Wing Structural Analysis in Experimental Pulsejet Powered UAV

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This assignment is a part of larger project in Students Propulsion Association from Faculty of Power and Aeronautical Engineering at Warsaw University of Technology. Our intent is to build an airplane and test performances of its engine in a real flight.
Introduction contains basics such as: pulsejet engine conception, how to approximate load on wing and elements from the laminates theory. The majority of work is focused on wing strenght that includes composite covering. Analysis was done for two different cases – amount of plies and their order. The objectives are: finding sufficient layup in laminate and simulate if the structures will not damage in assumed conditions. Calculations were made based on finite element method, using ANSYS software. The last paragraph includes validation of gathered data and presentation of future plans.

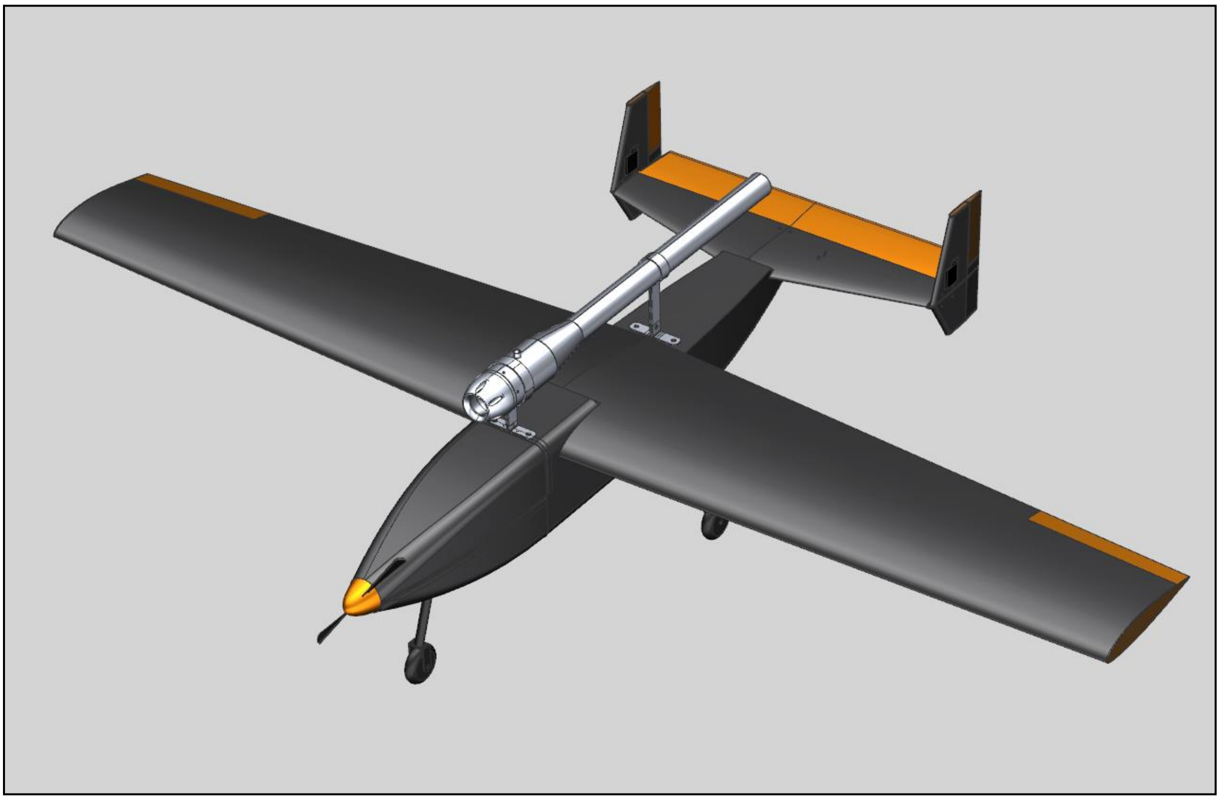

Figure 1. Model of plane created using NX software. Project: K. Pobikrowska.

I. Introduction

A. Pulsejet engine

The idea of building such a propulsion came out from its simplicity. Those were designed in order to provide light weight, ease of assembly and maintenance. Our inspiration to build it was Bruce Simpson, also known as “XJET” who has a passion for pulsejets and applies them onto small radio-controlled aircrafts.

In general, an oxidizer enters through the air inlet and combines with injected fuel. Owing to spark plug mix is heating in combustion chamber, simultaneously with increase of pressure. Exerted force is closing the valve, so gas can move only to the air outlet - which is giving desirable thrust. Adequate length of the tube is a result of ability to create negative pressure in combustion chamber that causes recur of fuel inject process. Also, the whole engine is fasten into a platform.

As any machine, those engines have defects: loud operation (which is caused by vibrations) and relatively few thrust. There were applications in planes, although pulsejets gave way to engines with better performances. Most known application is V-1 “flying bomb”.

Our pulsejet was built a few years ago and from the start target was to test it in flight. Currently it is being tested on a platform and optimized to aim thrust roughly 35 N.

Figure 2. Pulsejet engine built by members of Association.[1]

B. Wings

As is known, their task is to create enough lift to overcome forces of gravity and let the plane fly. Real loads that effects plane are heterogeneous due to the influence of aerodynamic and mass forces. Pressure differences causes lift formation.

In this work I will use Schrenk approximation to calculate pressures along the wingspan. Schrenk method relies on the fact that lift distribution is an average value from elliptic and “any contour” wing. It is also explained on Fig.3. We divide wing into pieces to receive distribution of lift coefficient along wingspan, assuming that aircraft lift coefficient is equal 1.

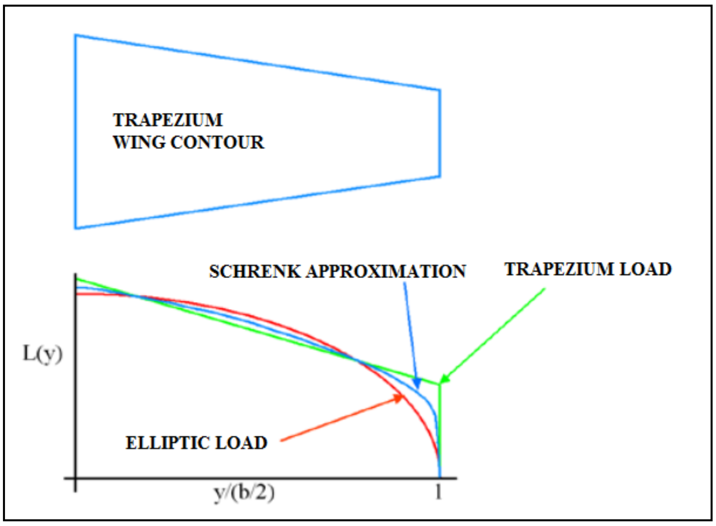

Figure 3. Schrenk approximation. Parameters: y – variable of wingspan; b – wingspan; L(y) – lift along wingspan.[2]

Mathematical formulas:

$$c^E(y) = \frac{4*S}{\pi*b} * \sqrt{1 - (\frac{2*y}{b})^2} \text{ - elliptic load} \quad (1)$$

$$c^T(y) = \frac{2*S}{b*(1+\lambda)} * [1 - \frac{2*y}{b} * (1-\lambda)] \text{ – trapezium load} \quad (2)$$

$$Cz(y) = \frac{1}{2} * [1 + \frac{c^E(y)}{c^T(y)}] \text{ – Schrenk approximation} \quad (3)$$

Then, to calculate pressure load:

$$p\,(y) = \frac{L}{S} = \frac{\frac{1}{2}*\rho*V^2*S*Cz}{S} = \frac{1}{2} * \rho * V^2 * Cz(y) \quad (4)$$

Where:

p – pressure [Pa];	L – lift [N];	ρ – air density $[\frac{kg}{m^3}]$;
V - cruising speed [m/s];	S – wing area [m^2];	Cz – lift coefficient [-];
b - wingspan [m];	λ – taper [-];	y – variable of wingspan [m];

C. Laminates theory

Assumptions:

- Plate is composed from orthotropic material layers bonded together
- Thickness is much smaller than other dimensions
- Displacements and strains are small
- Transverse shear and normal strains are negligible
- Each ply obeys Hooke’s law

Technology of laminates is based on making layup from layers that have different fiber directions. Their strength is the highest alongside direction 1 (Fig.4). That is why we place them under various angles – to receive possibly the best strength of covering. Most common angles are 0, 45, -45, 90 degrees as they are the easiest to realize in hand layup. Of course in large aviation industry few elements are made using this technology - as it is less effective and time-consuming.

Purpose of calculations in this theory is to estimate failure criteria. There are many methods to do it, such as: maximum stress, maximum strain, Tsai-Wu, Tsai-Hill etc. Ansys Composite Postprocessor allows to use all of them in one solution, showing which criteria proven lowest failure value in element. It is shown by the theory code and number of layer in which factor appeared. For example:

- e1t (2) - failure in 2nd layer and 1st direction, calculated by maximum stress theory;
- th (3) – failure in 3rd layer, calculated by Tsai-Hill theory;

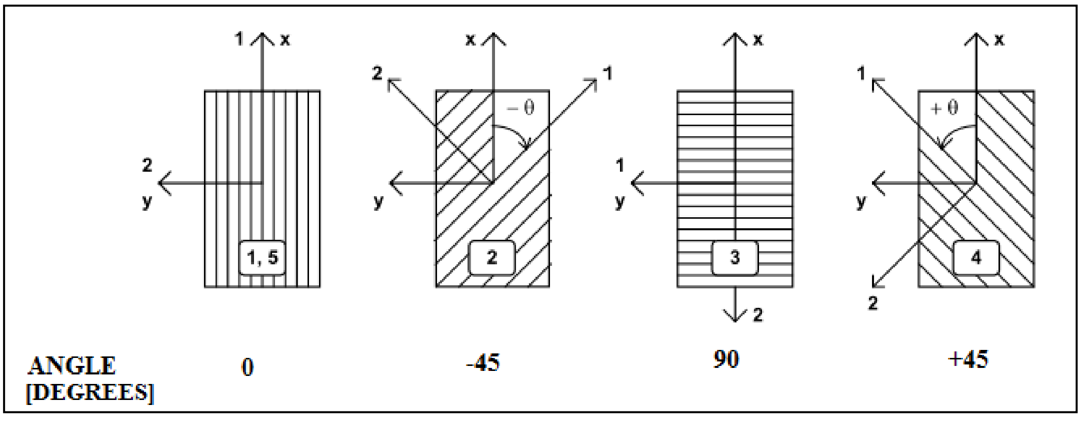

Figure 4. Different orientations that fibers might have.[3]

II. Analysis

A. Description of wing model

Model contains 3 groups of elements, divided by materials:

1. Wood – as wing primary construction, which includes:
    - Balsa wood - main spar (1) , lath (2), ribs (selected with green color).
    - Pine wood – blocks (3-5) and pin (6) which are fasten to fuselage.

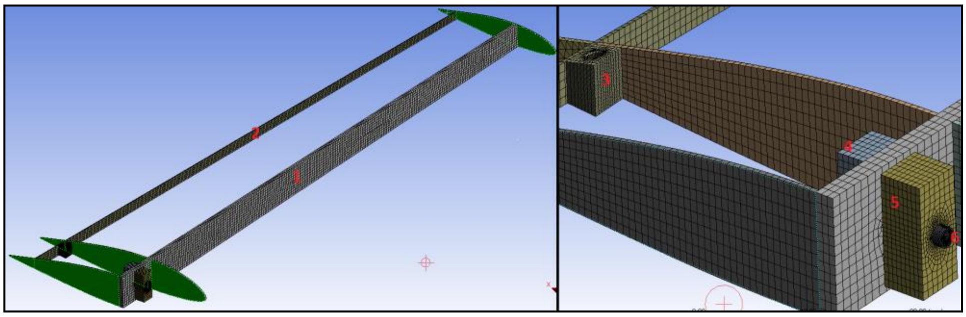

Figure 5. Structural wooden elements.

2. Styrofoam – as wing core – only to simplify process of building a real model (Fig.6A).
3. Composite covering – with separated top and bottom strap of main spar (7, Fig.6B).

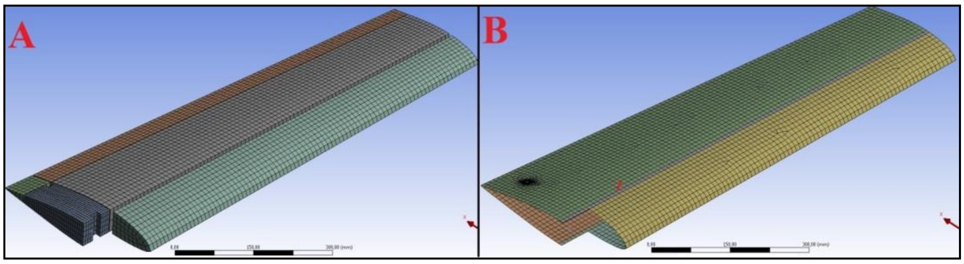

Figure 6. A) Styrofoam elements. B) Composite covering.

By means of ANSYS Composite Preprocessor I have created different stackups (set of plies with assigned fiber angles) to analyze them in two cases:

- alternative number of layers
- equal number of layers, but with different order

As we know, main spar carries more load and so does its strap – that is why it always has one extra layer. Layup sequence in Table 1 is top-down – first ply is the external one.

Main guidelines that can be found on the internet and in books are not suitable for RC models - such as laminate's middle surface symmetry or appropriate percentage of plies with specific orientation.

Stackup number	**Main covering**	**Straps**
1st case: alternative number of layers		
1	0, 90, 45, -45	0, 90, 45, -45, 0
2	0, 90, 45, -45, 90, 0	0, 90, 45, -45, 90, 0, 0
3	0, 90, 45, -45, 90, 0, -45, 45	0, 90, 45, -45, 90, 0, -45, 45, 0
2nd case: equal number of layers, but with different order		
1	0, 90, 0, 90, 45, -45	0, 90, 0, 90, 45, -45, 0
2	0, 90, 45, -45, 90, 0	0, 90, 45, -45, 90, 0, 0
3	45, -45, 90, 0, 90, 0	45, -45, 90, 0, 90, 0, 0

Table 1. Analyzed stackups.

B. Material data

The biggest problem in this analysis was finding suitable data. Except of composite fabric that can be found in ANSYS engineering data, most of parameters are impossible to track. The greatest example is wood – shops does not give specific information such as its type. When there is catalog of properties, authors rarely mention in which direction those parameters are validated and in case of anisotropic and orthotropic materials it is significant. Due to the lack of information, I have adapted average values and assumed that wood and styrofoam elements are isotropic.

Styroforam				**Composite fabric**			
E	20	MPa		E	X	45000	MPa
v	0,1	-			Y	10000	MPa
G	4	MPa			Z	10000	MPa
σ	0,3	MPa		v	XY	0,3	-
Balsa wood					YZ	0,4	-
E	3000	MPa			XZ	0,3	-
v	0,28	-		G	XY	5000	MPa
G	166	MPa			YZ	3846	MPa
σ	13,5	MPa			XZ	5000	MPa
Pine wood				σ_C	X	-675	MPa
E	8550	MPa			Y	-120	MPa
v	0,344	-			Z	-120	MPa
G	1200	MPa		σ_T	X	1100	MPa
σ	34,7	MPa			Y	35	MPa
				σ_S	XY	80	MPa
					YZ	46,15	MPa
					XZ	80	MPa
Legend: E – Young's modulus v – Poisson's ratio G – Shear modulus σ – Stress limits • C – compression • T – tensile • S – shear							

Table 2. Assumed material data.[4,5,6]

C. Assumptions

Plane was designed from the scratch by our leader, Katarzyna Pobikrowska. During that process she had calculated parameters and requirements which are necessary in this analysis:

Basic empty mass	8 kg
Mass of engine	1,8 kg
Length	1,2 m
Wingspan	1,95 m
Wing area	0,54 m^2
Cruising speed	45 m/s
Taper	0.7

Table 3. UAV parameters.

- Permissible flight load

Based on JAR-VLA, maximum flight load factor is n=7, caused by 12 m/s positive gust.

In theory it is relation: $n = \frac{L}{W}$, where: L – lift, W – weight. In steady flight, when those forces are equal, n=1. Although due to gusts and stages such as take-off and landing, n factor increases.

- Schrenk approximation and pressure load

Using Eq. (1-4) in paragraph I part B maximum calculated value of pressure is about p = 1.4 kPa, so 700N force. That corresponds with maximum flight factor n = 7, as we simulate analysis in those conditions despite small chances that they will appear.

Assumptions summary:

- Flight is steady and takes place in a standard atmosphere;
- Pressure load is simulated based on Schrenk approximation;
- Maximum flight factor is n=7;
- Simulation takes place on half of the wing, as it is when object has symmetry axis;
- Basic parameters and material data are adopted from Table 2 and 3.

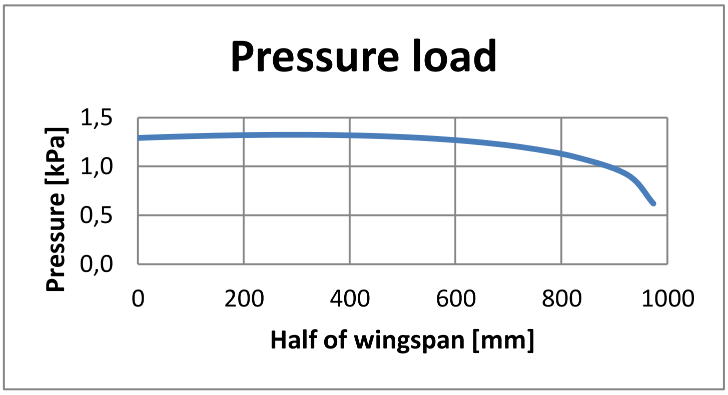

Figure 7. Pressure load.

D. Results of simulation

Only the most relevant results has been shown below.

- Failure criteria

1st case:

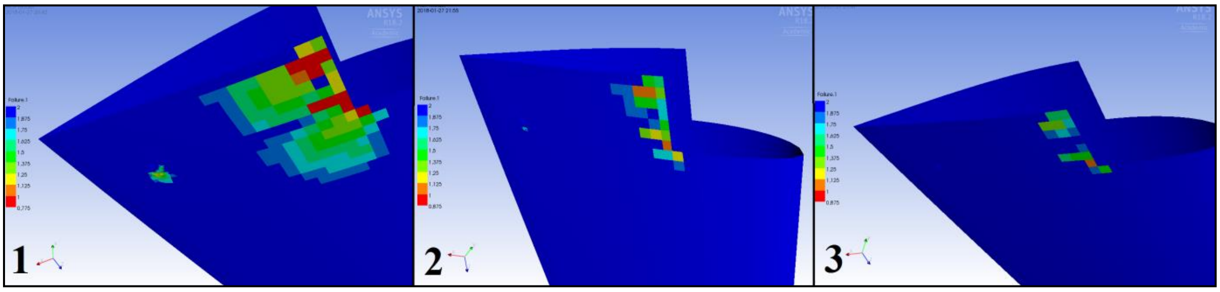

Figure 8. Impact on failure criteria caused by a different amount of plies. Examples 1-3 with stackups from table 1 (1st case).

Stackup	**1**	**2**	**3**
Minimum failure criteria	0,7789	1,0334	1,07757
Code	e2t (3)	e2t (6)	e2t (5)
Theory	Maximum stress	Maximum stress	Maximum stress
Direction	2	2	2
Layer	3	6	5

Table 4. Failure criteria for 1st case.

As expected, lowest values occured in part where wing is fasten to fuselage. The more plies in laminate, the higher failure criteria. Theoretically, when it is lower than 1, composite won’t be able to transfer the assumed load. All damages occur in second direction, which has lower strength parameters. In first example value falls under demanded 1. After adding two plies (0^0 and 90^0), it rises up to 1,0334. Although with another two layers (45^0 and -45^0) change is not that significant. Due to the character of load on a model majority of stress is transferred by 0^0 and 90^0 plies. However, this UAV is going to be laminated with hand layup – the more layers, the more trouble and supplies. That is why we have decided that in case of flight load equal 7, which is a lot for a small airplane like ours, it is no use to endeavour more plies in order to receive higher values of failure criteria - 2nd stackup should be able to transfer all loads.

2nd case:

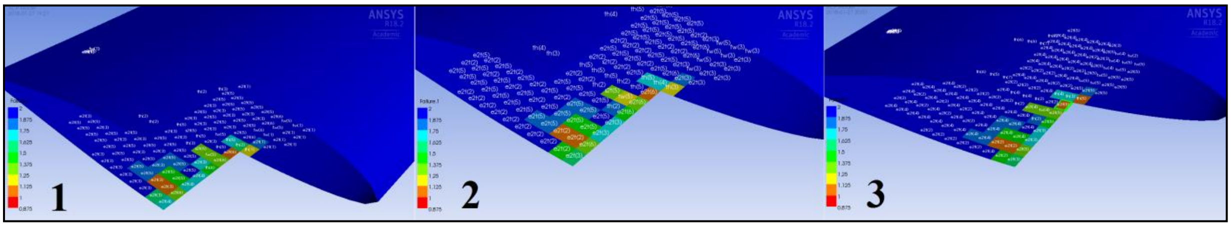

Figure 9. Impact on failure criteria caused by different order of angles on plies.

Stackup	**1**	**2**	**3**
Minimum failure criteria	1,0371	1,0334	1,0279
Code	e2t (6)	e2t (5)	e2t (6)
Theory	Maximum stress	Maximum stress	Maximum stress
Direction	2	2	2
Layer	6	5	6

Table 5. Failure criteria for 2nd case.

According to Table 5, order in which plies are lying does not affect greatly on failure criteria value. Also, in example number 3 where external layer is 45^0, it has the lowest value – as I have mentioned before, those plies transfer less loads.

Regarding to previous assumptions, 1st stackup from Table 5 is the most suitable for our model. Next results have been calculated using this layup.

- Deformations

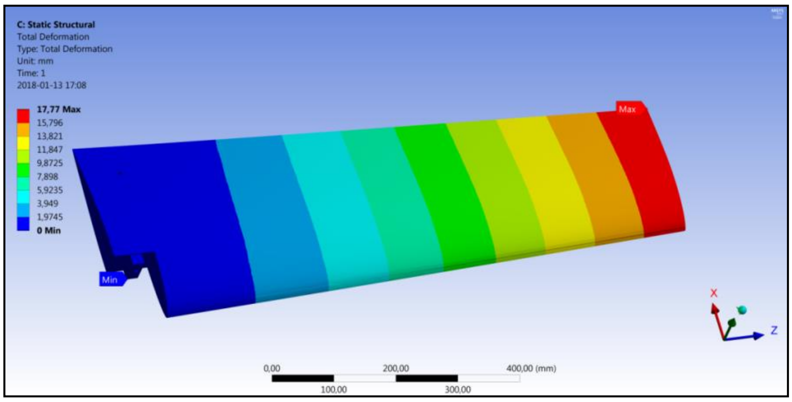

Figure 10. Total deformation of wing.

According to strength of materials theory[7], maximum deflections are at free element - end of the wing. At the same time, the closer the support - the smaller deflections gets, till they reach zero. It is typical feature of bending elements. When it comes to the value of maximum deformation – compared with half of the wingspan – it is around 2%.

- Equivalent stress

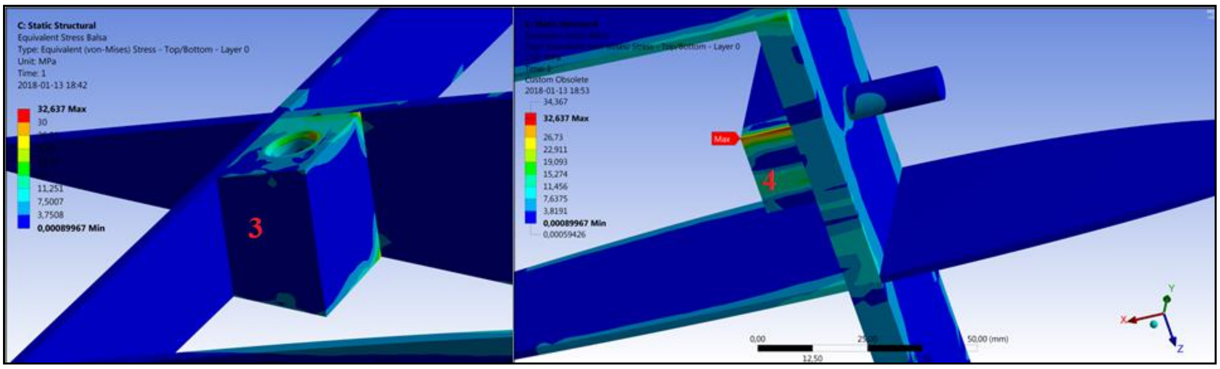

Figure 11. Equivalent stress on wooden elements.

As shown above, there is a concentration of stress due to the notch in pine block (3). Maximum stress value appears on block number 4, although it should never achieve 32 MPa because it is in one point - on the corner of element. Average maximum stress is about 27 MPa for pine components and 11 MPa for balsa.

- Stress on covering

I will show only the most load layer in 1 direction. In case of main covering it is layer 1 (0^0 angle, external), while on strap it is layer 1 (also 0^0 angle and external).

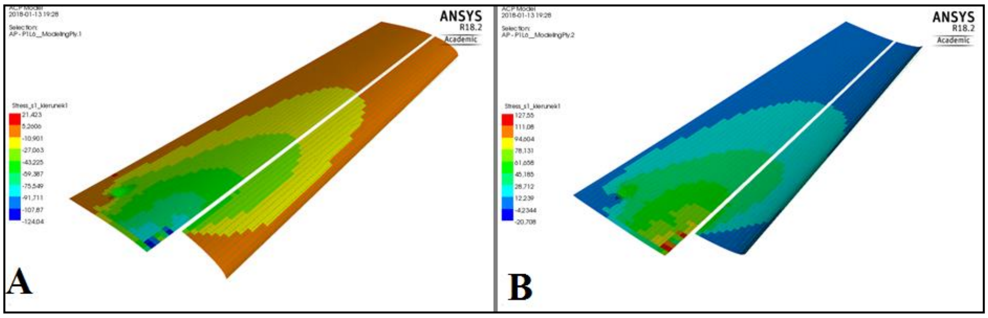

Figure 12. Stress values on main covering. A) Top; B) Bottom.

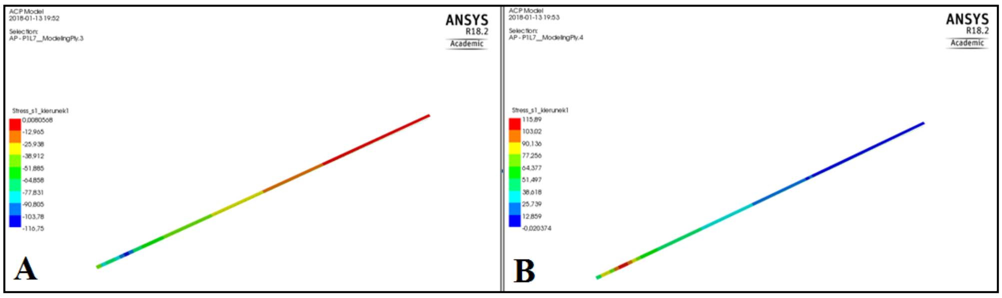

Figure 13. Stress values on straps. A) Top; B) Bottom.

Because of the different scale straps could not be on the same photo as the rest of the covering, it may be surprising that the main part has similar stress values as straps, although it is a matter of layer numbers. As I have mentioned before, straps have one extra layer due to carrying more load. Therefore cumulative stress in them is bigger.

<table>
<tr><th colspan="4">Flight load n=7</th></tr>
<tr><td></td><td colspan="2">Stress limit σ [MPa]</td><td>Stress σ_{max} [MPa]</td></tr>
<tr><td>Balsa wood</td><td colspan="2">13,5</td><td>11</td></tr>
<tr><td>Pine wood</td><td colspan="2">34,7</td><td>27</td></tr>
<tr><td>Styrofoam</td><td colspan="2">0,3</td><td>0,1</td></tr>
<tr><th colspan="4">Covering</th></tr>
<tr><td></td><td>Stress limit σ [MPa]</td><td>Stress σ_{max} [MPa] [MPa]</td><td rowspan="7">Note: these are stress limits for one layer, not for the whole layup.</td></tr>
<tr><td rowspan="2">Direction 1 (s1)</td><td>1100</td><td>127.55</td></tr>
<tr><td>-675</td><td>-124,04</td></tr>
<tr><td rowspan="2">Direction 2 (s2)</td><td>35</td><td>26,58</td></tr>
<tr><td>-120</td><td>-25,75</td></tr>
<tr><td rowspan="2">Shear in-plane (s12)</td><td>80</td><td>17,11</td></tr>
<tr><td>-80</td><td>-17,29</td></tr>
<tr><th colspan="4">Others</th></tr>
<tr><td>Deformation</td><td colspan="3">Maximum deflection 17,77 mm</td></tr>
<tr><td>Minimum failure criteria</td><td colspan="3">1,0371</td></tr>
</table>

Table 6. Juxtaposition of results.

Final layup is, in top-down sequence:

- [0, 90, 0, 90, 45, -45] on main covering
- [0, 90, 0, 90, 45, -45, 0] on straps

III. Summary

A. Conclusions

Designed UAV is going to be a test platform for pulsejet engine. Therefore it is not going to be agile or acrobatic. For such a small model it may be an exaggeration to estimate its permissible flight load. Indeed, flight load n = 7 is a very high value. That is why, despite little differences between maximum stress and stress limits or low failure criteria, we should not be afraid of structure damage – presumably, those conditions will never take place. During the analysis of composite covering, influence of order and amount of plies were shown. As we have seen, sequence of layers is not crucial in case of failure criteria, although number of layers is. Angles should be chosen after examination of forces, which are affecting simulated object.

B. Future plans

Right now we are conducting the final phase of building a full-size model of wings. The next step is to make analogous structure analysis for all construction including stabilizers, fuselage and landing gear. Then we will be able to build the rest of the model, assembly all structural and electric components, and finally – perform flight testing.

References

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[2] Galiński C., *Wybrane zagadnienia projektowania samolotów*, Warszawa, Wydawnictwa Naukowe Instytutu Lotnictwa 2016

[3] Wilczyński A., *Polimerowe kompozytowy włókniste. Właściwości, struktura, projektowanie*, Warszawa, Wydawnictwa Naukowo-Techniczne 1996

[4] http://plyty.fabertec.pl/styrodur-basf-3035cs-300kPa [sited: 27 February 2018]

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[6] Krzesiński G., Zagrajek T., Marek P., Borkowski P., *Metoda elementów skończonych w mechanice materiałów i konstrukcji*, Warszawa, Oficyna Wydawnicza Politechniki Warszawskiej 2005

[7] Bijak-Żochowski M., *Mechanika materiałów i konstrukcji Tom 1*, Warszawa, Oficyna Wydawnicza Politechniki Warszawskiej 2013