Smax			(D=n/N)		(1/□D)		(hours)
0.01	0.01	8	-2	1290	4.05	642.975	647.02	0.0031	186.92	24.34	0.00041	0.01	75.05	5	15.01055477
0.034	0.044	5.5	-1.95	886.875	-314.43	600.65	286.21	-0.3545	150.72	49.64	0.00088				
0.076	0.12	5	-1.85	806.25	-298.31	552.28	253.96	-0.37	137.83	66.95	0.00113				
0.17	0.29	4.5	-1.6	725.625	-258	491.81	233.81	-0.355	123.36	97.33	0.00174				
0.4	0.69	4	-1.25	645	-201.56	423.28	221.71	-0.312	107.83	154.09	0.00259				
0.91	1.6	3.5	-0.7	564.375	-112.87	338.62	225.75	-0.2	90.05	288.11	0.00315				
2.3	3.9	3	-0.15	483.75	-24.18	253.96	229.78	-0.05	72.01	640.26	0.00359				
7.1	11	2.5	0.3	403.125	48.37	177.37	225.75	0.12	54.74	∞	0				
27	38	2	0.65	322.5	104.81	108.84	213.65	0.325	38.15	∞	0				
152	190	1.5	0.75	241.875	120.93	60.46	181.4	0.5	24.48	∞	0				

WS DAMAGE 0.01 AND MAIN LIFE 15.01055HOURS IN CASE 2.

VII. Results

The maximum Stress obtained in the structure, material used for the structure, ultimate strength of the material are shown. In the case 1 max. Stress is more than the ultimate strength of the material. Therefore material is failing. In case 2 max. stress is less than the ultimate strength of the material. According to stress-strain diagram of respective material, structure will not fail for applied load . When the max. Stress is greater than or equal to the ultimate strength of structure than only structure is going to fail. The Total damage accumulated in both structure is less than 1. Therefore a crack will not get initiated from the location of maximum stress in both structures for given load spectrum.

TABLE V. RESULTS OF BOTH THE CASES

S.NO	CASE 1	CASE 2
Name of the structures	Wing to Fuselage Lug Attachment	Wing to Fuselage Lug Attachment
Material used	Steel Alloy, Heat treated AISI- 4340	Steel Alloy, Heat treated AISI- 4340
Ultimate strength of the material (in N/mm2)	1835	1835
Max. stress by FEA(in N/mm2)	2880	1290
Damage Accumulation	0.197	0.0133
No. of cycles	1.821	24..347

VIII. Conclusion

The structural analysis of aircraft wing to fuselage lug attachment for this paper has a safe life as per the loading conditions discussed earlier. Finite element analysis for structural analysis of aircraft wing to fuselage lug attachment is carried out using tet 4 3D elements. Maximum displacement is observed at the fork only and maximum stresses are observed at the rivet holes which are near to fork. After completion of case 1 of the fitting problem the maximum stress in the structure was 2880MPa which is higher than the strength of the steel material AISI 4340 considered here i.e. 1835Mpa. The material is failing. For the life of the component the case 2 with optimization in lug is carried out. After completion of case 2 of the fitting problem the maximum stress in the structure is 1290Mpa which is lower than the strength of the steel material AISI 4340 considered here i.e. 1835Mpa. The material is not failing. Thus obtained the desired validation. This paper summarizes, how to design a aircraft wing to fuselage lug attachment, how to do structural analysis, and fatigue life calculation and will also predict that the geometry is safe on the basis of static structural analysis and finally future scope is given.

IX. Future Scope

Fatigue crack growth analysis can be carried out on the model. Damage tolerance evaluation for the model can be carried out for a given load spectrum. A structural testing of the model can be carried out for the complete validation of all theoretical calculations. Introduce an anticorrosion layer for the surfaces. Introduce an intermediate sleeve to reduce the local stress concentration.

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