# VUT 001 MARABU FUSELAGE DESIGN METHODOLOGY

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**Abstract.** The article describes the design methodology of an experimental aircraft VUT 001 Marabu. The project is coordinated and partially manufactured by Institute of Aerospace Engineering of Brno University of Technology. The article is focused on design of composite fuselage and describes all phases of realization: initial design, aerodynamic design of aircraft configuration, detailed support structure design, manufacturing technology, and static test. The emphasis is placed on effective modern computer aided design methods used during the project.

**Keywords**: Aircraft design, CFD, FEM, CAM, CAE, composite structure.

## 1 Introduction

The VUT 001 Marabu (figure 1) is a two-seater, central-wing with glass/carbon-fiber composite fuselage and a metal wing and horizontal tail unit. The aircraft is meant for flight measuring and for autonomous controlling research. The aircraft is equipped with two engines. A piston engine with rear mounted propeller and a small jet engine located on the wing [1].
The whole project was supported by Czech Ministry of Industry and Trade under FI-IM3/041 grant ("Design and realization of VUT 001 Marabu aircraft for UAV applications in civil sphere"). Industrial partners První brněnská strojírna Velká Bíteš , Jihlavan – Airplanes and PlastServis are also engaged into the project [2],[3].


Figure1: VUT 001 Marabu

Aircraft VUT 001 Marabu was one of few practical projects fully coordinated, designed and also largely manufactured at university thanks to chief designer of the Marabu project Prof. Píštěk, who has a courage and patient to realize this project at university environment. The Aerospace institute modified its human resources structure according to company standard. The most of the work was made by doctoral students and especially, in the final stages of project, also by master science degree students who were divided into groups according to different activities (aerodynamics, structure design, stress analyses, systems and manufacturing). Institute of Aerospace Engineering (IAE) had unique opportunity to create and make up real product to verify some modern approaches, methods and solutions during designing of prototype.
One of the significant partial tasks of IAE was design and manufacturing of fuselage. An important part of the project was to focus as much as possible on using computer-aided designing during fuselage development. After successful project finish, it should allow us to effectively reconsider benefits of computer-aided design (CAD), computer-aided engineering (CAE) and computer-aided manufacturing (CAM) implementation to development process. Fuselage design methodology is described by block diagram in figure 2.

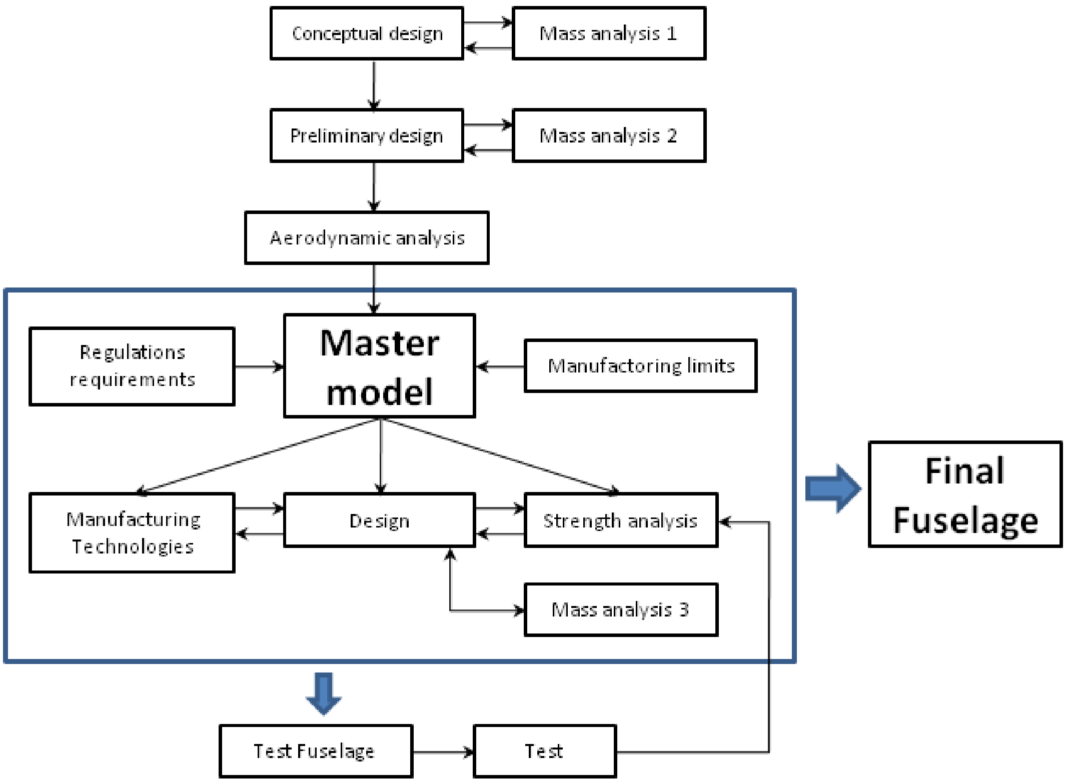


Figure 2: Design methodology block diagram

## 2 Conceptual and preliminary design

The fuselage, as one of the biggest structure parts, is developed already in early stage of project. It is subjected to all great changes during defining of aircraft concept and its specification. VUT 001 Marabu passed through three main concept changes (figure 3). CAD software proved to be efficient in this project stage. It allowed immediate and relatively fast access to all information needed about aircraft concepts.

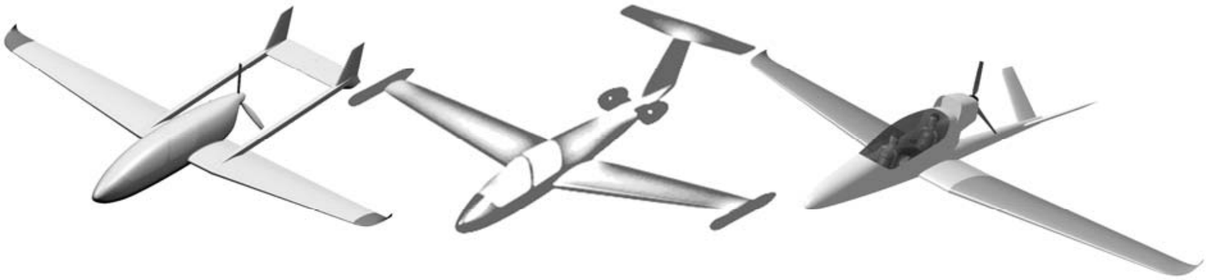

Figure 3: Evolution in preliminary design

Concept no. 3 (figure 3) entered to next stage of designing process. Its preliminary design contained the first real data and parameters, which were used as inputs to basic aerodynamic, ergonomic and stress analyses. Last but not least mass analysis was refined. This process was interactive and sufficiently fast because 3D parametric model was used.

## 3 Aerodynamic analyses

Modern CFD tools (CAE system) were used in aerodynamic design of aircraft at two levels - overall aerodynamic concept evaluation and particular details fine tuning. The aerodynamic properties of design were evaluated from qualitative and quantitative point of view using Navier-Stokes steady, fully turbulent finite volume based solution of flow field at different flight regimes. CFD approach was already used during conceptual design, and especially during selection of tail unit arrangement. Two different arrangements, V and T-tail surfaces, was studied with emphasis on interference with propeller induced flow (Figure 4).

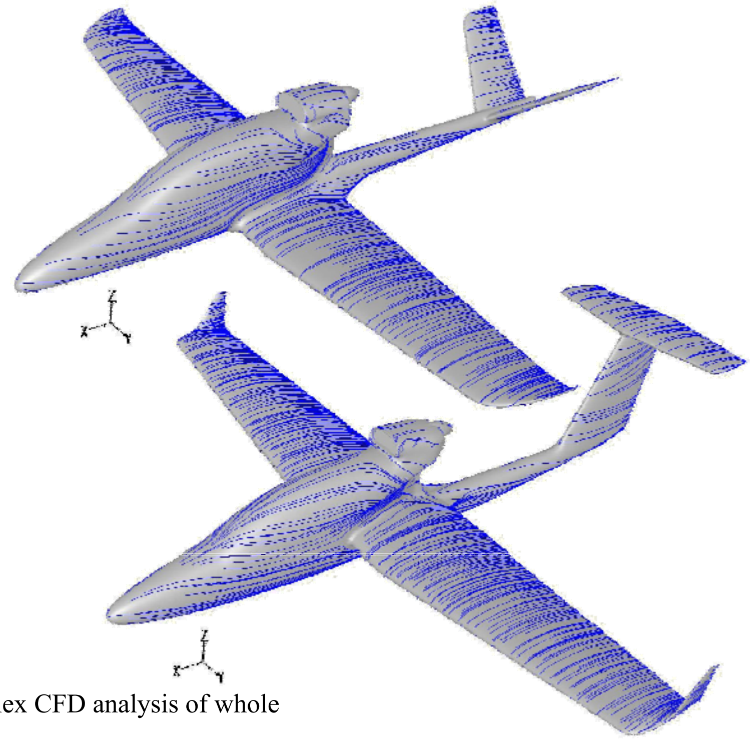

Figure 4: complex CFD analysis of whole configuration

At next step, exhaustive CFD analysis of isolated fuselage and whole 3D configuration were performed. Main goal of isolated fuselage analysis was to find drag characteristics (figure 5) and to choose the best wing incidence angle.

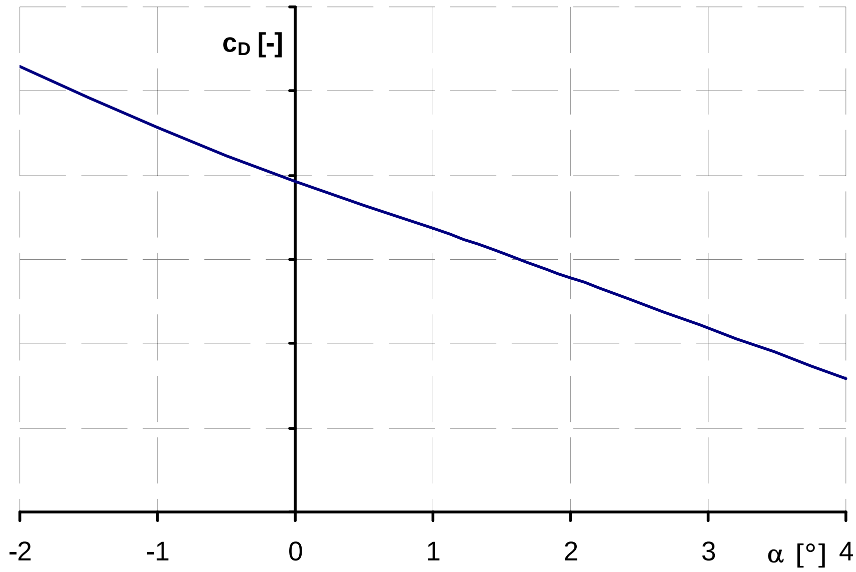

Figure 5: fuselage drag characteristics

During analysis there was found flow separation under engine nacelle behind cockpit at higher angles of attack. The shape of critical area was modified in a few steps and separation was delayed as far as internal structure design limitations and overall aircraft concept allowed. Predicted improvement in flow pattern at critical area can be seen in figure 5.

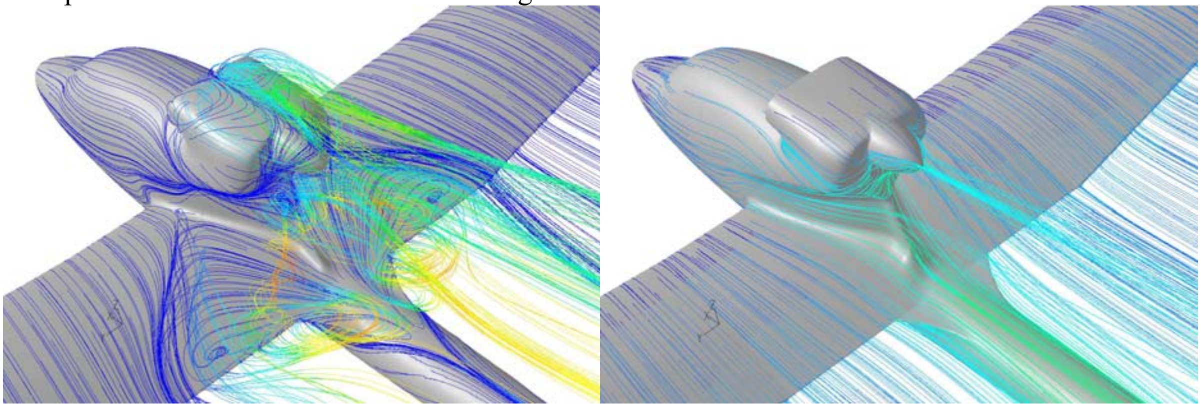
Figure 6: cabin and engine nacelle shape modification

At later stage of design computations of whole 3D configuration were used as basis for Pitot-static system location prediction. The target of analysis was to find a place on the fuselage where static and total pressures are not affected by change in angles of attack and sideslip. The location with minimum static pressure dependence on angle of attack is shown in figure7.

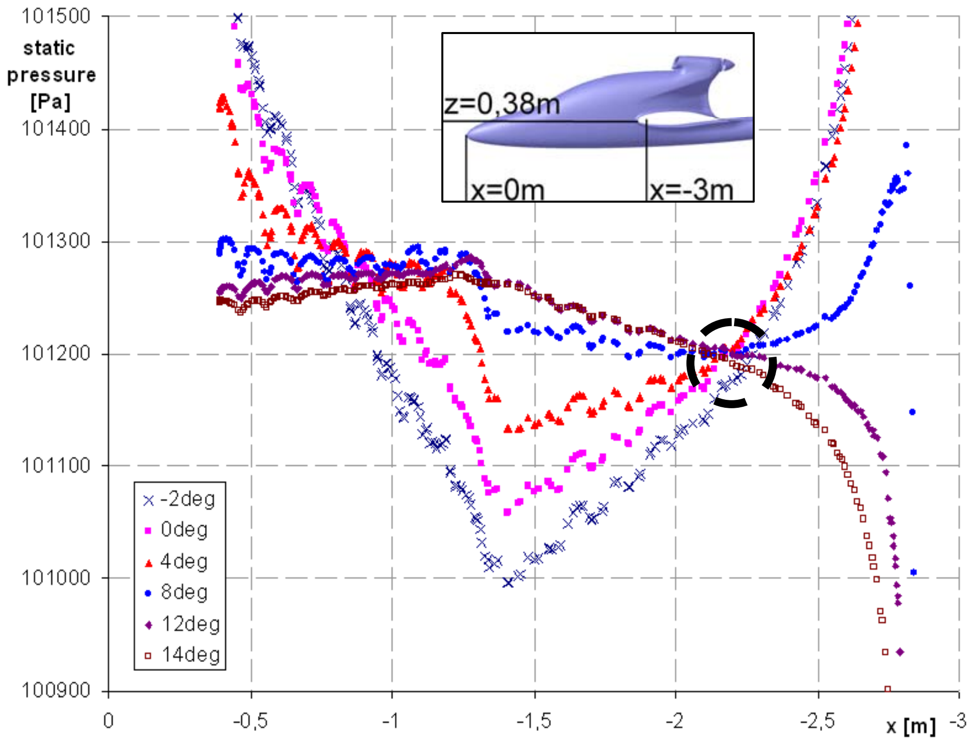

Figure 7: Static pressure distribution – pitot-static system design

Another example of extensive use of CFD method for solution of a particular practical problem is engine cooling analysis and nacelle shape design. Upon engine cooling system characteristics, aerodynamic design of engine nacelle shape was performed. Optimal position of coolers, air inlet and outlet shapes with regard to air supply and air exhaust from engine area with minimum drag was estimated. The cooling and cowling of piston engine is a very complex task, which is still solved mainly by experimental approach during test flights. CFD analysis of such a complex problem is still not very precise. Nevertheless it gave us at least early warning about critical problems and it directed us into detailed design of outer and inner shape of nacelle and ducting.

# 4 Detail design

## 4.1 Master model

The main documentation creation started after completion of preliminary aerodynamic and ergonomic analyses. In this case it was 3D master model of fuselage (figure 8). Fuselage master model is created by system outer surfaces, by system surfaces of walls and stringers and by hinge and leveling points. As such it contains all fundamental parameters and it is main documentation basis for all other structural parts which are somehow connected to fuselage. And also data for manufacturing and for next designing and refining analyses are based on fuselage master model geometry.

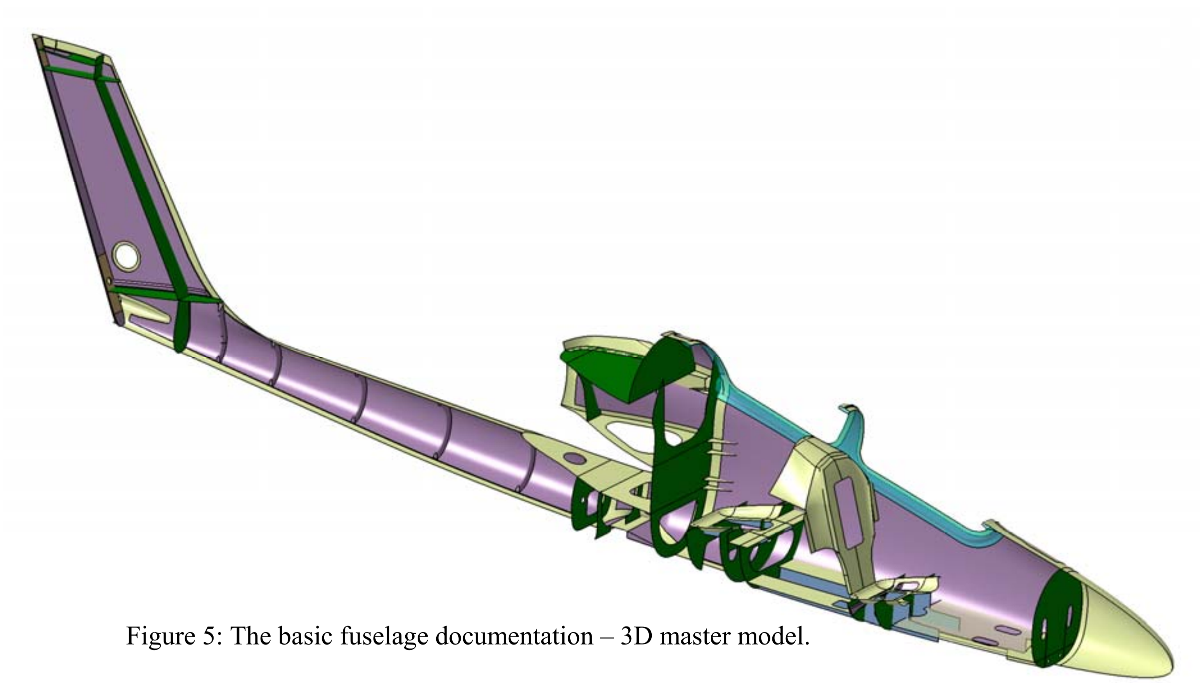

Figure 5: The basic fuselage documentation – 3D master model.

## 4.2 Regulation requirements and manufacturing limits

The aircraft is designed according to CS-VLA regulation requirements. The development was limited by some manufacturing constrains and regulations restriction:

- Fuselage manufacturing in two-pieces female mould with the mould joint in the longitudinal symmetry plane
- Minimization number of moulds from time and financial reasons
- Factor of safety for composite material supporting structures has to be increased on 2.25. Standard safety factor 1.5 is multiplied by 1.25 – special safety factor of temperature influence and 1.2 – special safety factor of moisture influence

## 4.3 Structure design and stress analysis

The fuselage structure was designed as mainly glass fiber composite laminate with sandwich core. Carbon fibers were used in local reinforcement. Pure glass fiber reinforcement plastic (GFRP) without sandwich core has proportional stiffness (Young modulus/density) 2.5 times worse than aluminum alloy. Thus GFRP without sandwich core is not effective in term of minimum structure weight. Sandwich core material was used for overall composite fuselage including flat bulkheads. Using of sandwich material led to vacuum bagging curing.
The complete 3-D model created before manufacturing and used in design phase of the project reduces the role of standard drawing documentation. Drawing documentation was used only in communication with external suppliers. During the manufacturing of the fuselage composite parts drawing documentation was not used at all.
We must consider that the complex shape of the fuselage does not have any straight edge necessary for planar length measuring. Therefore longitudinal dimension was measured as 3D dimensions from leveling points on the mould surface.

### 4.3.1 Lay-up design

All materials used in the fuselage were certified. Interglass 92110 and 92125 twill fabric, Interglass 92145 unidirectional fabric and unidirectional carbon tape were used. L-285 epoxy resin with L-287 hardener was also used.
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The layer mechanical properties were considered according to conservative values accepted by LBA (The German Civil Aviation Authority) [4](table 1).

| | 92110 | 92125 | 92145 | TCU |
|---|---|---|---|---|
| $E_1$ [MPa] | 16600 | 16600 | 30320 | 77000 |
| $E_2$ [MPa] | 16600 | 16600 | 7161 | 3400 |
| $E_x$ [MPa] | 10500 | 10500 | | |
| $\nu_{12}$ [-] | 0,2 | 0,2 | 0,37 | 0,28 |
| $G_{12}$ [MPa] | 3800 | 3800 | 2345 | 1620 |
| $G_x$ [MPa] | 7900 | 7900 | | |
| t [mm] | 0,17 | 0,3 | 0,23 | 0,49 |
| $\sigma_{1t}$ [MPa] | 95 | 95 | 294 | 420 |
| $\sigma_{1c}$ [MPa] | 95 | 95 | 200 | 420 |
| $\tau_x$ [MPa] | 95 | 95 | | |

Table1: Used composite material characteristics

Index „x“mechanical layer properties laid up under angle 45° to the angle of loading

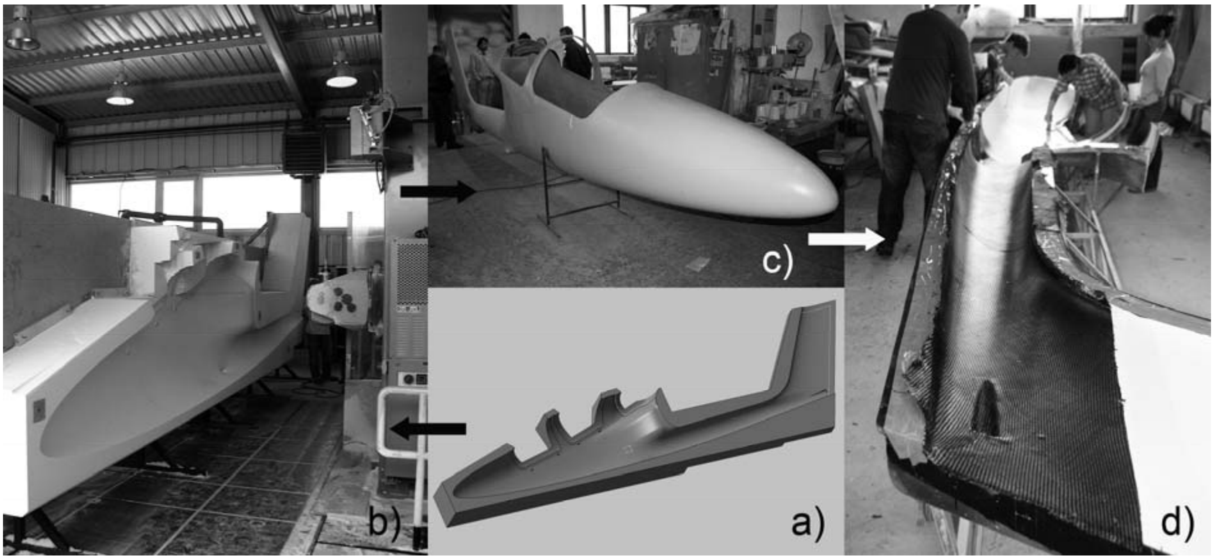
The composite material mechanical properties used at the fuselage design (especially $\sigma_{1tm}$, $\sigma_{1c}$) are considerably lower than we are able to reach in laboratory conditions. It is caused by imperfections in fabric structure which arise during the manufacturing. The imperfections decrease those strength characteristics considerably. If we want to use higher strength characteristics we would have to carry out analysis of influence of manufacturing on strength characteristics.
The composite plies in small aircraft fuselage are usually laid-up under two directions. Longitudinally –fibers are along the longitudinal fuselage axis and diagonally –fibers are under angle 45° to the longitudinal fuselage axis. The longitudinal plies are subjected primarily to normal loading (bending, tension, compression) support. The diagonal plies subjected to shear forces and torque moment. The normal loadings are distributed by diagonal plies very non – effectively. It causes degreased Young modulus of material with the fiber angle 45° to the angle of loading. The same principle arises during shear loading of a longitudinal laid plies.
The laminate design began with a survey of stacking sequences of similar aircrafts. Information about stacking sequences of the composite plies is available in maintenance manuals [5], [6] (necessary in case of repairing the structures). The manuals are accessible on the Internet.
The laminate design strategy was based on minimal weight requirement. Therefore the number of layers over the whole fuselage length was minimized. All the more local reinforcements were used in places of the load concentrations. Load intensity of the small aircraft structures is relatively low. Therefore only 1-2 layers are needed for each load direction (longitudinal and under angle 45°) for achieving structure load capacity, if the layers are designed to the maximum strength according to table1. In this way, a thin walled structure susceptible to buckling will be obtained. For that reason all available features against buckling were used: especially sandwich core, ribs and stringers. The

stringers were consisted only from diagonally layered plies. They were not designed for normal loading but as a prevention of a fuselage cross section collapse.
Local reinforcements were used at the horizontal tail unit hinges, in the connection between fuselage cone and keel, and in place of centre-wing. Unidirectional carbon tape was used along the whole fuselage (two tapes around the fuselage cone), as a keel spar flanges, flanges of the power unit box and reinforcement along cockpit edge [9].
All design strength analyses were carried out before manufacturing using software package MSC Paran/Nastan (CAE) with considering of geometrical nonlinearities. The linear 2D orthotropic material model was used for simulating of the composite material and linear isotropic material model for sandwich cores. The whole fuselage including sandwich core was modeled by shell elements. The elements properties were set down by Laminate Modeler module [7] which is implemented in preprocessor MSC. Patran.
The finite element model used in simulation was structurally simplified in order to reduce the simulation time consumption. One nonlinear simulation took approximately 40 minutes. The manual method of progressive plies modifying was applied. The final version was achieved in 9 iteration steps.
The main goal of the simulation was to specify the layers amount in structure, number of layers working as local reinforcements. It also showed that sandwich material has to be applied all over the structure of the fuselage. (The table of Herex sandwich core is rigid and not easily applicable into curved surfaces; that is why we considered to do not apply the sandwich in fuselage cone.)

### 4.4 Manufacturing technology

Chosen concept and usage of fiber reinforced plastics led to selection of manufacturing technology of fuselage monocoque in two-piece negative mold. Main structural walls were placed and adhesive bonded to manufactured fuselage monocoque. Most of all composite parts were manufactured by hand lay-up with consequent vacuum bagging.
It was needed to speed up manufacturing of monocoque mold after finishing of outer shape of master model. The mold was manufactured in few steps. The first step was creating of 3D model of mold, which was based on master model and it also contained marks of leveling points and some marks of system planes (figure 9a). It was not possible to manufacture mold with all important marks of system planes and technological stops due to lack of time. The milling of two-pieced negative mold was the second step (Fig. 8b). Polyurethane was chosen as stock for mold milling mainly due to economic reasons. Mold was milled by 6-axis CNC milling machine JOBS with max stock dimensions 7.5x3x3m (CAM). In next step, composite positive model of fuselage was manufactured in polyurethane negative mold. In consequence, surface finishing was done on positive model. Surface finishing of negative polyurethane mold would be very complicated and durability of this mold was also very weak. In last step, two-pieced negative composite mold was manufactured with the aid of positive model and one half of polyurethane negative mold. Durability of composite mold is approximately 10 manufacturing cycles.
Fuselage monocoque manufacturing started after completion of design and stress analysis. All necessary geometrical data were adopted from master model. For example information about surface area of particular layers, which were lay-up to mold, allowed to determine exact amount of matrix for achieving required fiber volume ratio 35%. And it was also possible to determine final weight of monocoque. Difference between weight assumption determined before manufacturing and weight of monocoque after releasing from mold is 5%. Also precise unfolded patterns were manufactured for all sandwich cores.
Monocoque manufacturing itself lied in lay-up and impregnation of particular structural layers and subsequent vacuum bagging of right and left half of mold (Fig. 9d). Some walls and stringers were

bonded into right and left half of mold after curing and before monocoque bonding. The rest of cross-sectional and longitudinal system was installed to manufactured fuselage monocoque.




# 5 Results and evaluation

## 5.1 Static test and comparison with analysis

The static test was carried out in IAE testing facilities (figure 10). The tested load cases were [8]:

1. Keel and horizontal tail unit loaded together – bend and torsion of the fuselage cone.
2. Maneuver on horizontal tail unit – bend of the fuselage cone.

Evaluated data during the test were deformation in selected points on the fuselage top surfaces, displacement of the structure and acting force. The deformations were measured by 27 strain gages located especially on the fuselage cone and keel. The HBM 10/120LY13 strain gages were used for tension/compression measuring and strain gages HBM 6/120XY12 for shear forces measuring. The strain gages were adhesive bonded on the structure using certified glue HBM Z80. The structure displacements were measured by aripots.

The data during the test were stored using by data logger ESAM Static. The loaded force was created by two hydraulic cylinders.

Both load cases were applied on the structure to 100 % of limit load level. During the test of the first load case to ultimate load level the test was stopped at 140% due to big deformation between keel and fuselage cone. During the second load case (maneuver on the horizontal tail unit) the structure withstood to 225% of limit load level without failure. On the basis of the first load case test result, the reinforcement of the connection between keel and fuselage cone was designed.

Figures 11 and 12 show the comparison of the simulation result with the static test result. The first represents dependence between limit loads from the second load case on displacement which was measured on the end of the fuselage. The difference between maximal values of the test and simulation is 12.5%. The second is limit loads versus deformation in selected point on the fuselage according to figure 9.

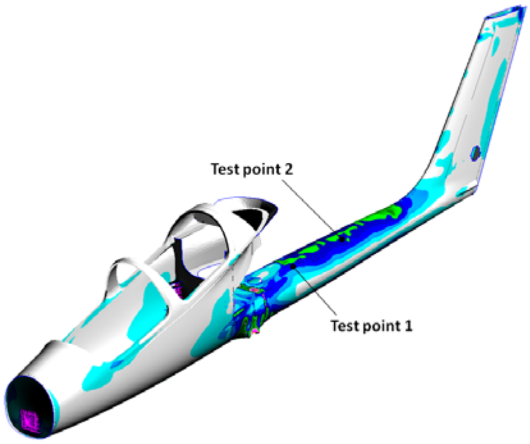


Figure 9: FEM model with highlighted deformation test points

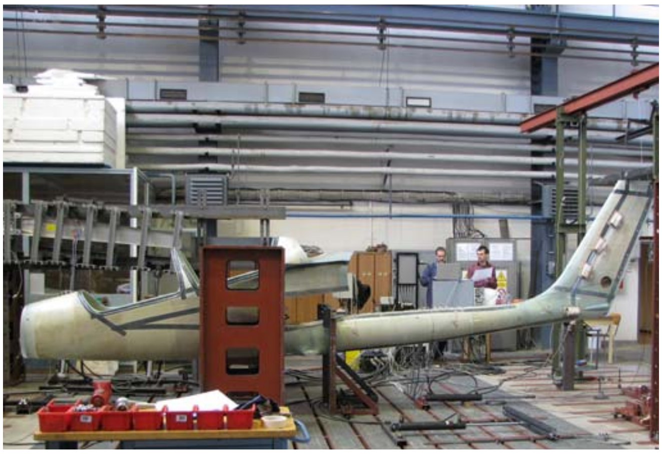


Figure 10: Static test

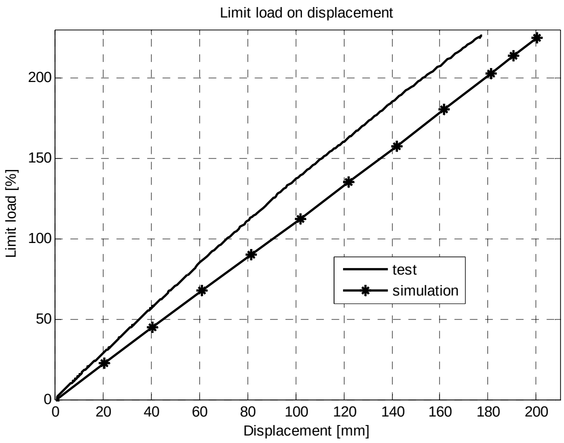


Figure 11: Displacement results at the end of fuselage.

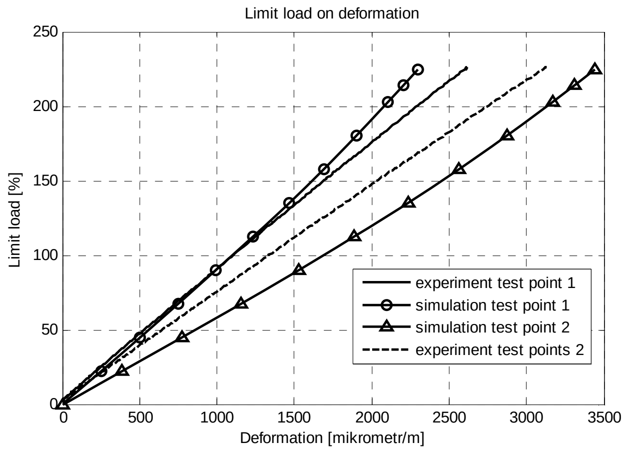


Figure 12: Deformation results

## 6 Conclusion

The IAE completed the realization of the VUT 001 Marabu project. The practical problems solved during the project were implemented into the lectures and increased the skills of the participated students. Also some problems suitable for subsequent research were found.
At this moment, benefits of computer-aided designing application can be evaluated from manufacturing point of view (interaction of CAD and CAM) and from design and stress analysis point of view (CAD, FEM). Benefits from aerodynamic point of view (CAD, CFD) will be reconsidered during flight tests.
Usage of 3D fuselage master model is crucial from manufacturing point of view, mostly when manufacturing is connected with CNC milling of fuselage monocoque mold. Fuselage monocoque creates approximately 80% of whole fuselage structure. Classic 2D drawing documentation is insufficient and it becomes useless. Only 3D documentation contains all parameters about structure and also all data for manufacturing and for analysis. Main deficit during manufacturing process was that it was not possible to manufacture mold with all important marks of system planes and technological stops for cross-sectional and longitudinal systems and for hinges.
The practical application of CAD and CAM led to sufficiently lightweight, technologically effective and dimensionally precise structure on the first attempt. Failure did not occur in the structure during the test. Weak parts of the structure were detected and their reinforcement was designed. The

reduction of some supports features will be reconsidered in the feature. The reserves can be in some ribs, stringers and local reinforcements.
The finite element model used at the analysis was created for design of the laminate structure before manufacturing was begun. And so, this model contains number of inaccuracies created by partial changes of structure during the manufacturing process and also inaccuracies created thanks to simplifying of the model for time consumption reduction. In near future a modification of the model is planning according to final fuselage structure and according to enhanced material properties.


[1] Píštěk, A.: Technická specifikace – zadání letounu VUT 001 Marabu (Technical Specification of VUT 001 Marabu Airplane), internal report LU53-2006-001.AS, 2006

[2] Píštěk, A., Hlinka, J.; Urík, T.; Šplíchal, M.: VUT 001 Marabu: Design philosophy and realization of an experimental aircraft supporting UAV applications in civil sector, Twenty-Fourth Internaltional Conference Unmanned Air Vehicle Systems, Bristol United Kingdom, 2009

[3] Hlinka, J.; Šplíchal, M.; Finda, J.: Equipment, systems and installations for experimental aircraft supporting UAV applications, International Symposium Light Weight Unmanned Aerial Vehicle Systems and Subsystems, Oostende, Belgium, 2009

[4] Interessengemeinschaft Deutscher Akademischer Fliegergrupen, Dimensionierungsrichwerte für den Segel und Motorsegelflugzeugbau, März 1988

[5] DA 40 Series Maintenance Manual, Diamond Aircraft Industries, Wiener Neustadt, Austria, 2003

[6] H 36 Dimona, Instructions For Continued Airworthiness, Hoffman Aircraft corp., Vienna, Austria, 1985

[7] MSC. MD Patran Laminate Modeler User’s Manual, Santa Ana CA, November 2007

[8] Vaněk F., Zatížení ocasních ploch a zadní části trupu VUT 001 Marabu, LU26-2007-001.L, 2007

[9] Mališ., Podklady pro výrobu trupu VUT 001 Marabu, LU04-2009-001.DE, 2009