ANALYSIS OF STRESS DISTRIBUTIONS IN OSA AIRCRAFT STRUCTURE USING FINITE ELEMENTS METHOD

Aleksander Olejnik
Military University of Technology
2, Gen. S. Kaliskiego St., 00-908 Warsaw
aolenik@wat.edu.pl

Lech Jarzębiński
EADS PZL "Warszawa-Okecie" S.A.
2, Gen. S. Kaliskiego St., 00-908 Warsaw
l.jarzebinski@chello.pl

Łukasz Kiszkowiak
Military University of Technology
2, Gen. S. Kaliskiego St., 00-908 Warsaw
lkiszkowiak@wat.edu.pl

Tomasz Łącki
Military University of Technology
2, Gen. S. Kaliskiego St., 00-908 Warsaw
tlacki@wat.edu.pl

Abstract. In this paper numerical analysis of very light aircraft airframe was described. Structure displacements and stresses in strength elements were presented. This paper includes example of flight envelope design point A analysis. The process of forces calculation acting on wings and V-tail and inertia loads acting on full structure was presented. Numerical model created in MSC.Patran was described. Results of an analysis were discussed.

Keywords. AIRCRAFT DESIGN, AIRCRAFT STRUCTURES, FINITE ELEMENTS METHOD

1 Introduction

The strength analysis of an aircraft airframe is one of the most important steps in aircraft design process. Assumptions from preliminary design project, verified by concept evaluation and aerodynamic shape design process, allow to calculate forces acting on airframe structure. Correctly calculated forces allow to design single elements of aircraft, main components and finally complete aircraft. In many cases, strength calculations of single element are not complicated and there is no need to use Finite Elements Method. When it comes to the proof strength of complete airframe, static and stiffness tests are performed. Because of costs of those test it`s good to make sure that airframe structure is free of design mistakes. FEM methods allow to calculate stresses and deformations of airframe of any degree of complexity. Numerical model allow to calculate stresses and deformations for any load case and can complement static tests of airframe. Any mistake revealed at this stage contributes to the reduction of costs due to design mistakes.

Today there are lots of different FEM software programs, but only few of them are approved by biggest aircraft manufacturers and aviation authorities. In this paper calculations of very light aircraft - OSA are presented. MSC Patran/Nastran software was used. Described calculations were made to prove the earlier calculations for aircraft components. Calculated forces will be used as loads for future static tests of airframe and results of numerical calculations will be compared to those from experiment.

2 OSA – Very Light Aircraft

OSA aircraft is two-seat, side-by-side aircraft, equipped with upper-wing supported by a strut. Airframe is made of aluminum and composite. The main structure elements are made of aluminum.

Front part of fuselage - cockpit, engine fairings, main landing gear beam, wing tip and tail tips are made of composite. The wings are equipped with slots and slotted flaps. Aircraft is powered by Lycoming IO-320-B1A (160 hp), driving 62 inch propeller. Aircraft is designed in two versions with V-tail and standard tail. Tricycle landing gear is equipped with steerable front wheel. Cockpit is designed for perfect visibility and comfort. Aircraft is equipped with glass cockpit by Garmin. There are two Garmin G3X screens with GNS530 ILS system. Control system consists of central mounted Y-type stick and rudder pedals on both sides. Aircraft was designed as reconnaissance aircraft with daylight and thermal camera and board computer with touch screen for special equipment.

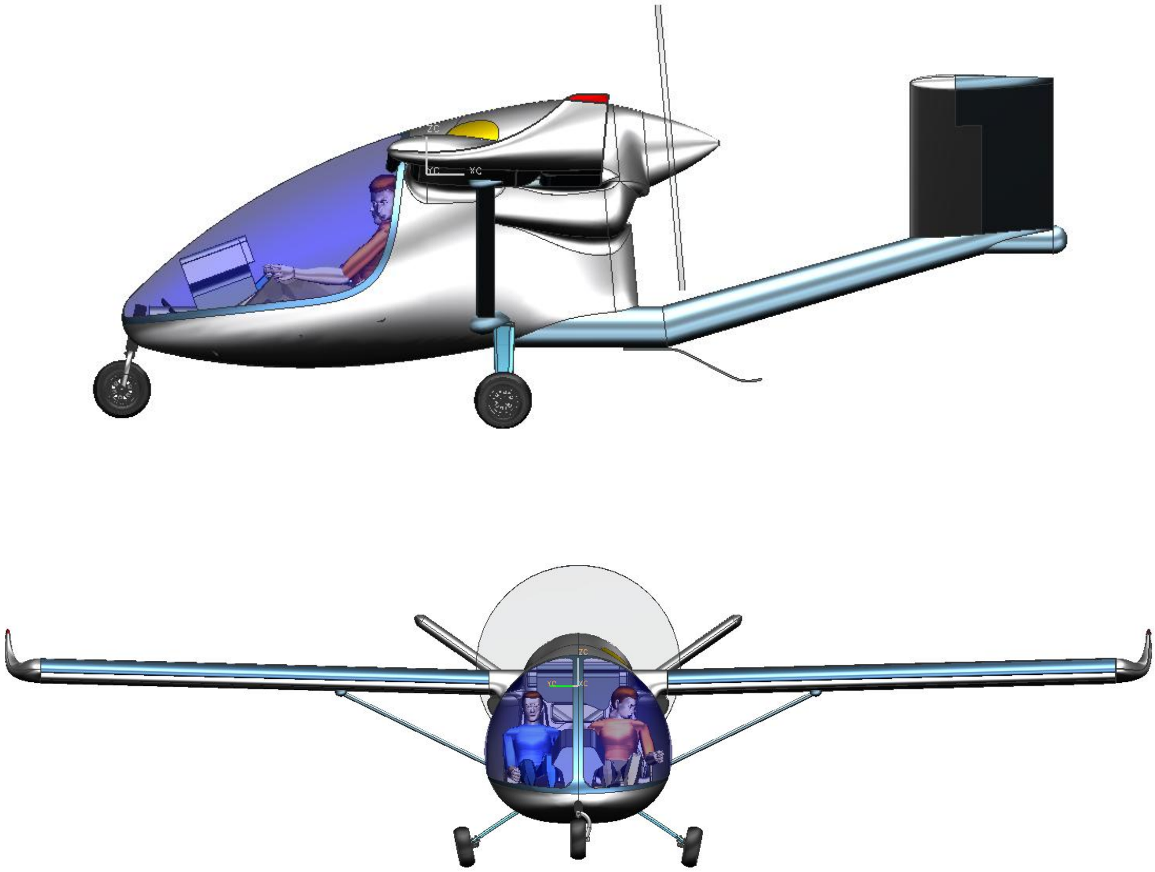

Figure 1: OSA aircraft.

3 Flight envelope

Flight envelope is a basic chart for calculation of force acting on aircraft during the flight. Flight envelope is a set of every possible flight conditions. It allows to calculate forces acting on airframe components for most demanding flight conditions. Flight envelope for OSA aircraft was calculated according to CS-VLA regulations and assumptions from preliminary design project and performance calculations.

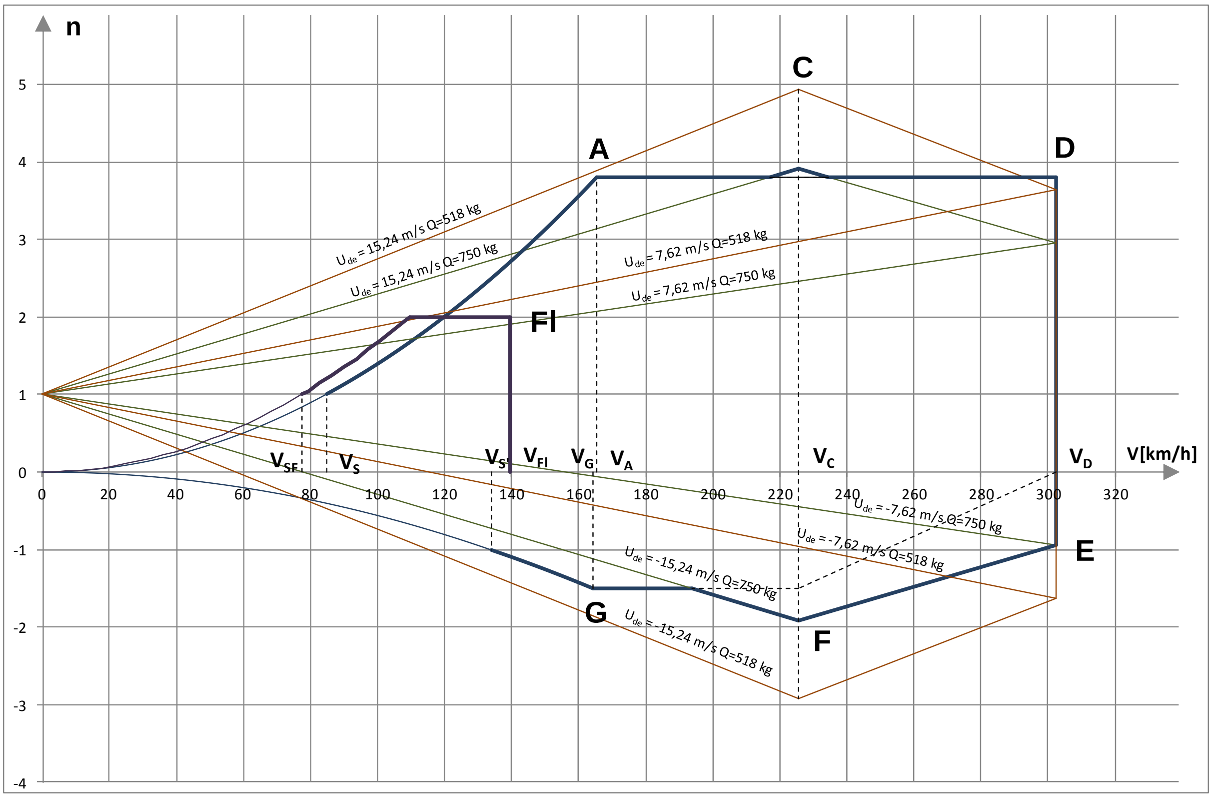

Figure 2: Flight envelope for OSA aircraft [4].

4 The calculations of forces acting on wings and tail

4.1 Wing loads

Wing loads were calculated using Multhopp`s method, Vortex Line Method. For preliminary calculations wing geometry was simplified to trapezoidal shape with NACA 2415 airfoil with a fixed slot. Wing tip – winglet was replaced by equivalent rectangular surface at the wing tip. The fuselage influence on wing was taken into consideration and conformal projections method was used for transformation of the fuselage cross section. The cross section of fuselage in wing area is transformed into stream line with boundary condition for wing fuselage transition. For calculations experimental characteristics of NACA 2415 airfoil with fixed slot were applied. [5,6]

According to A design point from flight envelope:

- Velocity : $V = V_A = 165.4$ km/h = 45.946 m/s
- Maximum load factor to be expected in service acting normal to the assumed longitudinal axis of airplane: n=3.8

Forces acting on plane for longitudinal stability:

- Aircraft weight: Q=750 kg
- The resultant force acting on center of gravity of the airplane: $P_z = n \times Q = 2850$ kG

- Force acting on horizontal stabilizer (established at this stage of calculation): $P_{ZH0} = 0$ kG
- The resultant force acting on wings:
  - force normal to X axis of the airplane: $P_{zN} = 2850 - 0 = 2850$ kG

The force P_{ZN} acting on wings decomposes into P_{NC} and P_{TC} calculated based on aerodynamic characteristics of the wings:

$$P_{NC} = 1392.4878 \text{ kG}$$
$$P_{TC} = -436.59062 \text{ kG}$$

$$P_{ZN} = P_{NC} \times \cos(\alpha_z) - P_{TC} \times \sin(\alpha_z)$$
$$P_{XT} = P_{NC} \times \sin(\alpha_z) + P_{TC} \times \cos(\alpha_z)$$

$$P_{ZN} = 2 \times (1392.4878 \times \cos(5°) - (-436.59062) \times \sin(5°)) = 2850.48\text{kG}$$
$$P_{XT} = 2 \times (1392.4878 \times \sin(5°) + (-436.59062) \times \cos(5°)) = -627.13 \text{ kG}$$

- Pitching moment on wings reduced 22% of MAC (corresponding to main spar plane) on symmetry plane:

$$M_S = 2 \times (-111.15195) = -222.30 \text{ kGm}$$

For automation of the calculations process a program in Fortran programming language was written. As a result of calculations, spanwise distributions of lift force, drag force and pitching moment reduced to 22% of MCA were obtained. Results were presented on charts on figure 3-7.

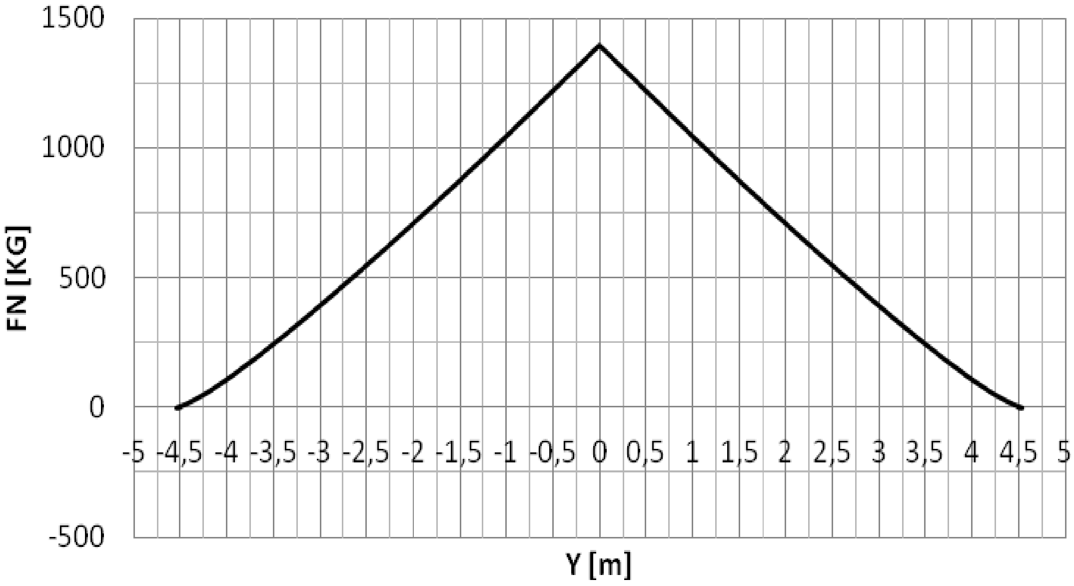

Figure 3: Spanwise distribution of force normal to wings chordal plane FN without considering the strut. OSA – A design point of flight envelope.

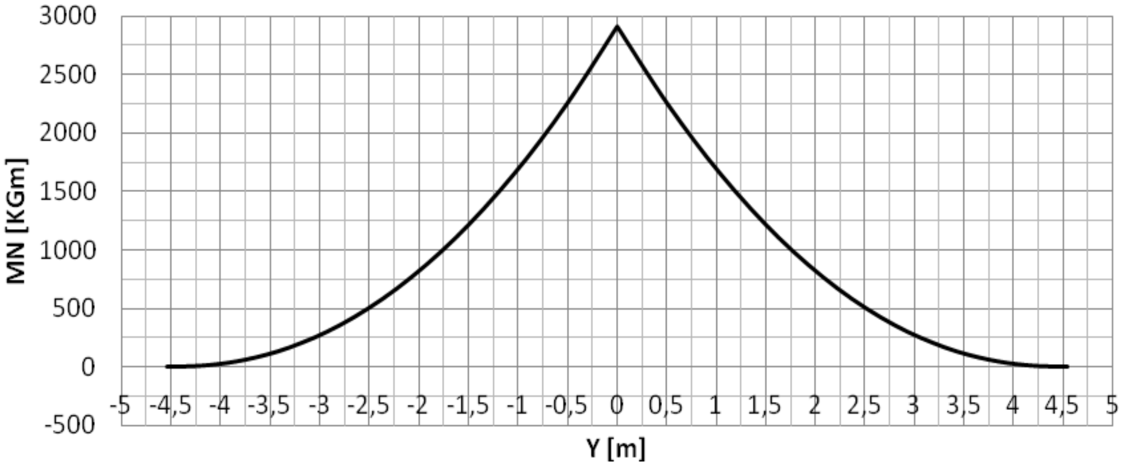

Figure 4: Spanwise distribution of bending moment MN (in plane perpendicular to wings chordal plane) without considering the strut. OSA – A design point of flight envelope.

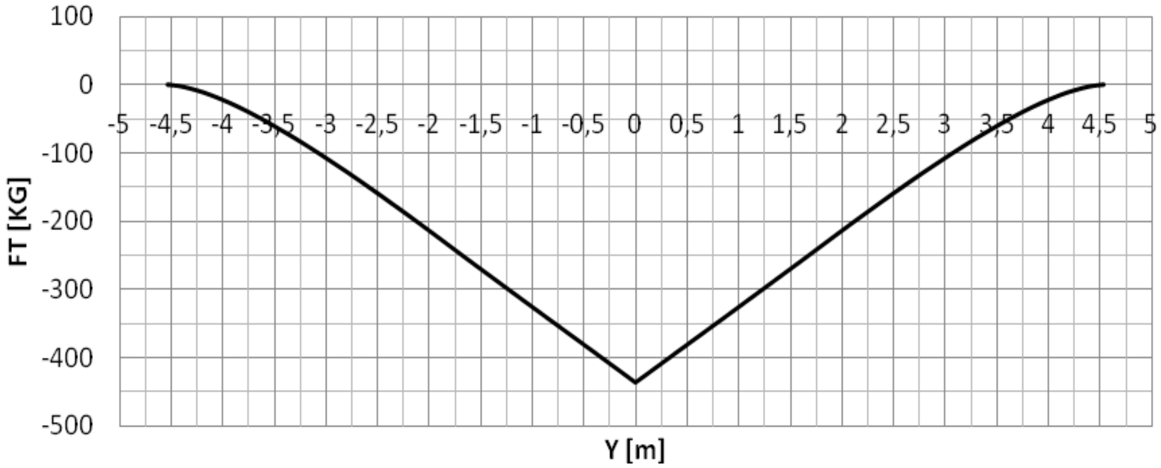

Figure 5: Spanwise distribution of force tangent to wings chordal plane FT without considering the strut. OSA – A design point of flight envelope.

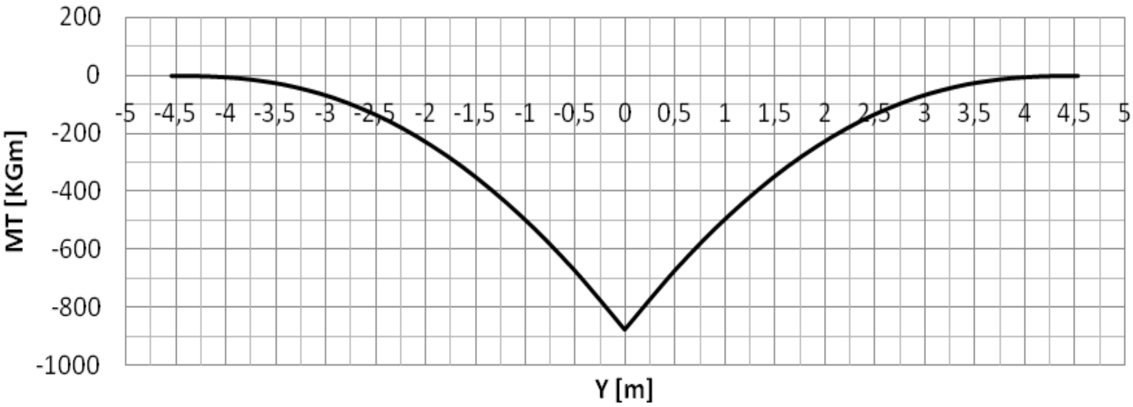

Figure 6: Spanwise distribution of bending moment MT (in wings chordal plane) without considering the strut. OSA – A design point of flight envelope.

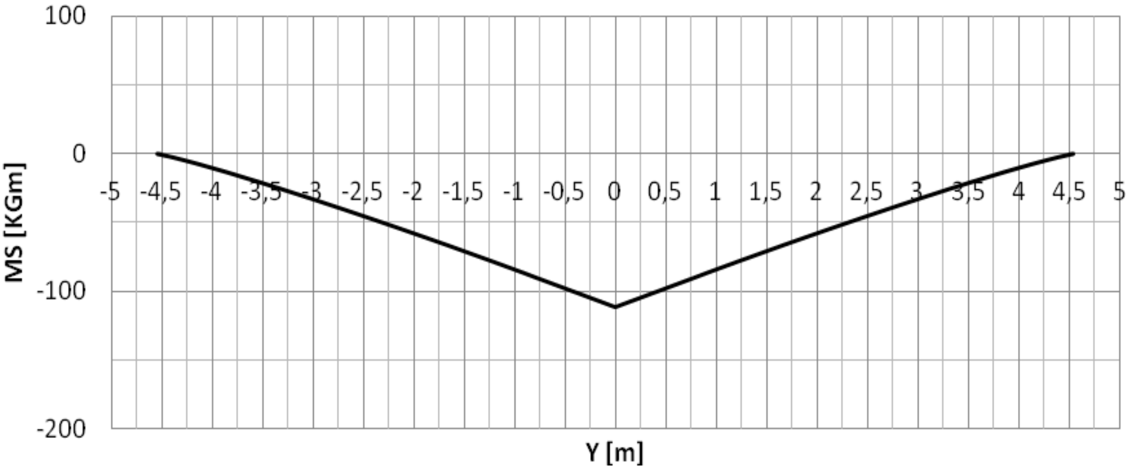

Figure 7: Spanwise distribution of torsional moment MS (reduced to 22% of MAC) without considering the strut. OSA – A design point of flight envelope.

4.2 Tail loads

Tail loads were calculated using Vortex Lattice Method (VLM) implemented on VORLAX program. First version of the aircraft was equipped with V-tail. The fuselage was modeled using vortices located at selected cross sections of the fuselage. The wings were modeled by vortices evenly distributed along spanwise direction and harmonically along wing chord. The airfoil curvature was modeled by curvature of airfoil mean line. Airfoil thickness was not taken into consideration. Tail beam was simulated by vortices evenly distributed along selected cross sections. V-tail surfaces were described by vortices distributed along span and chord. For calculations of loads acting on control surfaces, two panel sets were created. First set of panels simulated stabilizer, second set simulated tail control surface. [2, 7, 8]

Table 1: Load distribution for V-tail. OSA – A design point of flight envelope.

OSA: Q =750 kg; XMAC = 29.26%; V=VA=165.4 km/h; n=3.8, Deflection to balance pitching moment

N	RS	CN	CL	CY	CM	PN	PL	PY	XSP	YSP	ZSP
						kG	kG	kG	m	m	m
17	0.00616	-0.07652	-0.00034	0.00027	0.00135	-0.671	-0.484	0.385	4.431	0.132	-0.257
18	0.00616	-0.52523	-0.0028	0.0015	0.0059	-4.607	-3.99	2.137	3.404	0.138	-0.253
19	0.00667	0.09478	0.00045	-0.00036	-0.00065	0.9	0.641	-0.513	3.818	0.283	-0.152
20	0.00667	-0.70044	-0.00404	0.00217	0.00877	-6.654	-5.756	3.092	3.413	0.275	-0.157
21	0.00667	0.0092	0.00004	-0.00004	0.00044	0.087	0.057	-0.057	4.712	0.417	-0.058
22	0.00667	-0.74022	-0.00427	0.00229	0.00953	-7.032	-6.084	3.263	3.416	0.422	-0.055
23	0.00667	0.7211	0.00344	-0.00276	-0.00891	6.85	4.901	-3.933	2.984	0.574	0.052
24	0.00667	-0.73929	-0.00426	0.00229	0.00977	-7.023	-6.07	3.263	3.417	0.569	0.048
25	0.00667	0.95293	0.00455	-0.00364	-0.01192	9.052	6.483	-5.186	2.955	0.696	0.138
26	0.00667	-0.72502	-0.00418	0.00224	0.00982	-6.887	-5.956	3.192	3.416	0.717	0.152
27	0.00667	-0.07212	-0.00034	0.00028	0.00156	-0.685	-0.484	0.399	3.858	0.885	0.27
28	0.00667	-0.66002	-0.0038	0.00204	0.00915	-6.27	-5.414	2.907	3.412	0.862	0.254
29	0.00965	-0.18759	-0.0013	0.00104	0.00405	-2.578	-1.852	1.482	3.381	1.13	0.441
30	0.02122	-0.51619	-0.00947	0.00508	0.02296	-15.607	-13.493	7.238	3.295	1.091	0.414

sCN = -3.16463 sCLH =-0.02632 sCY = 0.01240 sPN = -41.12 kG
sPL = -37.50 kG sPY = 17.67 kG clH = -0.05264 pzH = -75.00 kG
cmH = 0.12363

4.3 Inertia loads

Forces acting on airplane during the flight:

- aerodynamic forces acting on external surfaces of aircraft;
- weight or inertia loads of airframe elements or equipment;
- thrust with torque, if thrust is produced by propeller or any other rotating parts;
- reactions of the ground.

A three dimensions airplane flight dimensions can be described as a linear motion of its center of gravity and rotations around this point.

According to Newton`s law:

$$\overline{P} = \frac{d}{dt} \quad \text{(momentum vector)}$$

$$\overline{M} = \frac{d}{dt} \quad \text{(angular momentum vector)}$$

For aircraft motion analysis the following assumptions have been made:

- weight of the aircraft is constant and equal to weight for currently considered flight condition;
- external loads cause deformation of the aircraft structure;
- load distribution changes with deflection of control surfaces.

The additional assumptions have been made:

- maneuvering loads changes are very slow compared to period of airframe oscillation (influence of vibrations was not taken into consideration in calculation of maneuvering loads);
- despite the fact that deformation of the airframe has influence on loads that caused that deformation, discussed effect is very small and it can be assumed constant during maneuver;
- in extreme flight conditions deformations for maximum loads are considered;
- loads changes caused by control surfaces deflections during the maneuver are very slow comparing to deflection speed of those surfaces (aerodynamic forces are equal to forces acting on aircraft surfaces in considered moment in time).

All of above assumption allow to simplify equations of motion.
As a result, system of equation describing aircraft motion can be simplified, as follows:

$$\sum X = m(\dot{u} - rv + qw)$$
$$\sum Y = m(\dot{v} - pw + ru)$$
$$\sum Z = m(\dot{w} - qu + pv)$$

$$\sum M_x = I_x\dot{p} - (I_y - I_z)qr - I_{yz}(q^2 - r^2) - I_{zx}(\dot{r} + pq) - I_{xy}(\dot{q} - rp)$$
$$\sum M_y = I_y\dot{q} - (I_z - I_x)rp - I_{zx}(r^2 - p^2) - I_{xy}(\dot{p} + qr) - I_{yz}(\dot{r} - pq)$$
$$\sum M_z = I_z\dot{r} - (I_x - I_y)pq - I_{xy}(p^2 - q^2) - I_{yz}(\dot{q} + rp) - I_{zx}(\dot{p} - qr)$$

There are several flight conditions that need to be calculated:

- unbalanced external forces acting on airframe results in linear and rotational accelerations, corresponding velocities are equal 0;
- some velocities are constant, external forces cause accelerations and velocities due to curved trajectory.

As a result, system of equation can be written down, as follows:

$$\sum X = m\dot{u}$$
$$\sum Y = m\dot{v}$$
$$\sum Z = m\dot{w}$$
$$\sum M_x = I_x\dot{p} - I_{zx}\dot{r}$$
$$\sum M_y = I_y\dot{q}$$
$$\sum M_z = I_z\dot{r} - I_{zx}\dot{p}$$

If the aircraft is treated as discrete mass point set, where distances between masses are constant, substituting to equations the mass of the aircraft, moment of inertia and external loads, linear and rotational accelerations can be determined. In the next stage, inertia forces acting on airframe elements and equipment of the aircraft can be calculated. Forces distribution acting on fuselage along its length are presented on figure 8 [3, 9].

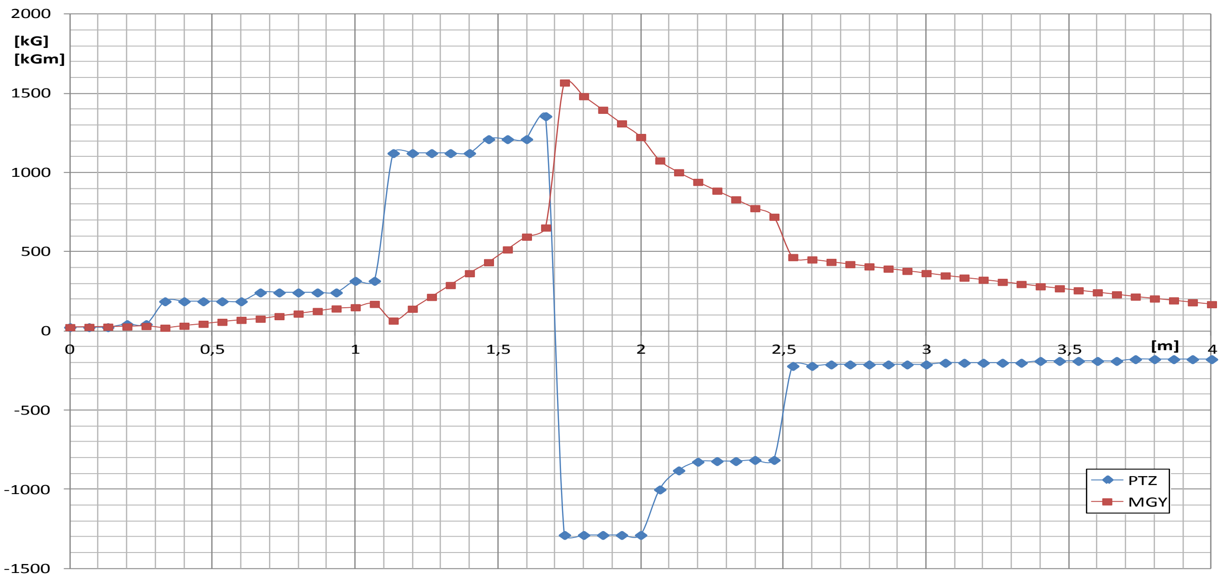

Figure 8: The A design point of the flight envelope – shearing force and bending moment along fuselage length.

5 An aircraft FEM model

According to an aircraft geometry created using Siemens NX software and design documentation an airframe FEM model was created. Calculations were performed using MSC Patran/Nastran software. During FEM model creating process a several simplifications were made. FEM model was built using QUAD and BAR elements instead of 3D elements. Properties of the elements correspond to airframe parts which are simulated and forces acting on those parts. Hinges were created between wing and aileron and flap. An additional element CELAS simulates stiffness of the aircrafts control system. To prevent overconstrain of the airframe components, hinges were created instead of bolts on fuselage-wing connection. The strut was created as a single rod element. FEM model of an airframe was created to simulate real structure as closely as possible. Skins of wings and tail surfaces were defined as shear panels. Every component of the airframe was divided in groups that allow to analyze stresses and deformations of single component. Wind shield was not created. Composite material was described as isotropic material with average elastic modulus and Poisson`s coefficient.

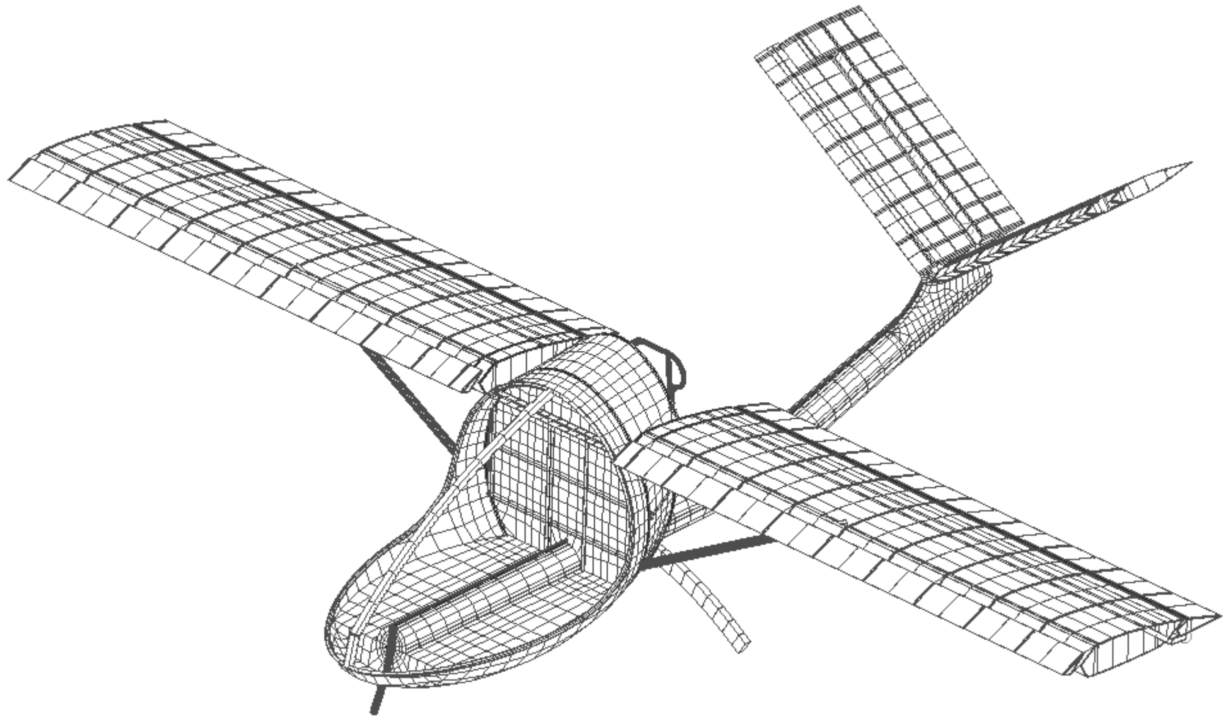

Figure 9: FEM model of complete airframe.

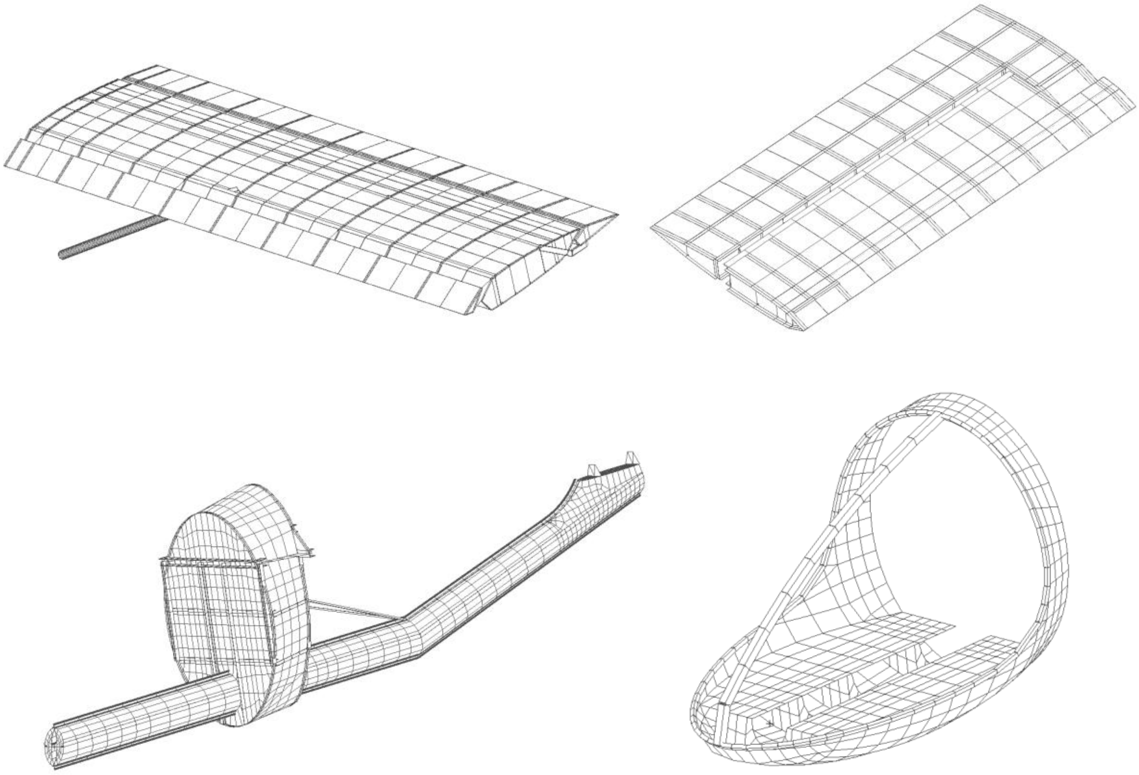

Figure 10: FEM models of the airframe components.

6 Analysis results

FEM model was completed with previously calculated external forces. Aircraft was fixed by rod elements. One end of those elements was connected to FEM model airframe another one was fixed. Due to balanced forces acting on airframe, resultant forces and moments should be equal 0. If there are no mistakes in calculations, the forces in rod elements should be close to 0.

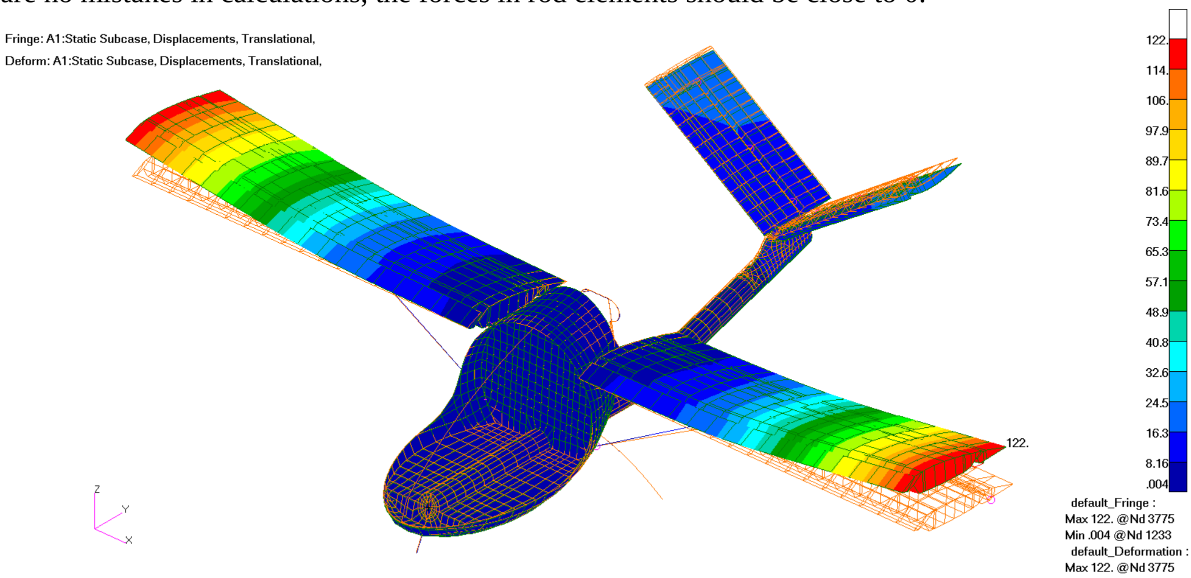

Figure 11: Airframe displacements for the A design point of flight envelope. (displacements in mm).

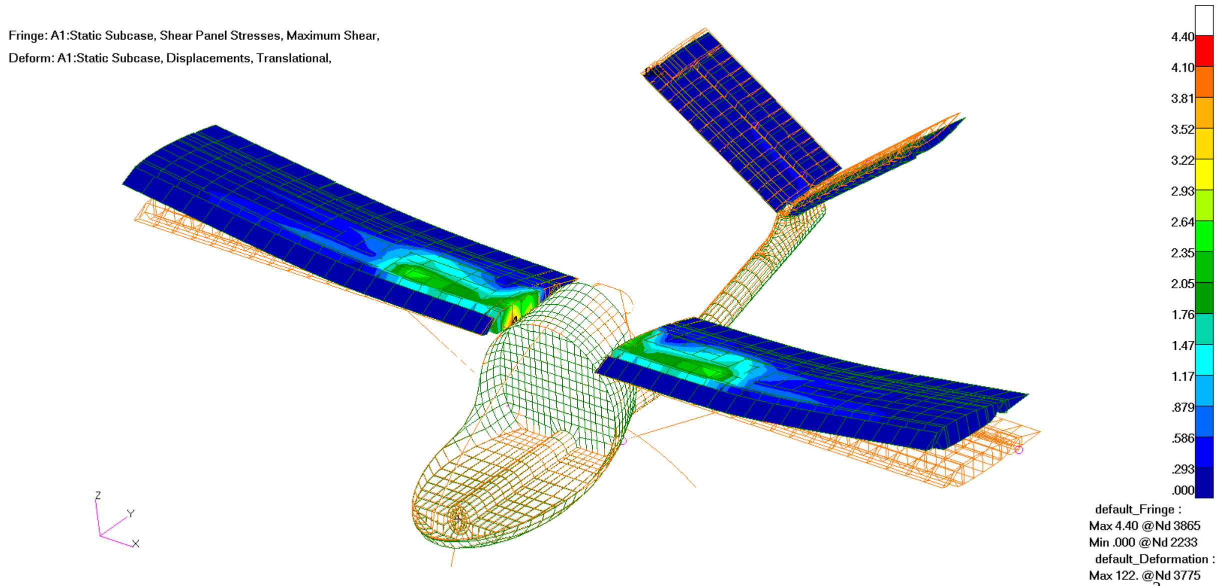

Figure 12: Shear in the airframe components for the A design point of flight envelope. (shear in kG/mm^2)

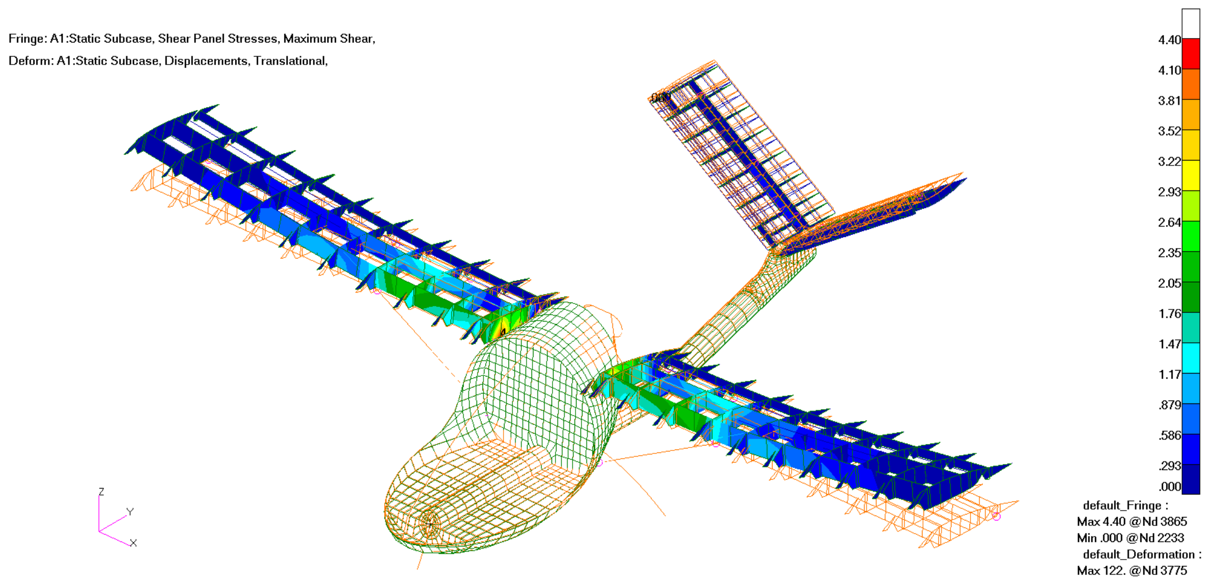

Figure 13: Stresses in spars and ribs of wing and tail surfaces for the A design point of the flight envelope. (stresses in kG/mm^2)

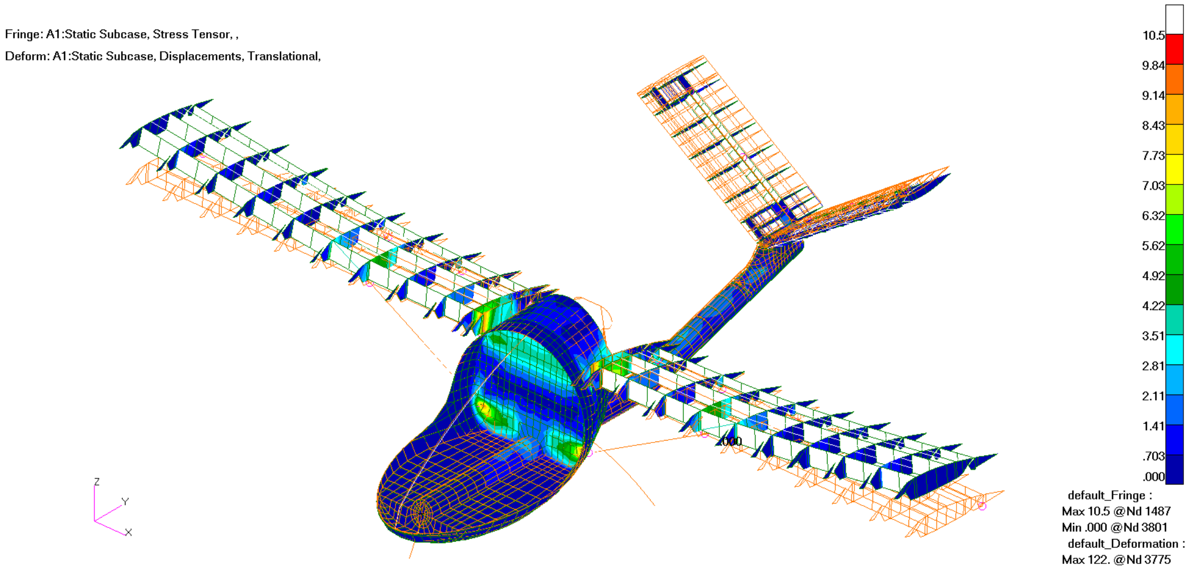

Figure 14: Von Misses stresses in fuselage structure for the A design point of the flight envelope (stresses in kG/mm^2)

7 Conclusion

Numerical calculations of airframe strength can aid design process. It is important for structures that should be durable and safe, especially when a minimum mass criterion is taken into consideration. The airframe is a structure which carries loads to its limits and because of that, information about places and elements with large stress concentrations need to be specified and carefully analyzed. For any load condition during the flight, plastic deformation should not occur and extreme loads (flight loads multiplied by safety factor) acting on the structure should not destroy an airplane. Structure elements with stress concentrations are potentially sources of damage of an airplane due to fatigue. Numerical calculations can be used as a starting point for static test experiments. The numerical analysis results will be compared to experimental results. Based on differences between numerical and experiment results, numerical model can be validated. Improved FEM model can be analyzed in terms of other load cases which were not investigated during static tests.

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