

# Optimum stacking in a multi-ply laminate used for the skin of adaptive wings

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This paper is proposing a method for determining the optimum stacking in a composite laminate material using finite elements method program for a specific application (the skin panel used for the adaptive wing). The main area of application for this method refers to the wings of the adaptive aircrafts, where the material of the wing's skin is requested to provide good strength while allowing large deformations. This optimization can lead to the reduction of the technological and operational costs. The analysis and result interpretation can then be extrapolated to similar materials with bigger plies stacking.

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## 1. Introduction

The main objective of this paper is to provide a method to determine the optimum stacking in a composite laminate material using finite elements method program.

The main area of application for this method refers to the wings of the adaptive aircrafts, where the material of the wing's skin is requested to provide good strength while allowing large deformations.

Other areas of application can take advantage as well from this method: automobiles, boats hull, aeolian turbines blades, etc.

For this paper is used a 5 ply stacking. Each ply can be made of CFRP (Carbon Fiber Reinforced Polymer) tape or fabric, leading to 32 possible combinations.

The scope is to find the corresponding optimum stacking that presents the best deformation to stress ratio, at the same loading and with the same boundary conditions.

This optimization can lead to the reduction of the technological and operational costs.

## 2. Adaptive wings

It can be said that aviation began with the glided flight done by the Wright brothers Flyer 1 airplane in 1903. The plane without fuselage was a biplane with textile fabric wings on a wooden frame. The command surfaces were represented by the elastic deformation of the wing tips.

With the increasing weight and performance of airplanes, technology has leaped to the rigid wing, with ailerons and hyper-sustentation devices. The wing is used to obtain the necessary lift force.

The development of aviation led to a multitude of wing shapes, because of the justified trend for improving the aerodynamic qualities of the aircraft. The increasing flight speeds leads to substantial changes in the sizing, design and configuration of the wing [1, 2].

The rigid wing geometry is optimized just for one stage of the flight, usually for the stage with the longest time span (cruising). This optimization doesn't take into account the local atmospheric parameters (pressure, wind speed). Small variations of the geometry and the aerodynamic coefficients can be obtained using flaps. This concept leads to increased fuel consumption and costs, reduced lift and increased drag.

In order to reduce the costs and the fuel consumption, the wing must change its geometry for each of the flight stages and for all atmospheric modifications.

Beside the classical wing (with ailerons and flaps), there were a few proposals for the wing that would allow geometry modifications during flight [1]. Here can be noted the aircraft wing with variable arrow, the tilt wing aircraft and the telescopic wing aircraft. These concepts made possible a better flight control at different speeds and atmospheric conditions, but lead to an increase in the technological and mechanical complexity.

In the last years, new concepts and proposals arise: the adaptive wings, the morphing wings [3], the use of intelligent materials [4], etc. These structures can reconfigure shapes, stiffness or aircraft commands to increase performance.

They are included in the ACARE taxonomy in the category “Emerging” and are expected to be built before 2020. It is expected that the first adaptive aircrafts will enter into operation in 2040.

### 3. Proposed adaptive wing structure

The proposed wing for which is calculated the optimum stacking for the skin is an adaptive wing [5] and its internal structure is presented in Figs. 1 and 2.

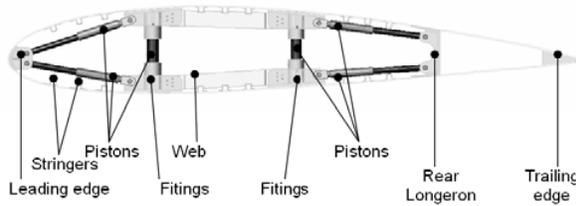


Fig. 1. Proposed adaptive wing internal mechanism.



Fig. 2. Proposed adaptive wing section.

The longerons are replaced with a series of pistons and are linked to the ribs by attachment fittings. The rib is replaced by a series of pistons and a web that is stiffening the structure. The components of the wing's structure can be observed in Figs. 1 and 2.

The longitudinal stiffness of the wing is assured by the flanges and web of the longerons, the stringers, the leading edge and the trailing edge. The lateral stability of the wing is assured by the fittings, partial web and the pistons.

The pressure in the pistons is commanded by an electro-pump device. The increasing and decreasing of the fluid pressure will generate the modification of the position of the piston, therefore the different dimensions of the wing.

The variation of the dimensions of the profile lead to the optimum airfoil needed for the current and local flight parameters [5]. The proposal of the presented wing structure can reduce aircraft drag and in flight fuel consumption by 10%, leading to smaller costs.

### 4. Laminate skin stacking definition

For the optimal definition of the CFRP laminate material used for the adaptive wing's skin, FEM analysis are made on a rectangular plate that provide sufficient similarities with an actual wing skin plate (from leading edge to trailing edge, in between 2 ribs).

The laminate stacking is made from the combination of two kinds of materials:

- Tape - pre-impregnated unidirectional CFRP;
- Fabric - pre-impregnated bidirectional CFRP.

The stacking is made from 5 plies, by combining the two above defined materials. Tape material is applied only with the fibers along the length of the plate, while the fabric is applied at 45 degrees relative to the tape direction.

The total possible combinations for 5 plies of the above mentioned two materials are 32 combinations. For the combinations of a different number of materials and plies, specific algorithms can be used.

In Table 1 are presented all possible 5 plies combinations of Tape and Fabric materials. Tape is noted with T and fabric is noted with F, while the combination is noted with the letter "L" followed by the combination number (from 1 to 32).

Table 1. All 5 ply combination of Tape and Fabric.

Ply	1	2	3	4	5
L1	T	T	T	T	T
L2	T	T	T	T	F
L3	T	T	T	F	T
L4	T	T	F	T	T
L5	T	F	T	T	T
L6	F	T	T	T	T
L7	T	T	T	F	F
L8	T	T	F	T	F
L9	T	F	T	T	F
L10	F	T	T	T	F
L11	T	T	F	F	T
L12	T	F	T	F	T
L13	F	T	T	F	T
L14	T	F	F	T	T
L15	F	T	F	T	T
L16	F	F	T	T	T
L17	F	F	F	F	F
L18	F	F	F	F	T
L19	F	F	F	T	F
L20	F	F	T	F	F
L21	F	T	F	F	F
L22	T	F	F	F	F
L23	F	F	F	T	T
L24	F	F	T	F	T
L25	F	T	F	F	T
L26	T	F	F	F	T
L27	F	F	T	T	F
L28	F	T	F	T	F
L29	T	F	F	T	F
L30	F	T	T	F	F
L31	T	F	T	F	F
L32	T	T	F	F	F

Note that combinations L1, L4, L10, L12, L17, L20, L26 and L28 are symmetric relative to the neutral axis, while the rest aren't.

Note also that some combinations are mirrored of other: L2 is mirror to L6, L3 is mirror to L5, L18 is mirror to L22, and so on.

### 5. FE Modeling and analysis

The geometric model of the skin plate with size 1000 x 500 mm is created in MSC Patran preprocessor. The geometric model is then meshed with elements SHELL type CQUAD (4 sides, 4 nodes) resulting a total elements number of  $40 \times 20 = 800$  elements using IsoMesh method. Fig. 3 presents the elements of meshed plate.

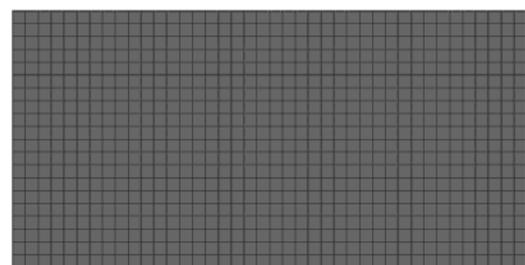


Fig. 3. Meshing of the skin plate.

The boundary conditions simulate a typical test for a plate bending.

The plate is considered clamped at both ends, blocking the movements on the X, Y, Z axis and rotations about Y axis on all end nodes. The rotations about X and Z are blocked as a result of the internal constraint in between the end nodes.

The applied load in one piston is extrapolated using the algorithm given in [6].

A force of 200 N is distributed uniformly on all 20 nodes on the entire width at the median transverse axis of the plate, as presented in Fig. 4.

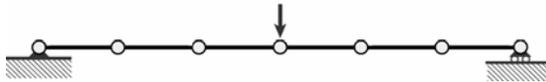


Fig. 4. Applied load and boundary conditions.

Hereafter, Table 2 presents the material properties for tape and fabric material, as given by the producer.

Table 2. CFRP Tape and Fabric properties MSC Laminate Modeler (module of MSC Patran) is used to define laminate composites.

		Tape	Fabric
Ply thickness	[mm]	0.25	0.25
Elastic Modulus 11	[MPa]	135000	60000
Elastic Modulus 22	[MPa]	8500	60000
Poisson Ration 12	[-]	0.35	0.05
Shear Modulus 12	[MPa]	4200	4200
Density	[kg/m <sup>3</sup> ]	1580	1550
Tension Stress Limit 11	[MPa]	1768	500
Tension Stress Limit 22	[MPa]	63	500
Compr. Stress Limit 11	[MPa]	823	480
Compr. Stress Limit 22	[MPa]	147	480
Shear Stress Limit	[MPa]	55	65
Bend. Shear Stress Limit	[MPa]	38	38

The composite material is created based on previously defined materials (tape and fabric). Material characteristics are introduced in LM\_Materials menu (layer thickness 0.25mm).

Composite material layers are created by using LM\_Ply menu.

LM\_Layup menu is used to define the order of the composite layup for the previously defined ply. Automatically are created groups, materials and properties.

The procedure is repeated to create all 32 layups which results by combination.

The job to be analyzed is defined in the preprocessor MSC Patran Analysis menu.

Analysis is set as a linear static analysis type [7].

## 6. Results and interpretation

Results are accessed with the post-processor MSC Patran and displayed using the Results menu [7]. The maximum deformation and maximum von Mises stress for the 32 layups previously created are checked.

Table 3 presents the results for each layup, giving the maximum von Mises stress ( $\sigma$ ) in between all plies, the maximum deformation ( $\Delta$ ) and the corresponding ratio ( $\Delta/\sigma$ ).

Table 3. Results for each layup.

	$\sigma$	$\Delta$	$\Delta/\sigma$
	[MPa]	[mm]	
L01	171	108	0.632
L02	232	166	0.716
L03	198	118	0.596
L04	172	108	0.628
L05	198	118	0.596
L06	232	166	0.716
L07	286	236	0.825
L08	230	172	0.748
L09	260	177	0.681
L10	215	271	1.260
L11	204	119	0.583
L12	201	127	0.632
L13	260	177	0.681
L14	204	119	0.583
L15	230	172	0.748
L16	286	236	0.825
L17	156	480	3.077
L18	316	280	0.886
L19	226	400	1.770
L20	152	467	3.072
L21	226	400	1.770
L22	316	280	0.886
L23	301	275	0.914
L24	307	243	0.792
L25	258	181	0.702
L26	203	128	0.631
L27	236	373	1.581
L28	218	275	1.261
L29	258	181	0.702
L30	236	373	1.581
L31	307	243	0.792
L32	301	275	0.914

Analyzing the above table it can be seen that the best ratio is given by layup L17 (5 fabric plies), with a ratio of 3.077, followed by L20 (symmetric 4 fabric plies with 1 tape ply in between) with a ratio of 3.072.

By far, for this application (skin panel of adaptive wing) the laminates based on more fabric plies is better than the laminates based on more tape plies. This can be seen in Table 4.

Table 4. Tape vs. Fabric - Number of plies.

Ply	1	2	3	4	5		$\sigma$	$\Delta$	$\Delta/\sigma$	
L1	T	T	T	T	T		L01	171	108	0.632
L17	F	F	F	F	F		L17	156	480	3.077
Ply	1	2	3	4	5		$\sigma$	$\Delta$	$\Delta/\sigma$	
L4	T	T	F	T	T		L04	172	108	0.628
L20	F	F	T	F	F		L20	152	467	3.072

The position of the ply in the stacking has a great influence in the overall results. The fabric ply is working better at the outside of the stacking, while the tape is working better closer to the neutral axis. This can be seen in Table 5.

Table 5. Tape vs. Fabric - Position of plies.

Ply	1	2	3	4	5		$\sigma$	$\Delta$	$\Delta/\sigma$
L10	F	T	T	T	F	L10	215	271	1.260
L26	T	F	F	F	T	L26	203	128	0.631
Ply	1	2	3	4	5		$\sigma$	$\Delta$	$\Delta/\sigma$
L12	T	F	T	F	T	L12	201	127	0.632
L28	F	T	F	T	F	L28	218	275	1.261

The mirrored stacking gives the same results for this kind of loading, therefore is no difference if the laminate is applied exactly opposite, as can be seen in Table 6.

Table 6. Mirror stacking - Position of plies

Ply	1	2	3	4	5		$\sigma$	$\Delta$	$\Delta/\sigma$
L27	F	F	T	T	F	L27	236	373	1.581
L30	F	T	T	F	F	L30	236	373	1.581

## 7. Conclusions

The presented method, using a FEM program, gives a general overview in order to understand that by using specific software for the engineering field helps significantly. The use of software is essential in the evolution of engineering.

The aircraft industry must offer performance and trust at the highest level for aircraft structures, this is the reason for using composite materials on a greater percentage of aircraft components. The use of software aims to reduce costs in design, achievement and development. This is the reason why a theoretical analysis of finite elements is very useful previous to tests on samples.

The method presented in this paper gives the possibility to quickly asses by FE model analysis the best way to find the optimum stacking for a specific application (in the presented case the application refers to the skin of an adaptive wing, that need to have good strength capability in the same time with a great deformation). The function for optimization was used as the ratio between the deformation and stress.

Analyzing the results of the analysis, conclusions can be drawn regarding the optimal way to arrange the 2 chosen CFRP pre-impregnated materials in the stacking.

For the current application, the best material to use is fabric, disposed at 45 degrees.

These results can be extrapolated to stackings with more plies, keeping in mind that fabric plies are more effective at the exterior and the tape plies are more effective at the interior. This will lead to lesser models to analyze and the possibility to exclude from the start the worst cases.

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