STUDY OF OPTIMIZATION OF COMPOSITE STRUCTURES WITH RESPECT TO INDUSTRIAL APPLICATIONS

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***Abstract*— This paper analyses the existing methodology of employing optimization techniques for structural optimisation of laminated composites at part level (local optimization) and also the limitations of this part level structural optimisation approach in industrial applications. The deviations observed in optimisation problem definition due to implementation of part level optimized solution at the assembly level (global optimisation) are presented. Multi shell closed structure is considered to carry out the numerical experiments, due to the reason; most of the structures of aircraft wing, wind turbine blade and helicopter rotor blades exhibit this configuration. Part level optimization is carried out on a single plate element of a multi shell closed structure and the optimized plate element is replaced in the multi shell closed structure. The observations made after replacement of optimized plate element through performing the numerical experiments are presented and well discussed.**

***Keywords* — Optimization, Composite Structures, Multi shell closed structures**

I. INTRODUCTION

Most of the aerospace and wind mill manufacturing industries have made some rapid strides in the field of advanced composite materials technology [1,4,17,18]. This is especially, due to their high strength to weight ratio properties, which assumes paramount importance with the ever increasing demand to carry higher and higher payloads for aerospace vehicles, lighter and stiffer structure to generate more power in case of wind mills.

The laminated composite structures offer flexibilities to designers so as to optimize the structure for a specific or even multiple design criteria. Most of the composite parts are in the form of laminated structures, which usually comprises of multiple layers. Each layer is extremely thin in shape and may possess different fiber orientation. Two laminates may have the same number of layers and the same fiber orientation but they can exhibit different mechanical properties based on the stacking sequence of the laminate. For laminated composite structures, the appropriate stacking sequence selection is a key parameter to make effective use of the material properties.

Laminated composite parts are usually made of unidirectional plies of a given thickness and with fiber orientations. The design of composite laminates is often articulated as a continuous optimization problem with thicknesses and orientations of ply as design variables. However, for many practical problems, the ply thicknesses are maintained uniform. Moreover, ply angles are set to a discrete set of angles such as 00, +450,-450, and 900 for ease of ply cutting and ply placement on the structure with high orientation precision.

The problem of designing such laminates using conventional design approach is associated with weight penalty and time consuming. Composite structures were designed by using classical optimization methods and ply orientations are considered as the design variables for optimization of structure.

II. RELATED WORK

Aerospace industry is much attracted to the fibrous composites due to its higher strength and stiffness properties, though the materials are cost effective. Most of the aircraft structure configurations are plates, beams and stiffened plates. The optimization carried on these structural configurations for different failure modes are discussed here.

Implementation of genetic algorithms to optimize the laminate stacking sequence for maximum laminate strength and stiffness with minimum weight by considering ply orientations as design variables is presented by Kevin J. Callahan and George E Weeks[8]. R.Kathiravan and R.Ganguli[19] presented implementation of Particle Swarm Optimization algorithm to optimize the laminate stacking sequence of a box beam walls for maximum laminate strength with minimum weight by considering ply orientations as design variables.

Rectangular panels are the basic configurations of aircraft structures used for wide applications. Optimization of rectangular plate structures with all edges simply supported boundary condition under bi axial compressive loads for maximization of buckling load capacity is discussed using simulated annealing technique by Ozgur Erdal and Fazil O

Sonmez[16], using genetic algorithm method by G. Soremekun, Z. Gurdal, R.T. Haftka and L.T. Watson[6] and using differential evolution algorithm method by M V A Raju Bahubalendruni, Srinivasarao TV and Mantha VenkataRamana[11]. The analytical equations from exact solutions are used to compute the critical buckling load factor of a simply supported rectangular plate. Maenghyo Cho and Seung[12] Yun Rhee presented optimization of laminates using genetic algorithm with repair strategy for maximum inter laminar or in-plane shear strength with free edges under different loads such are extension, bending and twisting loads.

Stiffened panels are the most preferred configuration to carry combined axial and shear loads and offers good resistance in buckling mode of failure. Optimization of stiffened panels with simply supported boundary conditions to maximise the buckling load capacity without weight penalty by considering ply orientations of skin and stiffener is presented by Wei Wang, S.Guo, Nan Chang and Wei Yang[21]. J.Enrique Herencia, Raphael T Haftka[7] implemented linear approximation method to perform a piecewise layup sequence optimization on stiffener web and skin structure independently to reduce the mass of the structure by considering the ply orientations are design parameters.

Implementation of differential evolution algorithm to optimize laminated composite structures such as plates, beams of different cross sections, stiffened panels for bending, buckling and natural vibrations using ply orientations as design variables to minimize the weight is presented by Mantha VenkataRamana and M V A Raju Bahubalendruni[13].

The present work is focused on most preferable configuration of laminated composite structures used in aero structures and wind mill blade applications. A simplified multi-shell closed structure is prepared using CATIA and generated a coarse finite element model(FEM) using Hypermesh. Layered composite material properties, loads and boundary conditions is applied to perform the finite element analysis (FEA) in Nastran Solver. The element forces are extracted from the Nastran output file and then analyzed for different failures. The element is optimized for weight using differential evolutionary algorithm (DEA). The optimized element layup sequence is placed in the coarse finite element model and the FEA is performed to extract the internal element forces on the optimized element. The optimized element is then analyzed for different failures. The behaviour of optimized element is compared and the consequences due to replacement of optimized element are illustrated. The schematic representation of work flow is represented in Fig.1.

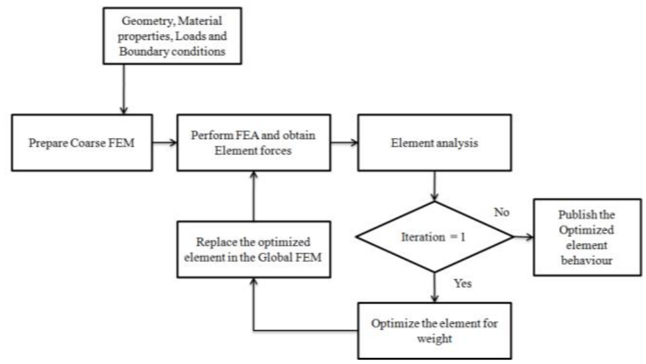

Fig.1. Schematic representation of work flow

III. PROBLEM FORMULATION

Most preferred configuration of aircraft wing and wind turbine blade is multi shell closed aerofoil structure. For minimalism, one end fixed multi shell structure (with 4 rib, 2 spar and 2 Skins shown in Figure.2) is subjected to distributed pressure load on the bottom skin is considered to perform the finite element analysis. For more simplification a coarse model mesh is generated and is used to obtain the internal forces acting on each element of the structure. Following are the material properties used for the structure.

Thickness of ply t= 0.1mm

Youngs Modulus of Elasticity along fiber direction E_1= 148 GPa,

Youngs Modulus of Elasticity in transverse to fiber direction E_2= 9.6 GPa,

Shear Modulus G_{12}= 4.8 GPa and

Poisson’s Ratio υ_{12}=0.31

In order to perform Finite Element Analysis, 3D closed shell structure is modelled and mesh is generated. CQUAD-4 element is chosen to model the rectangular geometry; MAT8 shell element is chosen considering the thickness of the element and defined orthotropic material properties. Layered composite element property is assigned by choosing PCOM P property [2,15].

The geometrical 3D multi shell closed structure is represented in Fig.2 with all dimensions and an exploded view of assembly is shown in Fig.3 to illustrate the internal detailed construction of ribs and spars.

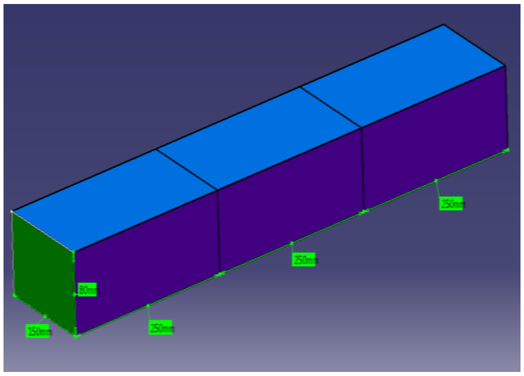

Fig.2. 3D closed shell geometrical model

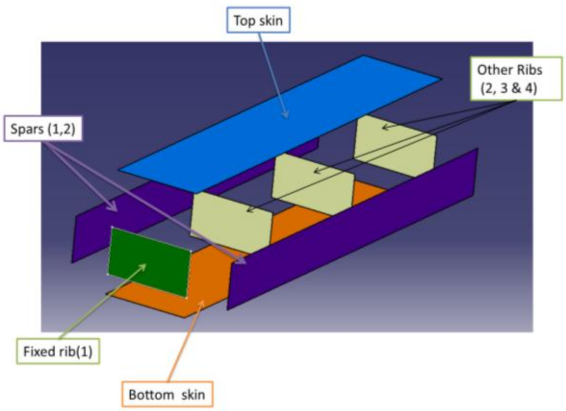

Fig.3. Exploded view of 3D closed shell geometrical model

From the CAD geometry, material properties, load data and boundary conditions data, pre-processor file is generated. Fig.4 represents the meshed model of 3D closed shell geometrical model.

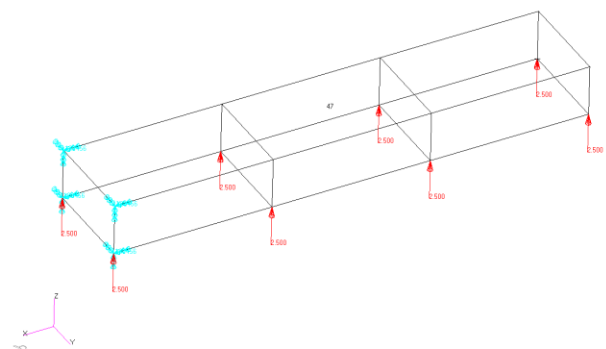

Fig.4. Meshed 3D closed Shell geometrical

A. Finite Element Analysis and Outcomes

A Nastran input file is shown in Figure.5 with element definition, element connectivity, material properties and layup sequence definition details. For static case, analysis is performed and the resulted internal forces acting on each element is captured through the Nastran output file shown in Figure.6.

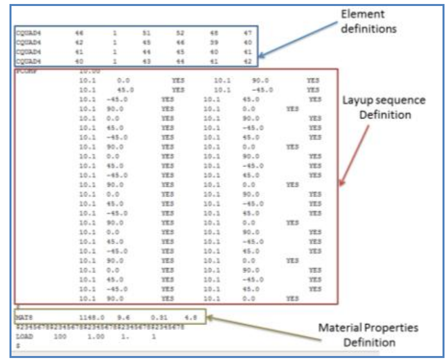

Fig.5. Nastran Input file format

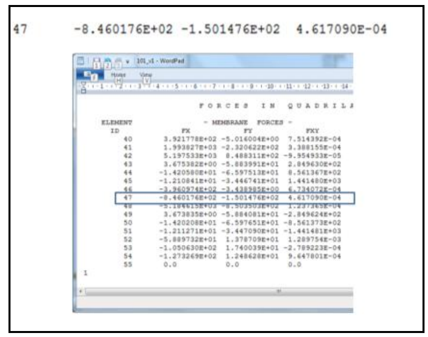

Fig.6. Internal forces on element 47 from Nastran output file

B. Element level Analysis

Due to the pressure load acting on the bottom face of the structure, the top skin elements subjected to compression which cause buckling of the skin elements as a major failure of the skin[5,14,20]. Hence the skin element must be designed for static and buckling strength.

Skin element-47 loading is schematically represented in Fig.7, Where (λ) is the load amplitude.

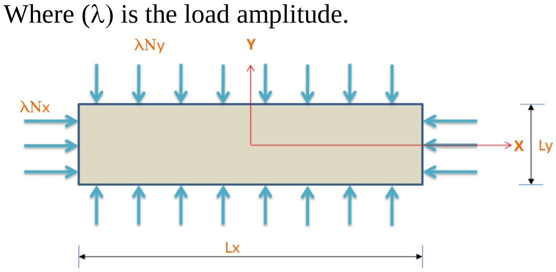

Fig.7 Skin element 47 loading

From Figure.2, the geometrical dimensions of rectangular panel are.

Length of the Plate in x, y directions (mm) Lx= 250; Ly=150

From Figure.6, the forces acting on Skin element 47 are

In-plane load along x, y directions (N) Nx = -846; Ny = -150

The critical buckling load factor due to the internal forces for skin element 47 is calculated from exact solution [3,9,10].

Critical buckling load factor $\lambda_{cb} = \min(\lambda_{ij})$ (1)

$$\lambda_{ij} = \frac{\pi^2\left[D_{11}\left(\frac{i}{Lx}\right)^4 + 2(D_{12}+2D_{66})\left(\frac{i}{Lx}\right)^2\left(\frac{j}{Ly}\right)^2 + D_{22}\left(\frac{j}{Ly}\right)^4\right]}{\left[Nx\left(\frac{i}{Lx}\right)^2 + Ny\left(\frac{j}{Ly}\right)^2\right]} \quad (2)$$

The buckling load factor (λ_{ij}) must be calculated for different sets of i and j, $(i,j=1,2,3,\ldots)$. The lowest resulting value of (λ_{ij}) is the value of interest. For simply supported orthotropic plates, the lowest buckling load corresponds to a mode that has a half wave in at least one direction (i.e. either i or j is equal to unity). Where [D] is laminate bending stiffness matrix and the elements are D_{11}, D_{12}, D_{22} and D_{66}, which depends on the laminate stacking sequence. The computation of [D] is as follows.

The stress strain relationship in the 1-2(local) co-ordinate system can be represented as

$$\{\sigma\} = [Q]\{\varepsilon\} \tag{3}$$

Expanding

$$\begin{Bmatrix} \sigma_1 \\ \sigma_2 \\ \tau_{12} \end{Bmatrix} = \begin{bmatrix} Q_{11} & Q_{12} & Q_{16} \\ Q_{12} & Q_{22} & Q_{26} \\ Q_{16} & Q_{26} & Q_{66} \end{bmatrix} \begin{Bmatrix} \varepsilon_1 \\ \varepsilon_2 \\ \gamma_{12} \end{Bmatrix} \tag{4}$$

Where

$\sigma_1, \sigma_2, \tau_{12}$ Stresses in 2D local co-ordinate system

$\varepsilon_1, \varepsilon_2, \gamma_{12}$ Strains in 2D local co-ordinate system

$[Q]$ Lamina stiffness matrix in 2D local co-ordinate system

The stiffness matrix in the x, y co-ordinate system depends upon the orientation of the layer.

Hence

$$[\bar{Q}] = [T_\sigma]^{-1}[Q][T_\varepsilon] \tag{5}$$

$$\begin{bmatrix} \bar{Q}_{11} & \bar{Q}_{12} & \bar{Q}_{16} \\ \bar{Q}_{12} & \bar{Q}_{22} & \bar{Q}_{26} \\ \bar{Q}_{16} & \bar{Q}_{26} & \bar{Q}_{66} \end{bmatrix} = [T_\sigma]^{-1} \begin{bmatrix} Q_{11} & Q_{12} & Q_{16} \\ Q_{12} & Q_{22} & Q_{26} \\ Q_{16} & Q_{26} & Q_{66} \end{bmatrix} [T_\varepsilon] \tag{6}$$

$$[T_\sigma] = \begin{bmatrix} \cos^2\theta & \sin^2\theta & \sin 2\theta \\ \sin^2\theta & \cos^2\theta & -\sin 2\theta \\ -\sin\theta\cos\theta & \sin\theta\cos\theta & \cos 2\theta \end{bmatrix} \tag{7}$$

And

$$[T_\varepsilon] = \begin{bmatrix} \cos^2\theta & \sin^2\theta & \sin\theta\cos\theta \\ \sin^2\theta & \cos^2\theta & -\sin\theta\cos\theta \\ -\sin 2\theta & \sin 2\theta & \cos 2\theta \end{bmatrix} \tag{8}$$

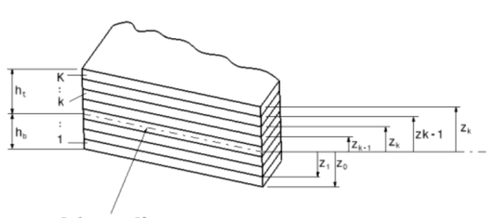

Fig.8. Schematic representation of laminate stack-up

z_k is position of k[th] layer (thickness) from the reference plane in z direction.

h_t is the distance from the laminate reference plane to the top of the ply.

h_b is the distance from the laminate reference plane to the bottom of the ply.

Global bending stiffness matrix can be written for the laminate stack-up shown in Fig.8.

$$[D] = \int_{-h_b}^{h_t} z^2 [\bar{Q}]\, dz \tag{9}$$

$$D_{ij} = \int_{-h_b}^{h_t} z^2\, \bar{Q}_{ij}\, dz \quad (\text{ for i,j = 1,2,6}) \tag{10}$$

The equation can be written as

$$D_{ij} = \frac{1}{3}\sum_{k=1}^{K} \left(\bar{Q}_{ij}\right)_k \left(z_k^3 - z_{k-1}^3\right) \tag{11}$$

In the current problem, the curvatures at reference plane (k_x, k_y, k_{xy}) are zero and shear force is negligible. Since the layup of the plate is considered as symmetrical, in-plane out of plane coupling matrix [B] = [0]. The laminate mid- plane strains can be obtained by multiplying the global compliance matrix with the load tensor.

The global compliance matrix is $[a]=[A]^{-1}$, where [A] is laminate extension stiffness matrix.

$$\begin{Bmatrix} \varepsilon_x^0 \\ \varepsilon_y^0 \\ \gamma_{xy}^0 \end{Bmatrix} = \begin{bmatrix} a_{11} & a_{12} & a_{16} \\ a_{12} & a_{22} & a_{26} \\ a_{16} & a_{26} & a_{66} \end{bmatrix} \begin{Bmatrix} \lambda Nx \\ \lambda Ny \\ 0 \end{Bmatrix} \tag{12}$$

The ply level strains in 2D global coordinate system will be same as the mid-plane strains since there curvature values are zero. Ply level strains in 2D local coordinate system are calculated by using the transformation matrix and ply level stresses are calculated from equation (4). The reserve factors for each ply at ply-top and ply-bottom are calculated using the below equation (13).

$$R_f = \frac{1}{\sqrt{\left(\frac{\sigma_1^2}{X^2} - \frac{\sigma_1\sigma_2}{X^2} + \frac{\sigma_2^2}{Y^2} + \frac{\tau_{12}^2}{S^2}\right)}} \tag{13}$$

If $\sigma_1 > 0$, then $X=X_t$, If $\sigma_1 < 0$, then $X=-X_c$,

If $\sigma_2 > 0$, then $Y=Y_t$, If $\sigma_2 < 0$, then $Y=-Y_c$,

Where

Failure strength along longitudinal direction in tension and compression (X_t X_c)

Failure strength along lateral direction in tension and compression (Y_t Y_c)

Failure strength in shear(S).

The laminate bending stiffness matrix elements for $\{[0,90,45,-45]_S\}_6$ are mentioned in below Table.I

585315.424	178073.184	835.6088	(N-mm)
178073.184	575288.119	835.6088	
835.6088	835.6088	194711.1307	

Table.I bending stiffness matrix elements

Substituting the in-plane forces and stiffness matrix elements in equation (1) and (2) results

$$\lambda_{cb} = 1.02$$

For the loads *Nx*=-846N and *Ny*=-150N, the laminate results critical reserve factor using Tsai-Hill failure Criteria R_f = 1.574. The structure is safe from static failure and buckling failure mode.

IV. The Optimization Problem

Design objective of optimization problem is to minimize the weight of the part, subjected to design constrains as performance of the structure (maximum allowable displacement, critical reserve factor, buckling load capacity, etc.,) based on the application of the structure, where ply orientations are the design variables under pre assigned conditions (Such are loads data, material data and Geometry other than the thickness of the laminate).

The optimization problem for the plate element is formulated with the following design objective, constraints, variables and pre-assigned conditions.

Design Objective: To minimize the weight of the rectangular plate.

Design Constraint: Critical buckling load factor (λ_{cb}) ≥ 1.0, Critical Reserve factor(R_f) ≥ 1.5

Design variables: ply orientations $\theta_i \in (0^0,45^0,90^0,-45^0)$, i∈ (1,2,3,4…47)

Pre assigned Conditions:

Geometry:

Lx= 250mm; *Ly*=150mm ;(thickness of plate= 4.8mm to be optimized)

Loads:

Nx=-846N; *Ny*=-150 N (compressive loads)

Ply properties:

Thickness = 0.1 mm E_x= 148 GPa, E_y= 9.6 GPa, G_{xy}= 4.8 GPa and υ_{xy}=0.31.

A. Implementation of DEA

Differential Evolution Algorithm [22] is employed to find out the optimized layup sequence for minimum weight. Due to the stochastic nature of algorithm, the algorithm is implemented for 25 times for different combinations of generations, population, cross-over rate and mutation factor. The most repeated feasible solution is presented. The schematic representation of Implementation of DEA is presented through flow chart shown in Figure.9.

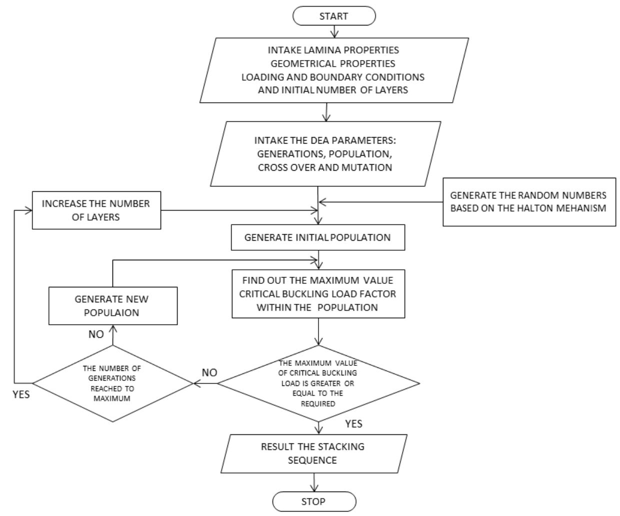

Fig.9. Flow chart of DEA implementation

B. Results

For the optimized layup sequence [$\pm45_5$, 90_3,0_4, $\pm45_2$, 0_2]s; the laminate bending stiffness matrix elements calculated using equation(11) are mentioned in below Table.II.

398091.5956	250580.3041	13648.2767	(N-mm)
250580.3041	435693.9907	13648.2767	
13648.2767	13648.2767	265223.9594	

Table.II. Bending stiffness matrix elements for optimized layup sequence.

Using equation (1) and (2) , the critical buckling load factor and critical reserve factor using Tsai-Hill failure criteria for optimized layup sequence are calculated and presented in Table.III.

	Thickness of Plate(mm)	Nx (N)	Ny (N)	layup sequence	Critical Buckling load factor (λ_{cb})	Critical Reserve Factor
Initial Design	4.8	846	150	{[0,90,45,-45]$_s$}$_6$	1.02	1.574
Optimized Design	4.6	846	150	[$\pm45_5$, 90_3,0_4, $\pm45_2$, 0_2]$_s$	1.013	1.615

Table.III Initial and Optimized design results

V. Implementation of Optimized Element in FEM

Now the optimized layup sequence is assigned to skin element (47) as shown in Figure.10a. Expanded layup sequence used for the Skin element 47 is represented in Figure.10b.

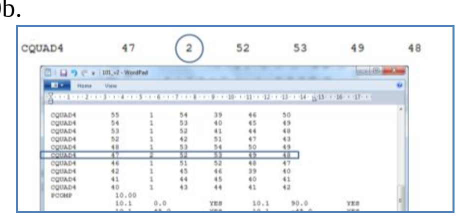

Fig.10a. Modified Nastran Input file - Element representation

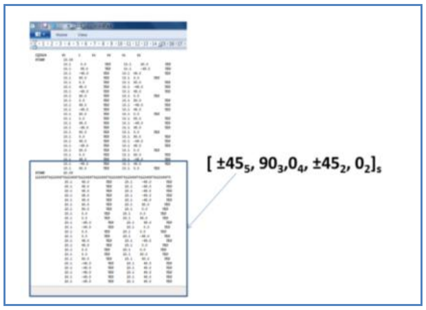

Fig. 10b. Modified Nastran Input file – Layup representation

Finite Element Analysis is performed and the resulted input forces from the Nastran output file are shown in Fig.11.

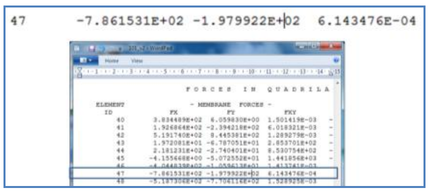

Fig. 11 Internal forces for the Skin Element 47

A. *Optimized Element Analysis*

Because For the obtained internal forces (*Nx*=-786N and *Ny*=-198N), The critical buckling load factor for the skin element (47) is calculated from equations(1),(2)

λ_{cb} = 0.957 and the critical reserve factor =1.759

The above indicates that the structure fails due to buckling, though it offers good static strength.

VI. CONCLUSIONS

In an assembled product, the distribution of loads on each part is dependent on the stiffness of the part. The forces acting on each part can be obtained by performing the finite element analysis with suitable assumptions in meshing the assembled product. Though the element size has great impact on the analysis results, due to the large number of parts in assemblies like aircraft structural assembly, performing analysis with fine mesh leads lots of computational time, thus coarse model mesh is used for extracting the internal forces in each element and is analyzed for all possible failure cases. Typically stiffened panels and plate elements are considered for this case, the optimization of such plate elements at local level and implementing the optimized plate element layup sequence at assembled product is studied.

From the results obtained for the closed shell structure at global level with initial and optimized layup sequence for skin element-47, it is understood that that part level optimization of structure does not fulfill requirements at global level. Though the observation results are most occurred in local vs global optimization problems, the replacement of the part level optimization results at global level affects the pre-assigned parameters, for which the optimized layup sequence may not fulfill the required design objective. The change in pre-assigned parameters due to the layup sequence change must be considered while performing optimization and also part level optimization results cannot be suggested for the industrial applications as well.

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