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Structural optimization of a aircraft's fuselage using topology

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Abstract— The main objective of this paper is to provide an optimum preliminary basic design for the central part of the fuselage structure using the topology optimization method and to check whether the design is safe or not. Several loading cases like the aerodynamic loads, structural loads etc were considered for the analysis. The central part of the fuselage was modeled using CATIA and it was analyzed using ANSYS software for different parameters like material properties, loads acting on the fuselage etc.. And even the topological parameters were applied. After the 1st analysis or the 1st iteration the model was imported back to CATIA and the regions of low stresses were removed and again analyzed and after subsequent analysis or iterations a fuselage structure with optimum material, lower stress acting regions, optimized structural volume and few other desirable objectives are obtained. Later the analysis is done to get an optimized shape and size of the fuselage.

Keywords —Topology Optimization, Optimized Material, Subsequent Analysis.

I Introduction

Topology optimization is a mathematical method that optimizes material layout within a given design space, for a given set of loads, boundary conditions and constraints with the goal of maximizing the performance of the system. TO is different from shape optimization in the sense that the design can attain any shape within the design space, instead of dealing with predefined configurations. It involves the optimal distribution of material within the structure. It is used to find a preliminary structural configuration that meets predefined criteria. This type of optimization sometimes gives a design that can be completely new and innovative. Typically, the design process starts with a block of material called the design domain. The design domain is comprised of large number of candidate elements, and topology optimization process selectively removes the unnecessary elements from the domain.

Topology optimization is used by engineers at the concept level of the design process to arrive at a design proposal that is then fine tuned for performance and manufacturability. This replaces time consuming and costly design iterations while improving design performance. In some cases, results from a topology optimization, although optimized, may be expensive or infeasible to manufacture. These challenges can be overcome through the use of manufacturing constraints in the problem formulation. Using manufacturing constraints, the method yields engineering designs that would satisfy practical manufacturing requirements. In some cases results from topology optimization can be directly manufactured using additive manufacturing.

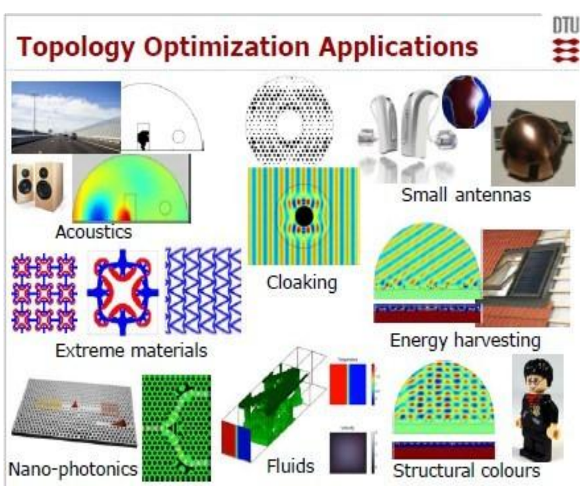

Fig. 1 Applications of Topology Optimization

II. Problem definition of the project:

1. Less resistance of the fuselage structure to withstand the loads acting on it.
2. Low stiffness of the fuselage which deforms quickly.
3. High volume of the fuselage structure which results in the use of extra material.
4. Increase in overall cost of the material of the fuselage.

III. Objective of the project:

1. To optimize the fuselage shape for weight reduction, minimizing the use of the material for the fuselage.
2. To get a structure of the fuselage that satisfies all the design constraints with minimum material.
3. To get a highly efficient product.
4. To get a higher quality product with overall lower development cost.
5. To get a fuselage structure capable of withstanding any higher loads.
6. To minimize the volume of the structure.
7. To get a fuselage structure of minimum weight and higher stiffness.

IV. Methodology.

Software used: CATIA software is used to model the fuselage and later ANSYS software is used for the analysis and optimization of the fuselage structure.

Parameters needed for the analysis:

1. Material of the fuselage: The material selected here for the analysis is aluminum alloy 7075 and the relevant material properties of AL 7075 can be used and the properties of the material are mentioned in the next chapter.

Material considered and the material properties: The material that was considered here was aluminum alloy 7075 and it was selected based on certain factors and was selected in ASME codes.

Factors considered for selecting the type of material:

- Young's modulus: A material with higher young's modulus was required because of its higher stiffness.
- Rigidity modulus: A material with higher rigidity modulus is required because of its resistance to shear loads.
- Density: A material with less density is always needed in aircraft applications because of its lower weight.
- Hardness: A material with higher hardness is required because of its resistance to surface indentation and cracks.
- Thermal resistance: A material with higher thermal resistance is required because at higher altitudes the temperature rises and the material may expand and that leads to failure, so if a material of higher thermal resistance is used the failure or the deformation of the material can be avoided.

So considering all these factors, according to ASME codes the aluminum alloy 7075 was selected which has got all the optimum material properties.

Material properties of aluminum alloy 7075:

- Density: 0.1015 lb/in³ or 2.81g/cc
- Ultimate tensile strength: 572 Mpa
- Tensile yield strength: 503 Mpa
- Modulus of elasticity: 71.3 Gpa
- Poisons ratio: 0.33
- Fatigue strength: 159 Mpa
- Shear modulus 26.9 Gpa
- Shear strength: 331 Mpa

2. Loads acting on the fuselage: The different loads that were considered were aerodynamic pressure loads, load due to self weight of the aircraft, shear load, bending moment load, loads due to the torque.

Loads considered and values of the different loads: the different loads that were considered to act on the fuselage are loads due to self weight, aerodynamic pressure loads acting as outside pressure loads and inner pressure loads, loads due to torque, loads due to bending moment and loads due to shear. The values and the calculation of the loads are mentioned below.

- Loads due to self weight: The self weight of the fuselage was considered as approximately 5 ton or 5000 kg and when converted to kilo Newton the value of the self weight was about 49.05 KN and it is considered to act on the lower part of the fuselage structure.
- Loads due to the aerodynamic pressure loads: The aerodynamic pressure loads acts as outside pressure and inner pressure forces on all the sides, i.e. on the upper and lower surfaces of the aircraft and also on the cross sectional areas and when it acts on the cross sections it will be different on the upper and lower cross sectional area as both the semi- circular cross sections are of different area.
- Inner pressure that was considered was atmospheric and the value was 101.325 Pascal and when converted to KN the values were same on the upper and lower inner surfaces of the fuselage and it changed on the cross sections. The inner pressure on the upper semi circle was 997. 27 KN and on the lower semi circle was 1124.47 KN and on the upper and lower surfaces it was 42.619 KN.
- Outside pressure force too was same on the upper and lower surfaces and varied on the cross sections and the values of the

outside pressure on the upper and lower surfaces were 18.096 Kpa and in KN the value was 761.2 KN. The outer pressure force on the upper semi circular cross section was 178.12 KN and on the lower semi circular cross section it was 200.83 KN.

- The calculation is shown here: Inside pressure force on the upper and lower surfaces = 101.325*42.0264 * 0.01 = 42.619 KN.
- On the upper semi circle cross sectional area the inner pressure force was 101.325* 9.8423 = 997.27 KN.
- On the lower semi circle cross sectional area the inner pressure force was 101.325*11.097 = 1124.47KN.
- On the upper semi circle cross sectional area the outer pressure force was 18.096 * 9.8426 = 178.12 KN.
- On the lower semi circle cross sectional area the outer pressure force was 18.096 * 11.097 = 200.83 KN.
- Load due to torque: $T = (\tau * J_P) / R$

$= [(331*10^6) * 19.60] / 1.8795$

$= 3451.76 * 10^6$ N-m.

- Load due to bending moment: $M_b = (E*I)/R$

$= (71.3*10^9*0.6126)/1.8795$

$= 371.76 * 10^9$ N-m.

- Shear load:

On the upper surface $= 331* 10^6* 9.8423$

$= 3257.8 * 10^6$ N.

On the lower surface $= 331* 10^6* 11.0977$

$= 3673.34 * 10^6$ N.

V. DESIGN OF FUSELAGE.

Model of the fuselage: The Fuselage model that is considered here in the project is BOEING 707-120B.

Dimensions of the fuselage:

- Length of the fuselage: 42.06 m (138 feet).
- Diameter of the upper semi circle: 3.54 m.
- Diameter of the lower semi circle: 3.759 m.
- Cross sectional area of upper part: 9.84 m^2.
- Cross sectional area of lower part: 11.097 m^2.

3D modeled view of the fuselage using CATIA

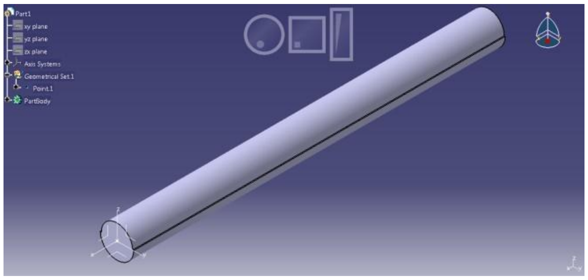

Fig. 2: 3D View in CATIA.

Orthographic views of the fuselage structure:

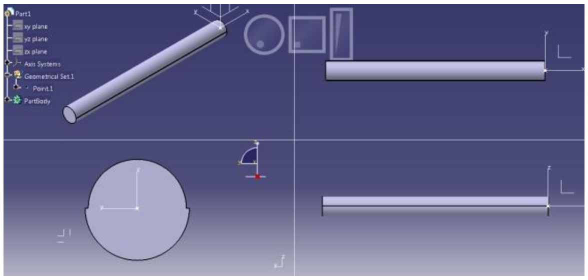

Fig: 3: Orthographic views of the fuselage.

VI. Results and Discussion.

Meshing: The meshing results and also the results for one iteration for 1mm thickness of the fuselage are given below and subsequent iterations will be continued.

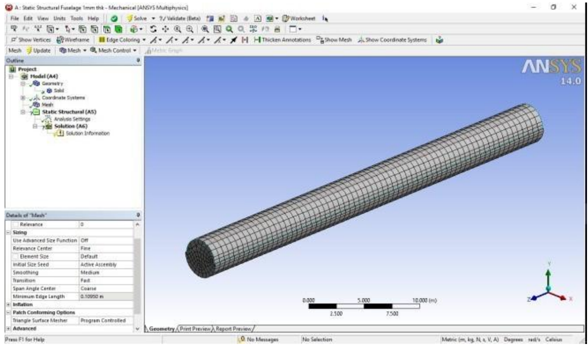

Fig: 4 mesh

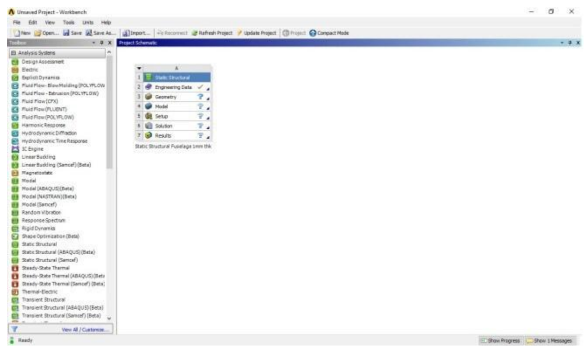

Fig. 5

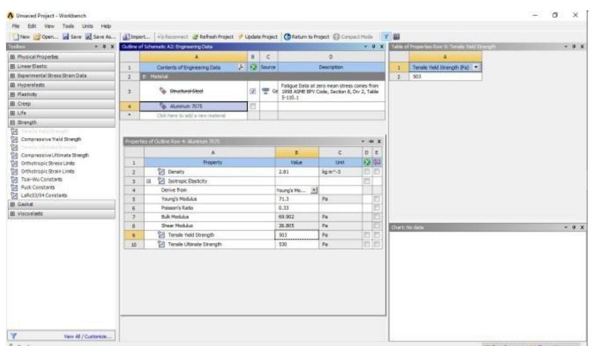

Fig: 6: material properties.

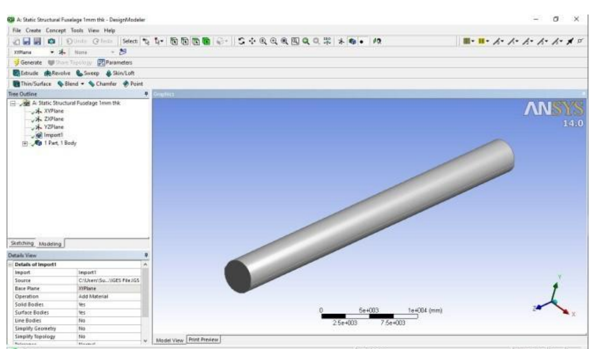

Fig: 7 Static fuselage 1mm thick.

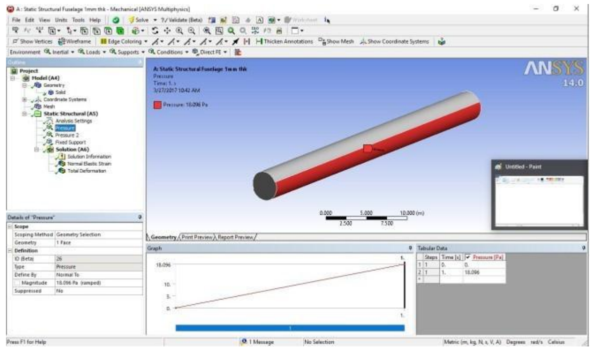

Fig: 8 Aerodynamic pressure loads.

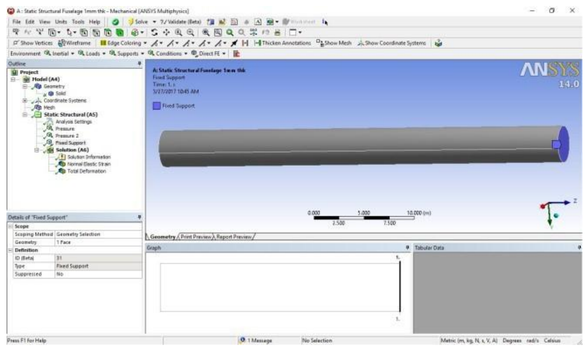

Fig: 9 constraints.

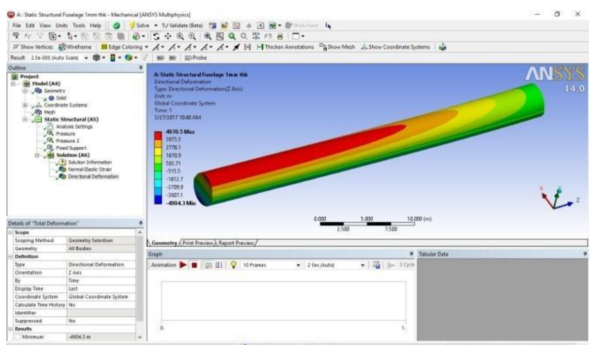

Fig: 10 deformation.

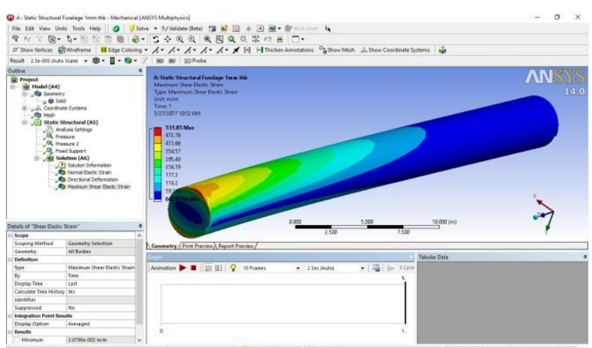

Fig: 11 maximum shear strain.

VII. Conclusion

The present project gives optimized results for the fuselage structure, first the analysis is carried out and the optimized regions of stress are determined and later the shape and size optimization is carried out and again a fuselage with an optimized size and shape is obtained.

IX. Future work

The present study focuses on only the central part of the fuselage but a future scope can be there where the entire aircraft can be optimized and even a higher efficient structure and power output can be produced.

Acknowledgment

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