29th Congress of the International Council
of the Aeronautical Sciences
St. Petersburg, Russia; September 7-12, 2014

AERODYNAMIC DESIGN OF FLYING WING WITH EMPHASIS ON HIGH WING LOADING

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Keywords: *Aerodynamic design, CFD*

Abstract

This paper presents an aerodynamic design process of an aircraft in flying wing configuration. This is part of a greater project multipurpose unmanned an aircraft to fly in turbulent atmosphere. This project is conducted at the Warsaw University of Technology at the Faculty of Power and Aeronautical Engineering, Department of Aircraft Design.

The main goal of presented project is to design a vehicle which will be able to fly in a high turbulent atmosphere. Within the frame of the project, the shape of the aircraft, the airfoil selection and design of control surface were made. The final configuration was analysed and the basic aerodynamic characteristic are presented. Moreover, the analysis of the trim condition was made

1 Introduction

The main goal of presented project is to design a vehicle which will be able to fly in a high turbulent atmosphere. Moreover, it was assumed that a catapult will be used to assist the take-off. Therefore, the research was focused on finding such configuration of the flying wing [1] that will be able to generate a sufficient high lift force. The shape of the wing, span-wise chord distribution, wing twist and the geometry of flaps and ailerons were investigated. All these modyfication should allow to reduce the take-off speed and to obtain a high wing loading ratio.

2 General assumption to the project

2.1 Project Assumption

According to the project assumption the initial configuration was created. Fig. 1 presents the layout of the initials configuration of the aircraft with flying wing configuration.

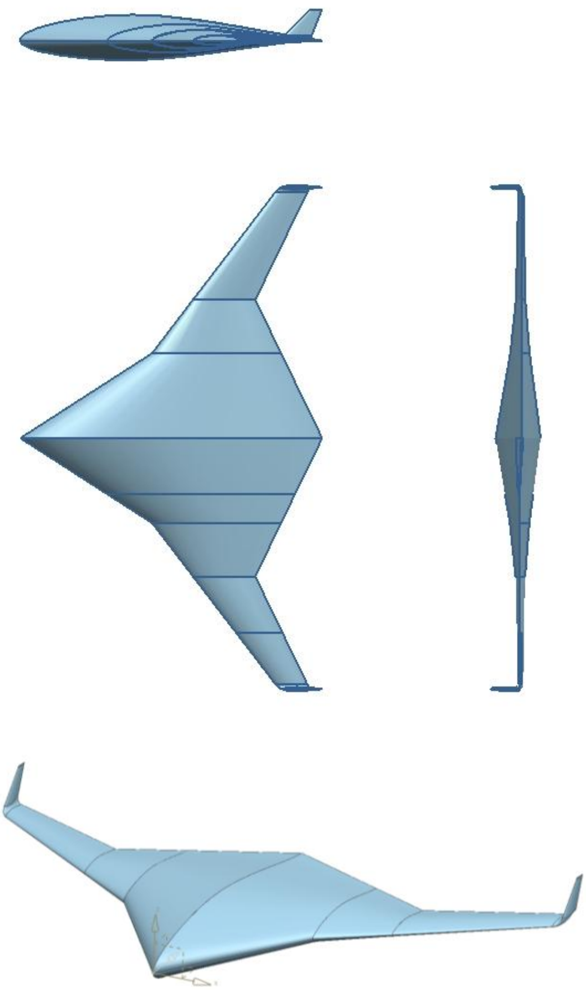

Fig. 1 The concept of the aircraft

The concept assumed design of the aircraft with the flying wing configuration. The pair of the vertical stabilizers on the tips of the wing is provide to obtained lateral stability.

The wing has a double cranked leading edge and trailing edge. It was assumed that an inner part of the wing has a greater sweep angle of leading edge than the outer one. The initial configuration has no a geometric and aerodynamic twist.

The reference data of the aircraft is presented in the Table 1.

Table 1 Reference data of preliminary configuration

Name	Value
Reference area S	$0{,}920\ [m^2]$
Span b	$2{,}000\ [m]$
MAC $\bar{c}$	$0{,}692\ [m]$

2.3 CFD method

The most of the work in this project was focused on aerodynamic analysis. Three different methods was used during the work. For conceptual calculation a panel method [2] was used. This method is faster but less accurate. The VSAERO [3] software which is based on the potential flow calculation was used. Moreover, the method is enhanced on the boundary layer method. The second one, more advanced, is based on Euler equation [4] and was used to calculate two segmented airfoil. For this kind of calculation the MGAERO[5][6]software was used.

Moreover, for the initial calculation of an airfoil the method, based on Navier-Stokes [4][7] equations was used.

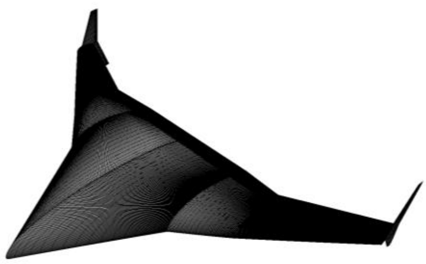

Fig. 2 Computational Mesh

The example of geometry and surface grid of a numerical model has been presented in the Fig. 2.

2.2 Airfoil selection

The selection of the airfoil for the flying wing configuration is a very important issue. It is expected that the airfoil provides the following requirements: high lift coefficient and low pitching moment. Both of them are connected with the trim condition.

NACA 64_2-215 [8] airfoil was used to the conceptual configuration of the aircraft. However, calculation result of the maximum lift coefficient was not satisfying. So, the search of a new airfoil was started. The two segmented airfoil was found. It allow to obtain a high lift coefficient. Next, the redesign process was started to fit the airfoil to the aircraft. For further analysis two models of the two segmented airfoil with different thickness ratio (17% and 19%) was chosen. The initial calculation of mentioned airfoils were made by the Fluent. Fig. 3 presents an exemplary calculation of the two segmented airfoil.

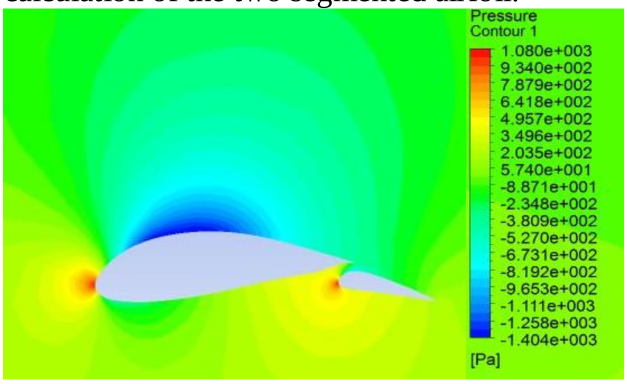

Fig. 3 Pressure distribution for the two segmented airfoil

During calculations the flap effectiveness for both airfoil were investigated. The aerodynamic calculation of airfoil reveal that expected lift force is sufficient for the project requirements. However, use of this kind of airfoil causes significant increase of a drag force and a pitching moment coefficients too.

2.4 Control surfaces

The number and the magnitude of the control surface were changed according to the project stage. At the first stage of the project the inner flap and the elevons place on the outer wing were considered. The main aim for both of them was to increase in the lift force and satisfy the trim condition. Moreover, elevons was used to provided used to roll control too. Fig. 4 presents the layout of the preliminary aircraft configuration with highlighted control surfaces (green colour).

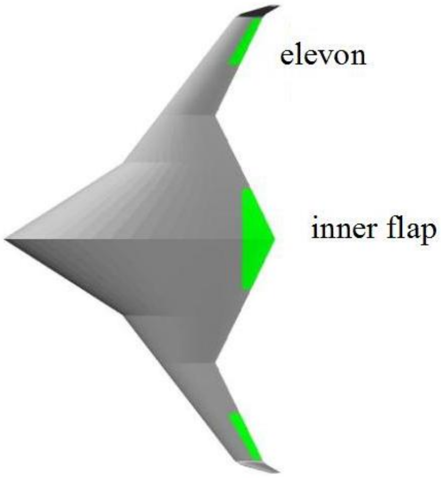

Fig. 4 The preliminary concept of the aircraft control surfaces

According to the results of aerodynamic analysis (paragraph 4), the number of control surfaces was increase to three. It was implemented intentionally. Much more control surfaces increase the possibility to obtain the best trim condition.

The last model (Model No 9) is equipped with three pairs of independent control surfaces. All of them are provided to the longitudinal control and satisfy the trim conditions. Fig. 5 presents the initial geometry of the flying wing with the two segmented airfoil. In this case the inner flap is deflected.

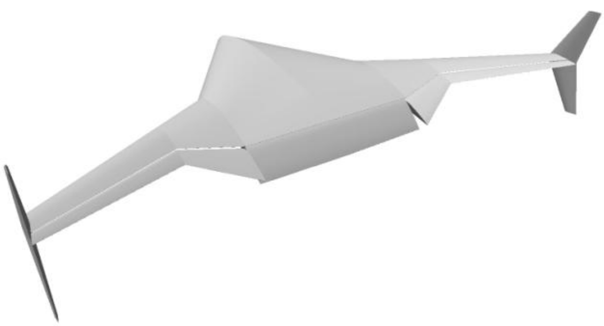

Fig. 5 Geometry for flying wing with two segmented airfoil (Model No7)

2.5 Propulsion

The configuration of propulsion was not considered in the first stage of the project. The last considered model (Model No 9) was equipped with the model of propulsion presented in the Fig. 6.

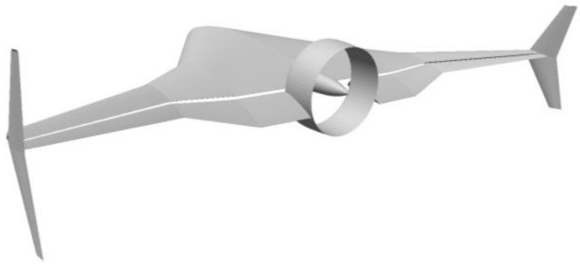

Fig. 6 Model No 9 of the aircraft with the propulsion model

The propulsion model impact on the size of the inner flap is significant.

3 Aerodynamic design

Presented aerodynamic design shows the main models' configuration which were being analysed. The results of analysis have an important impact on the subsequent model modifications. Some models are not presented because result were not satisfied.

The design process starts form conceptual model No 1. this configuration has the NACA airfoil and has not an aerodynamic and geometric twist. The analysis The Fig. 7 presents the preliminary configuration.

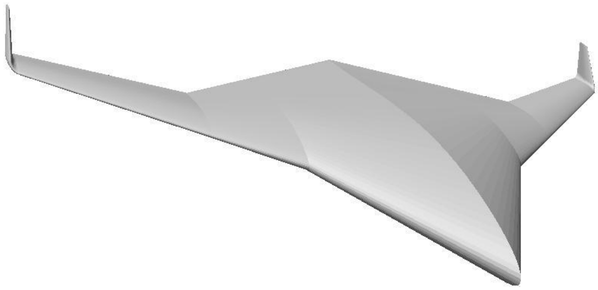

Fig. 7 Model No 1

The modification of the model No 2 was not satisfying requirements. So, the resulting geometry is not presented. The next generation is the Model No 3. The most important modification was implementation of the geometric twist. Fig. 8 presents the airfoil angle incidence linear distribution in a spanwise direction. Note, that negative value of incidence angle means that local an angle of attack increases.

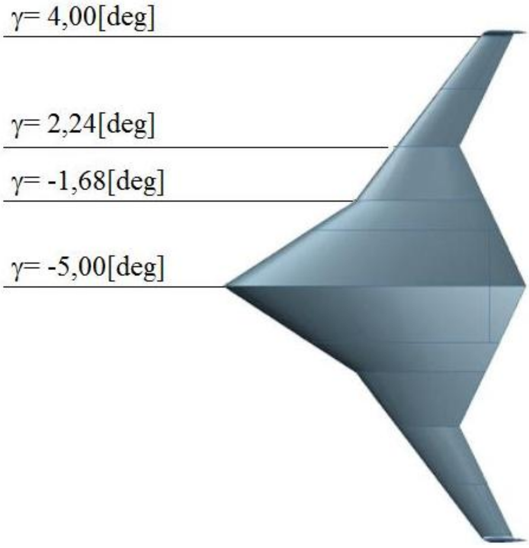

Fig. 8 Model No3

Proposed modifications of the model No 3 still did not gave satisfying results. So, it was decided that the concept of using the NACA airfoil was fulfilled.

The Model No 5 (Fig. 9) was equipped with a new airfoil. The two segmented airfoil described in paragraph 2.2 was used. Moreover, the vertical stabilizers on the tips of the wing were extended above and under the tips of the wing.

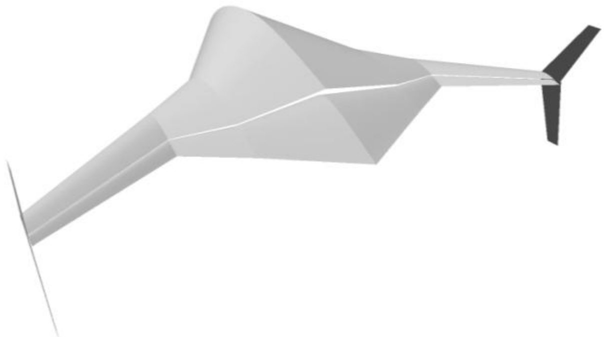

Fig. 9 Model No5 – configuration with two segmented airfoil

The wing had a linear aerodynamic twist. The root airfoil had a 19% thickness ratio but the tip had a 17% thickness ratio.

It was assumed that the aircraft is equipped with three pairs of independent control surfaces. It caused a few geometrical problems with flap deflection. So, the modification of trailing edge of the wing was made. Fig. 10 presents the new geometry of the aircraft (especially the shape of the trailing edge) and the all available control surfaces.

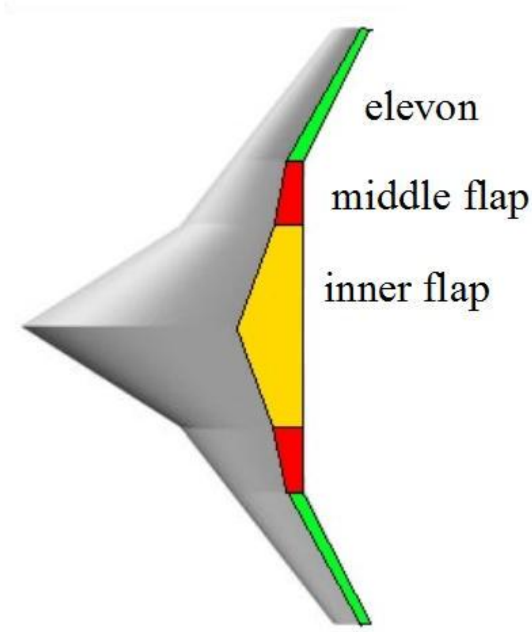

Fig. 10 Model No 6 – the new Trailing edge

Model No 7 (Fig. 11) was improved by adding the non-linear geometrical twist. Presented proposal of the twist distribution comes from aerodynamic analysis of the previous model.

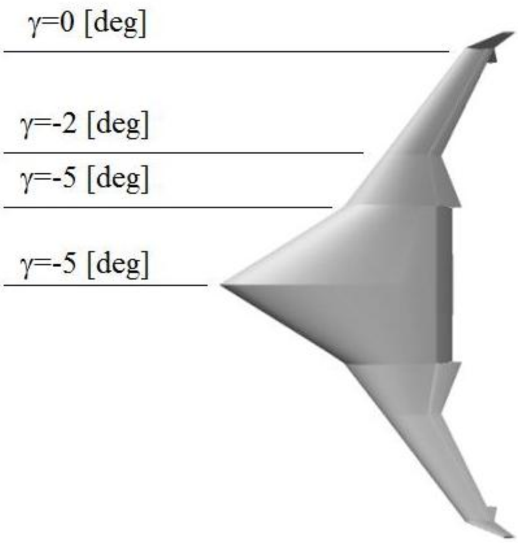

Fig. 11 Model No 7 - the proposal of the twist angel distribution in spanwise

The model No 8 was based on the previous model (Model No7). In this case the effectiveness of the control surface was investigated. Fig. 12 to Fig. 14 present the deflection of the all independent control surface.

Fig. 12 Model No 8 with deflected inner flap

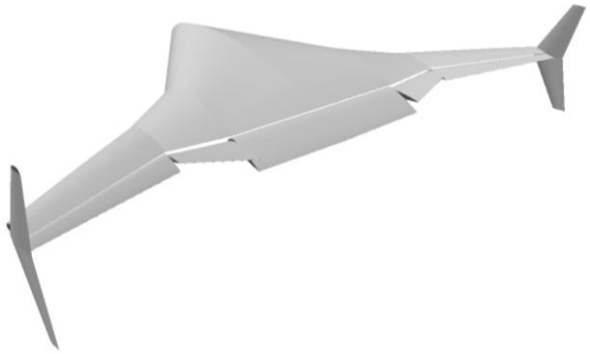

Fig. 13 Model No 8 with deflected middle flap

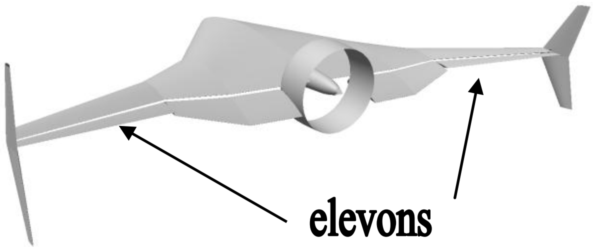

Fig. 14 Model No 8 with deflected the elevons

The last modification of the flying wing model was adding the propulsion. The model presented in the Fig. 15 and Fig. 16 consist of the model of propulsion. The most important modification was reduction of the inner flap span. It was caused by the propulsion location.

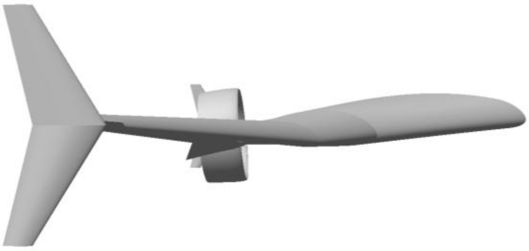

Fig. 15 The side view of the Model No 9 with propulsion model

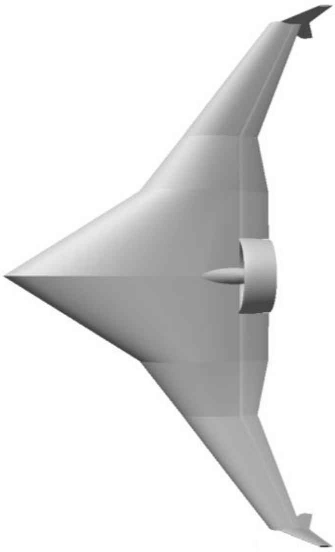

Fig. 16 The top view of the model No 9 with propulsion model

4 Results

4.1 Aerodynamic analysis

An aerodynamic calculation was made only for the one Mach number equal 0,1. The first analyse was made by the simplest panel method.

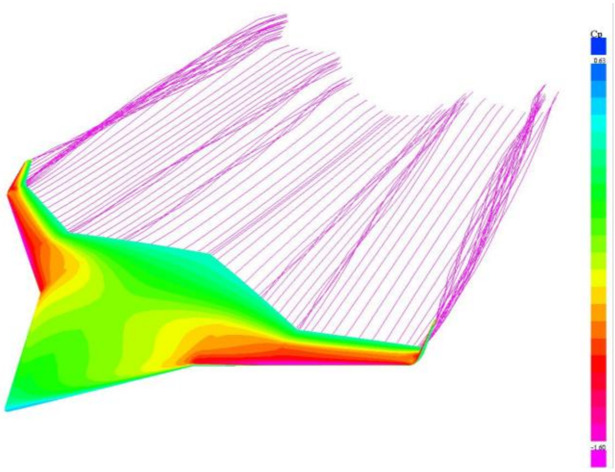

Fig. 17 Cp distribution for the Model No 1

The Results of the preliminary aerodynamic calculation computed by VSAERO software reveal that the maximum lift coefficient is not sufficient. So, the decision about changing the airfoil was taken. The Fig.

18 presents the results of the airfoil's modification.

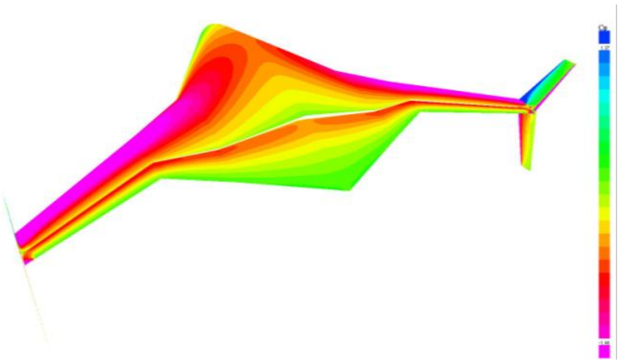

Fig. 18 Cp distribution for Model No 5 – the first model just after change the airfoil, angle of attack α=4 deg

The increase of complexity of the model and the wish to increase the accuracy of calculations, the method of calculation was changed. The further results were obtained by the MGAERO.

The next computational model (Model No 8) was equipped with three pairs of flaps. During analysis effectiveness of all flaps were investigated. Fig. 19 presents the Cp distribution for the model No 8 with deflected the inner flap.

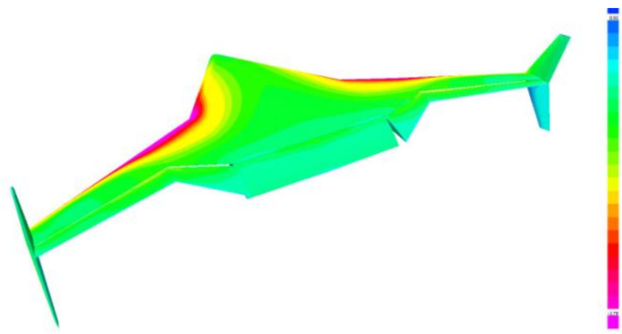

Fig. 19 Model No 8 – Cp distribution for the model with deflected inner flap

The last set of computation was made for the MODEL No 9. As was mentioned in paragraph 2.5, the aerodynamic analysis for the complete aircraft with the propulsion model were made. Moreover, the analyse of effectives of all control surfaces was made too. Fig. 20 presents the Cp distribution of a clean configuration of the aircraft.

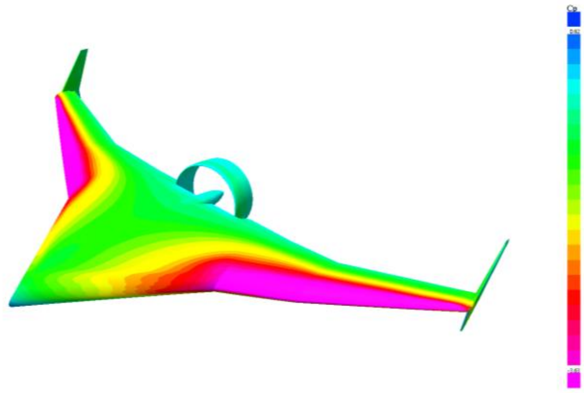

Fig. 20 The CP distribution for the last configuration of flying wing

The aerodynamic characteristic presented on the Fig. 21 and Fig. 22 are results of aerodynamic analyses of Model No 9. The calculation was made for the references value presented in the Table 1 and the pitching moment coefficient was referred to the 33% of MAC.

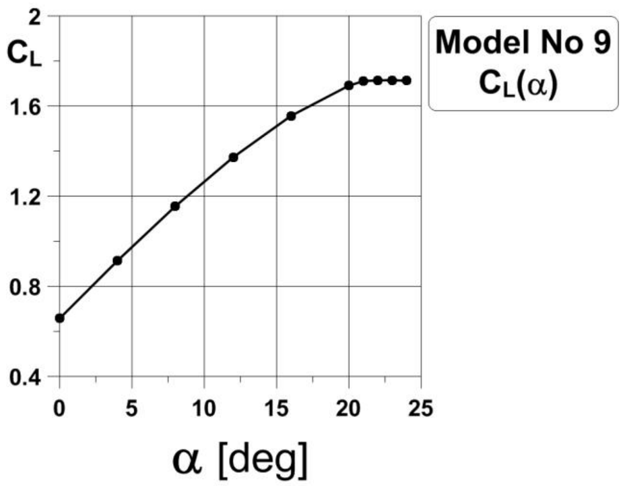

Fig. 21 Lift force vs. angle of attack for the Model No 9.

The maximum lift force obtained for the last model (Model No 9) equals $C_{L\ MAX}$ = 1.72 and the derivatives lift coefficient respect to the angle of attack equals $dC_L/d\alpha$ =3.564 [1/rad].

Fig. 22 and Fig. 23 present the diagrams of pitching moment for versus angle of attack and the drag polar for Model No 9.

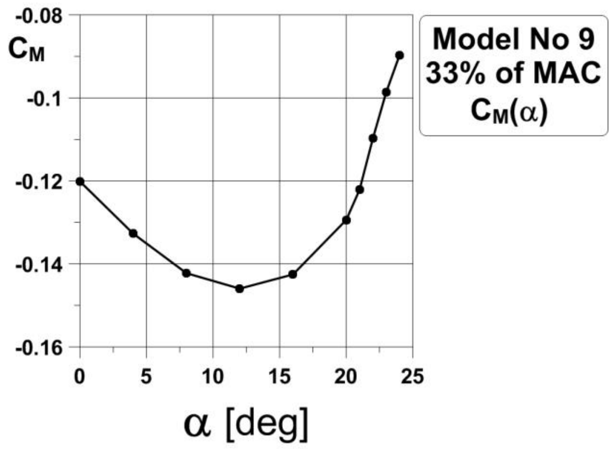

Fig. 22 Pitching moment coefficient vs. angle of attack.

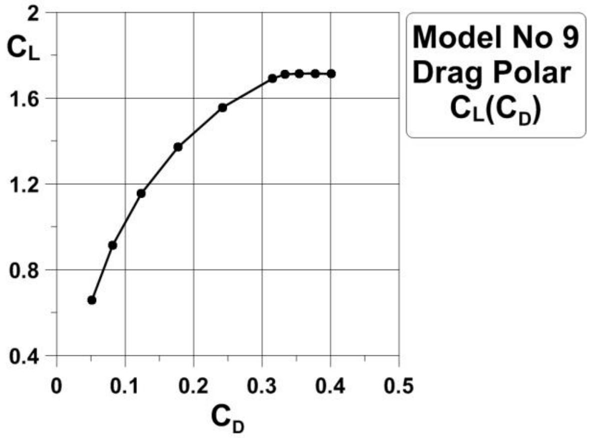

Fig. 23 Drag polar for Model No 9

4.2 Trim condition

The trim condition is a very important issue for the aircraft in the flying wing configuration. Deflection of control surfaces allow to obtain the equilibrium state but causes the lost of the lift force too.

The aircraft was equipped in three control surfaces (Fig. 12,13,14) It was assumed that equilibrium state may be obtained by deflection of all control surfaces. It means that all control surfaces may by deflected in a few quite different configurations e.g. only the inner flap and elevons were deflected.

The optimal deflection of flaps allow to minimize the lost of the lift force. It may be obtained by optimization. Two of objective function were defined. The first one assumed only pitching moment minimization:

$$OF = |C_M| \tag{1}$$

The second one assumed that both the pitching moment and drag force coefficient were minimized:

$$OF = |C_M| + C_D \tag{2}$$

Trim condition analysis was made only for the last model of the aircraft with the engine model (Model No 9). First, the control derivatives like lift force and pitching moment coefficient respect to the flap deflection were calculated. The results are presented in Table 2 and Table 3.

Table 2 Derivatives of lift force coefficient in respect to the flap deflection

Derivative	Value of Derivative
$dC_L/d\delta_{F1}$	0.0079 1/deg
$dC_L/d\delta_{F2}$	0.0078 1/deg
$dC_L/d\delta_{F3}$	0.0145 1/deg

Table 3 Derivatives of pitching moment in respect to the flap deflection

Derivative	Value of Derivative
$dCm/d\delta_{F1}$	-0.0029 1/deg
$dCm/d\delta_{F2}$	-0.0069 1/deg
$dCm/d\delta_{F3}$	-0.0082 1/deg

Fig. 24 and Fig. 25 present the range of deflections of all control surfaces necessary to obtain the trim condition. It is a results of the optimization process for both defined objective functions (eq.1 and eq.2). For the obtained configurations of deflection L/D ratio versus angle of attack and versus lift force coefficient were calculated (Fig. 26 and Fig. 27).

Deflections for all flaps were quite different between the first objective function and the second. In the first case (eq.1), middle flap was the most deflected flap, but inner flap was the most deflected in the second case. Moreover, low deflection of elevons (Fig. 25) was the correct result because elevons may use only to the roll control.

The configuration of deflection for the second case increase the L/D ratio significantly (Fig. 26 Fig. 27). It was caused by decreasing the drag force coefficient versus flap deflection.

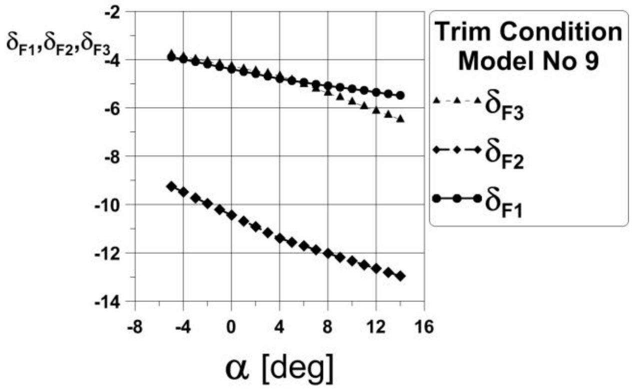

Fig. 24 Deflection of all control surfaces as a result of optimisation for the first objective function (eq.1)

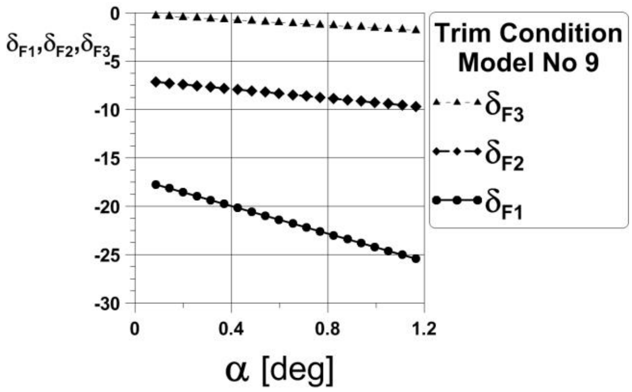

Fig. 25 Deflection of all control surfaces as a result of optimisation for the second objective function (eq.2)

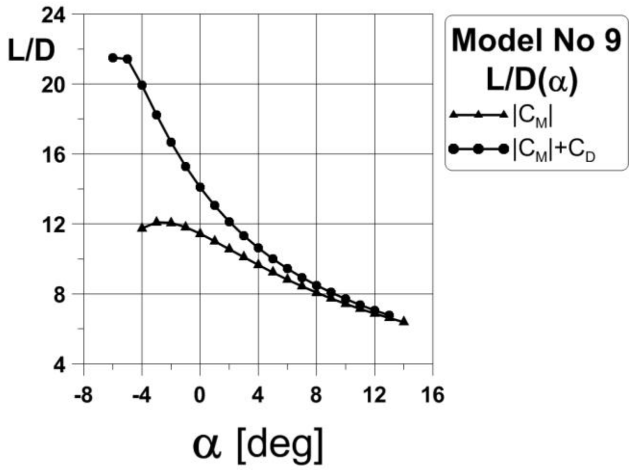

Fig. 26 L/D ratio vs. angle of attack as a results of the pitching moment minimization

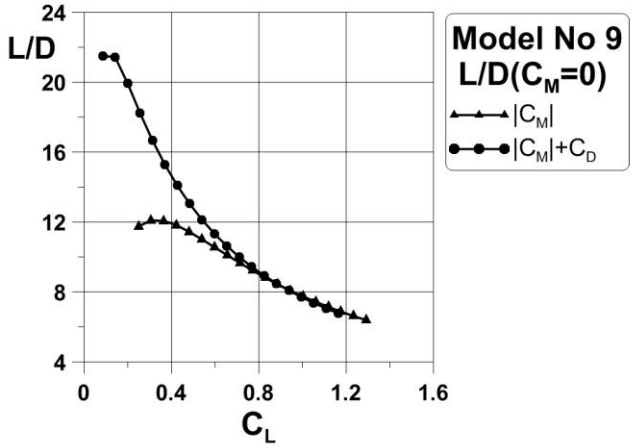

Fig. 27 Comparison L/D ratio vs. lift force

Next, the comparison of the drag polar for results of both objective functions (Fig. 28).

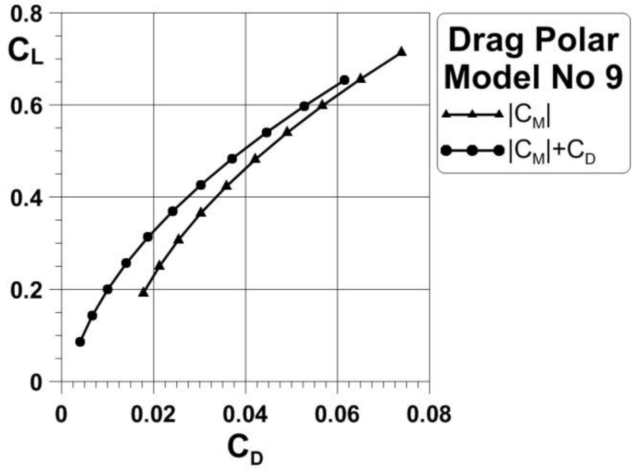

Fig. 28 Comparison of drag polar obtained by first objective function (eq.1) and second objective function (eq.2)

It was assumed in the basic computation case that all flaps may be deflected. Another analysis of trim condition was made for reduced number of control surfaces. The first case assumed deflection of only the inner flap and elevons. For the second case deflection of the inner and middle flap were assumed. The result for first case was compared with the basic case, when all flaps were deflected (Fig. 29). The result for second case was similar to the result of basic case.

The last calculation assumed what was the necessary deflection of single flap to obtained the trim condition. It was made to check the possibility of use control surfaces in emergency case. The Table 4 presents the results of the calculation.

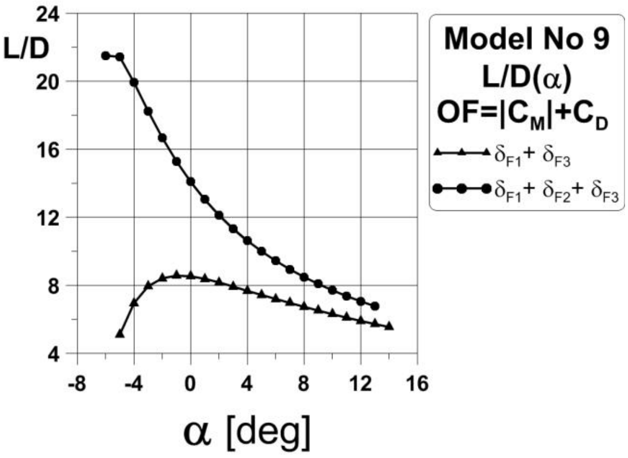

Fig. 29 Comparison of different configuration of flap deflection

Table 4 Trim condition for the single flap deflection for angle of attack α=0[deg]

	inner flap	middle flap	elevon
δ_{F1}	-41.30	0.0	0,0
δ_{F2}	0.0	-17,30	0,0
δ_{F3}	0.0	0.0	-14,73

The single deflection for all flaps is significant high. So, the only a pair of deflected flaps should be used.

5 Conclusions

The paper presents the design process of the aircraft in the flying wing configuration. A lot of models were investigated and a lot of modifications were made. To obtain the higher lift coefficient the two segmented airfoil was used. A few problems occurred with considered solution but all was resolved. The resulting geometry of the aircraft satisfying the project assumptions.

The optimisation was used to find the best deflection of control surfaces. The computation reveal that it is possible to obtained the trim condition and keep the high L/D ratio. However, minimum two from three control surfaces should be used to obtained the trim condition.

It should be noticed, that result of presented aerodynamic analysis is the preliminary configuration of aircraft. Next step should be calculation of the dynamic stability and perform in the wind tunnel tests

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