

# Lightweight unmanned aerial vehicle for emergency medical service – Synthesis of the layout

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## Abstract

The article presents the innovative unmanned aerial vehicle project for emergency medical services. Designed unmanned aerial vehicle combines vertical takeoff and landing characteristics with fast forward flight capability that are vital to perform such an emergency medical mission. The main purpose of the designed unmanned aerial vehicle is to deliver the necessary medical package to the place where access is difficult, and estimated arrival time of conventional ambulance is too long. The cost of the support of such unmanned aerial vehicle could be significantly lower than in case of medical helicopter, which is not necessary in some cases. Designed unmanned aerial vehicle can also be used for fast delivery of essential medical substances (e.g. blood). The selection of configuration was the first and crucial step of the design. After analysis of many different copter configurations, together with selected crash reports analysis, the coaxial quadcopter configuration crossed with conventional airplane was selected. All power units for VTOL capability are electric, and they are doubled for redundancy purposes, with maximum T/W ratio about 2.0. Such configuration allows to sustain a stable flight (vertical phases) in case of one motor failure. Two versions of the vehicle are designed: fully electric (power units for the forward flight and vertical takeoff and landing are electric) and mixed where forward flight unit is a small piston engine. The final layout was the result of conceptual investigation and preliminary research, MDO and trade-off analysis, where as many aspects as possible were considered. The main problem was to meet the vertical takeoff and landing capabilities, relatively long range and endurance, expected payload (3 kg) and the requirement not to exceed 25 kg of maximum take-off weight. Paper presents the design process from initial requirement to the final configuration accepted to be manufactured.

## Keywords

Hybrid UAV, VTOL, MDO, aircraft design, stability

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## Introduction

Currently, the market of small unmanned aerial vehicle (UAVs) (up to 25 kg) is dominated by multicopters. This is due to the ability of vertical take-off and landing and the wide range of such vehicles. The main limitation of multicopters is their range due to the fact that the necessary lift force is created all time, even during steady level flight, by the propellers, which is not energy efficient. A much more efficient method of horizontal flight is classical aircraft in which the lift force is generated on the wing. The disadvantage of the aircraft for typical drone purposes is the need for the runway. Therefore, there are many ideas on how to combine vertical takeoff and landing (VTOL) and horizontal capabilities in one vehicle. A good survey

of such vehicles, often named as a hybrid unmanned aerial vehicle, was presented by Saeed et al.<sup>1</sup> Authors present widely the problems of such vehicles but also express the belief that hybrid UAVs will gain strongly in popularity. They say:

For now, the hybrid UAV development is still in its infancy, and there is a huge space for the hybrid UAVs to become more mature in terms of many

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**Figure 1.** UAV for emergency medical service in action.

**Table 1.** Selected forward flight VTOL configuration.

| Number of propulsion systems                       | Vectoring/combination     | Type               |
|----------------------------------------------------|---------------------------|--------------------|
| One propulsion system for hover and forward flight | Thrust vectoring          | Tilt prop          |
|                                                    |                           | Tilt duct          |
|                                                    |                           | Tilt wing          |
|                                                    |                           | Tilt rotor         |
|                                                    |                           | Tail sitters       |
| Two propulsion systems                             | No thrust vectoring       |                    |
|                                                    | Separate systems          | Lift + cruise      |
|                                                    | Combined system for hover | Lift + lift/cruise |

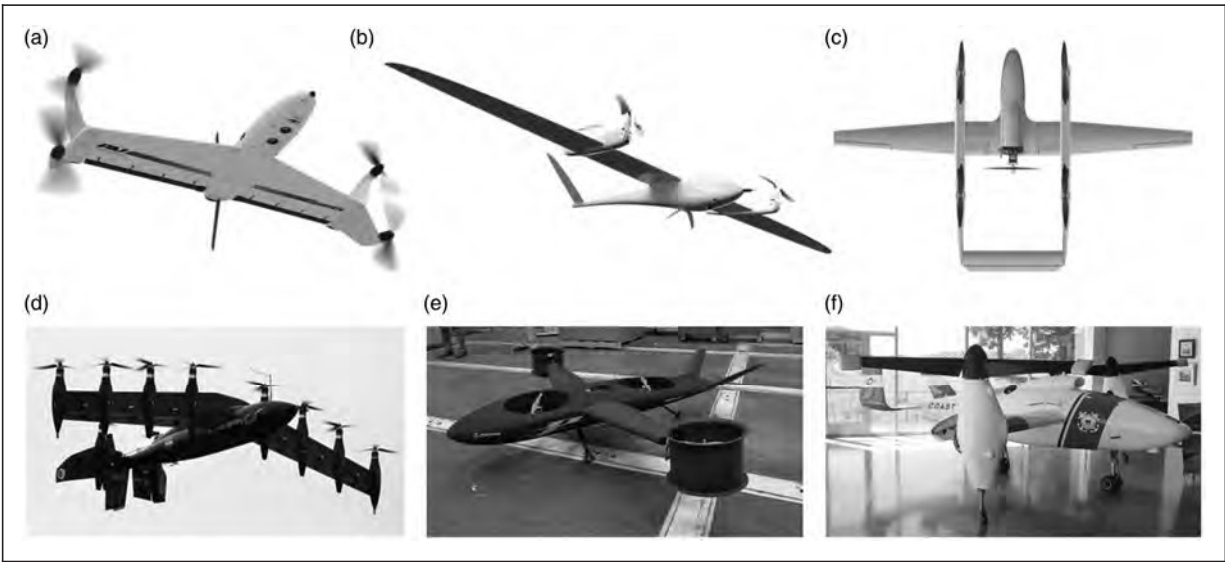


**Figure 2.** UAV selection (Tail Sitter,<sup>4</sup> Tilt Prop,<sup>5</sup> QuadPlane,<sup>6</sup> Tilt Wing,<sup>7</sup> Tilt Duct<sup>8</sup> and Tilt Rotor<sup>9</sup>).

critical perspectives such as design philosophy, dynamics modeling, control, guidance, navigation, and robustness. Nevertheless, given their rapidly growing popularity, it is believed that the hybrid

UAVs will have a bright future and promptly form an essential pillar of the UAV market.

Presented paper contains selected problems related to the design of the hybrid UAV – the configuration

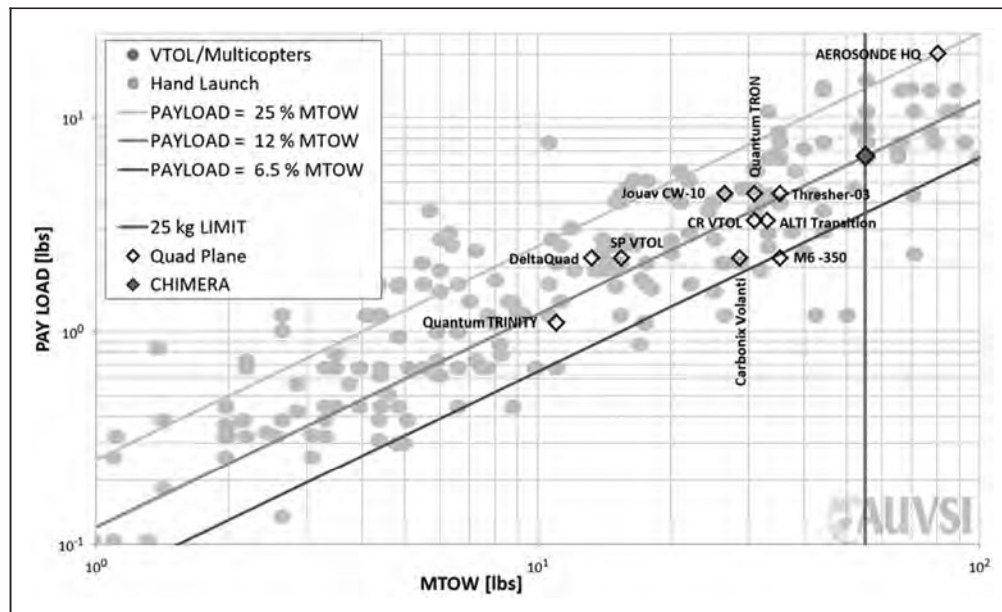


**Figure 3.** Payload versus MTOW trend analysis.

selection, aerodynamic design, stability analysis and some layout considerations. Especially, the following problems had to be considered:

- efficiency of the propulsion system – the power necessary for hovering is about 10–20 times greater than power necessary for a stable horizontal flight, which means that if we have one propulsion system for both the vertical and horizontal phase, it will work with 10% of the nominal power in 99% of the mission time;
- there are many possible configurations<sup>1</sup> and except the efficiency, the safety aspects<sup>2</sup> must be considered;
- adverse effects of VTOL drives on UAV aerodynamics in horizontal flight mode;
- transition phase: aerodynamic interaction between VTOL and fixed wing during the transition flight phase – poorly understood and challenging process.

The vehicle prototype is currently in the manufacturing phase. The test flights with the use of simplified demonstrator were performed successfully. A maiden flight is expected in spring 2020.

### Initial guess

The designed has to incorporate two crucial characteristics, which are necessary to perform emergency medical service. Namely, it has to combine VTOL characteristics with fast forward flight capability. First being an answer to difficult destination (Figure 1) access and second, to requirements for urgent delivery and remote location. Above requirements are only satisfied by a fixed wing (FW) aircraft

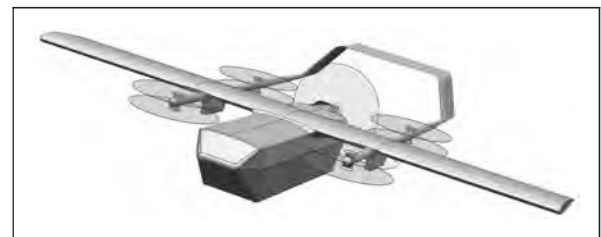


**Figure 4.** Quad plane configuration.

with VTOL capability. Additionally, safety factor required all vital systems redundancy (together with all control surfaces) and moreover parachute rescue system, which further decreased possible configuration selection.

### Fixed wing VTOL aircraft

There are plenty examples of fixed wing aircraft VTOL propulsion concepts<sup>3</sup> which provide a solution to above-mentioned requirements. Unfortunately, these concepts have been designed with human personnel onboard and liquid fuel in mind. There are several factors why they are not easily scalable down to the size of the lightweight UAV. These include, among others, a reduction in the effectiveness of aerodynamic properties due to small Reynolds numbers, the technology of the power plants used and the degree of structure complexity.

All selected scalable down concepts can be divided based on the number of separated propulsion systems for hover and forward flight, possibility of thrust vectoring and number of propulsion systems for hover only as presented in Table 1.

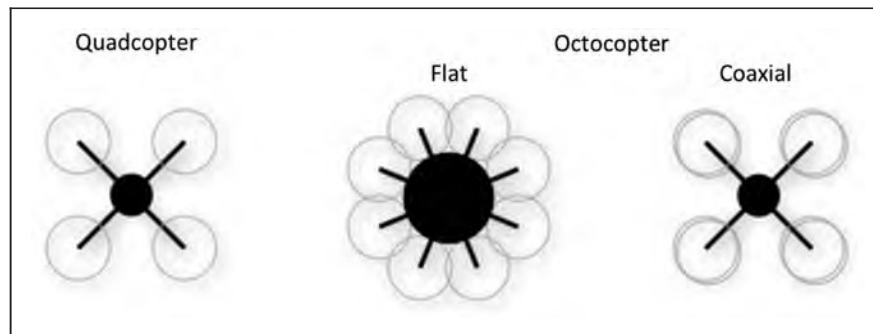


**Figure 5.** VTOL propulsion arrangement.

After investigation of all fixed wing VTOL concepts together with market research of available UAVs, the most promising solution for lightweight UAV with 25 kg maximum take-off weight limit was selected and are presented in Figure 2.

### Fixed wing VTOL configuration

The selection process starts with definition of different thrust requirements for hover and forward flight. To balance aircraft weight in hover, propulsion system has to generate thrust force equal to its weight or bigger to accommodate trimming or wind gusts. In forward flight, where lift is wing generated, required thrust is basically equal to drag force, which usually is about or less than 10% of the aircraft weight. This means that, on average, 10 times more thrust is needed for hover than for forward flight. If the same propulsion system is used for both flight modes, then in forward flight (95% of total flight endurance), only 10% or less of engines' maximum power is used in making the longest flight segment energy inefficient.

The weaknesses and disadvantages of presented in Figure 2 configurations are briefly discussed below:

- Tail sitter and tilt wing: the most prominent feature is a vertical position of wings during takeoff and landing, which is unacceptable during wind and gust conditions or at least makes flight control very complicated and hazardous. Additionally, using only 10% or less of maximum propulsion power results in more mass for energy storage.
- Tilt prop and tilt duct: using only 20% or less of their maximum power for forward flight makes 95% of flight duration inefficient.
- Tilt rotor with variable pitch propellers: one of the most complex kinematic solutions for FW VTOL and complicated flight dynamics. The failure of any link in this kinematic chain will result in disastrous loss of control. There is little space for redundancy in such configuration.

**Table 2.** Recommended values of thrust-to-weight ratio.<sup>10</sup>

| Multicopter         | T/W  |
|---------------------|------|
| Camera platforms    | 1.43 |
| Aerial photography  | 1.67 |
| FPV aerial explorer | 2.00 |
| FPV race copters    | 2.22 |

- All except tail sitter and QuadPlane have motor TILT MECHANISM working under the load, which adds weight and increases the risk of failure during operation.
- QuadPlane: VTOL engines and propellers are not used in forward flight but are left in freestream adding to total drag.

The winning configuration belongs to the Lift + Cruise class, where two separate propulsion systems are used for hover and forward flight. This configuration is sometimes called QuadPlane (Figure 4). It helps to optimize both propulsion systems efficiency for very different work conditions. The selection of the QuadPlane configuration was the first crucial step of the design, and its advantages and characteristics are summarized below:

- Conventional layout design – very well tested
- No moving parts – lower failure risk
- Redundant VTOL propulsion (coaxial)
- Robust flight dynamics
- Relatively simple vertical take-off and landing
- Relatively simple transition phase

### Fixed wing VTOL trend analysis

Before propulsion sizing can be performed, MTOW has to be estimated. Collection of 166 multicopters and 58 small hand launch aerial vehicles are available from AUVSI organization webpage



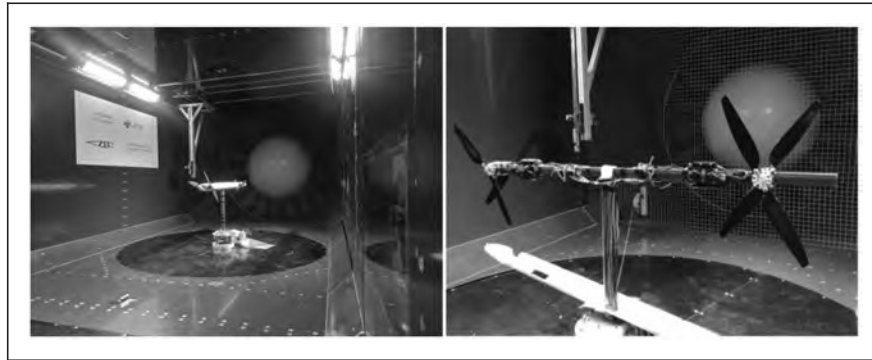


**Figure 6.** Closed-loop wind tunnel with turntable in test section with six-component balance.

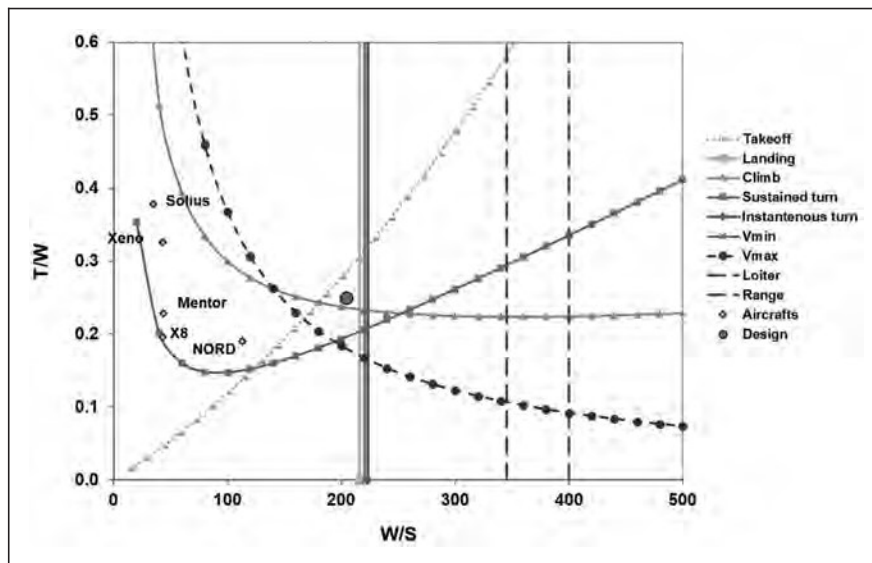


**Figure 7.** Wing loading and thrust loading analysis.

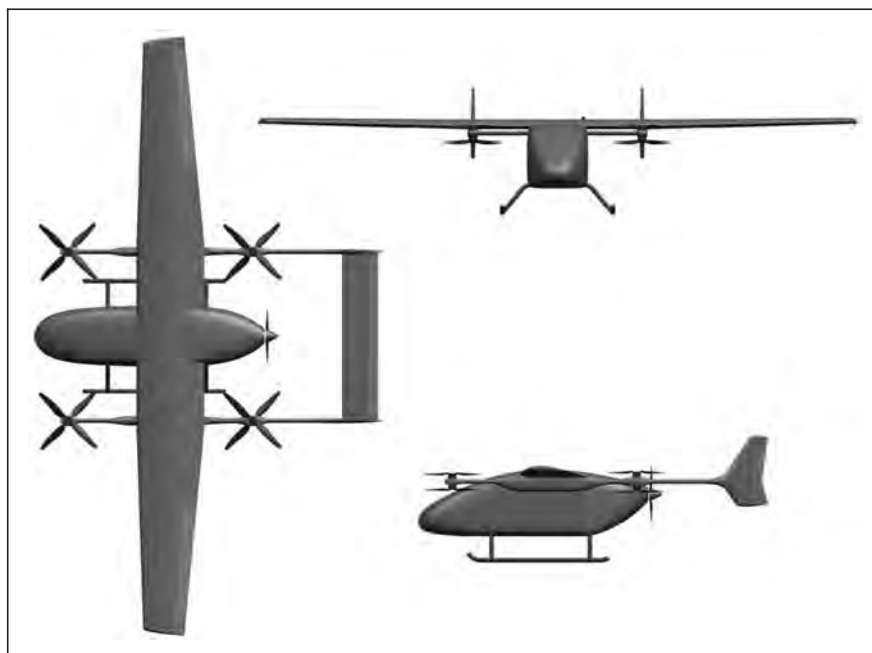


**Figure 8.** Three views of the selected configuration.

(<https://www.auvsi.org>). They are all presented (as the background picture) together with the selection of selected and commercially available QuadPlanes (marked as diamond) in Figure 3.

**Table 3.** Main characteristics of the flight demonstrator and its optimized values.

| Parameter                     | Optimized value |
|-------------------------------|-----------------|
| Cruise speed (m/s)            | 22              |
| Power required for cruise (W) | 374             |
| L/D                           | 14.3            |
| Endurance (h)                 | 1.64            |

The presented data show that the average payload to weight ratio is about 12%, which means that requested payload weight of 3 kg requires 25 kg MTOW, which is also, the limit for such class UAVs defined by Civil Aviation Authority. This was an estimate that has served as an initial value for more accurate mass analysis.

It is worth noting that none of the QaudPlanes presented, weighing up to 25 kg MTOW, exceeds the payload to mass fraction of 17%, and the heaviest of them CW-10 has a payload capacity of only 2 kg. It is as well the maximum payload weight for QaudPlanes in less than 25 kg MTOW class. Considering the facts presented above, the maximum allowable MTOW weight, i.e. 25 kg, was selected.

*Propulsion system*

The selected configuration of aircraft geometry and fixed wing VTOL is composed of two separated propulsion systems: one for forward flight and second for VTOL maneuvers. In order to extend range and duration of forward flight, two versions of propulsion have been designed: fully electric (power units for the forward flight and VTOL are electric) and mixed where forward flight unit is a small piston engine.

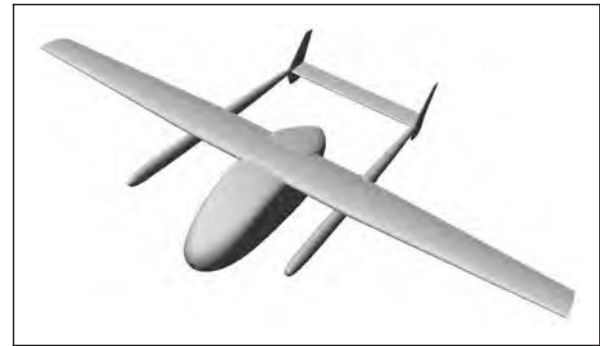


**Figure 9.** The computational model of the PW-CHIMERA.

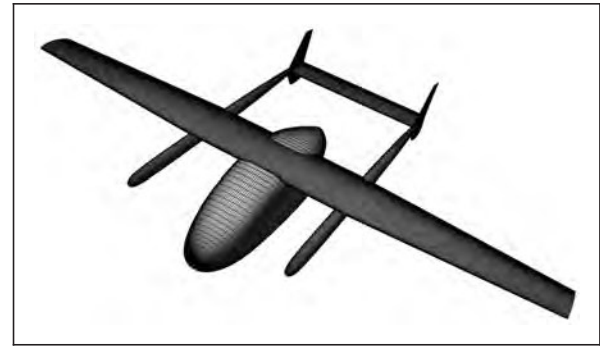


**Figure 10.** Computational mesh – 3582 elements.

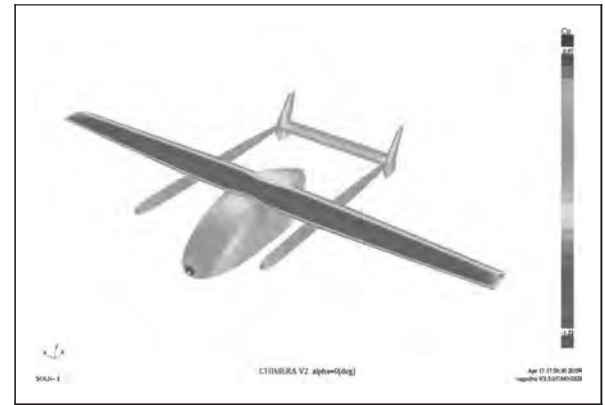


**Figure 12.** The  $C_p$  distribution on the aircraft surface for  $AoA = 0^\circ$ .



**Figure 11.** Control surfaces models: elevator (on the left) and ruder (on the right).

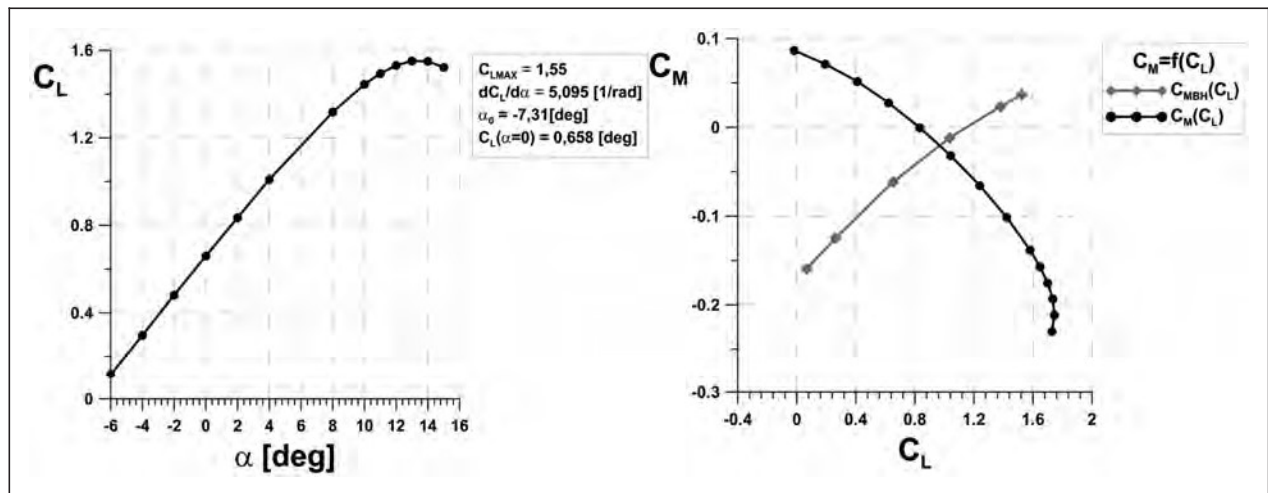


**Figure 13.** The lift coefficient (on the left) and the pitching moment with and without the tail impact (on the right).

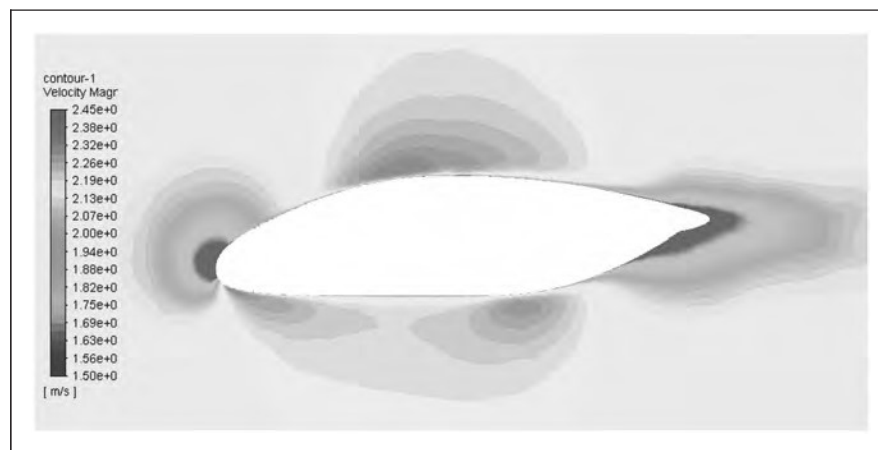


**Figure 14.** The Cp distribution around the fuselage (the flow in the symmetry plane).

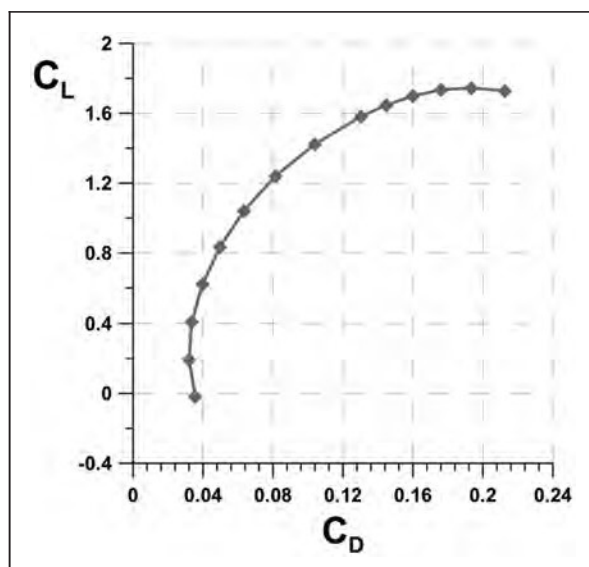


**Figure 15.** Drag polar with fuselage drag correction.

In both versions, identical electric VTOL system is considered.

### VTOL propulsion system sizing

The VTOL system sizing is the single most difficult part of this project. Several possible configurations of multicopters with different engine number have been analysed as a base for VTOL propulsion system initial investigation. Finally, octocopter in coaxial configuration (Figure 5) with contra-rotating pairs of rotors was selected, where VTOL engines were mounted on tail-booms at four points (Figure 4). Such defined VTOL propulsion arrangement allows for following features:

- Redundancy and safety: all thrust engines are doubled
- Smaller diameter propellers result in shorter response time during disturbance
- More compact design and smaller mass

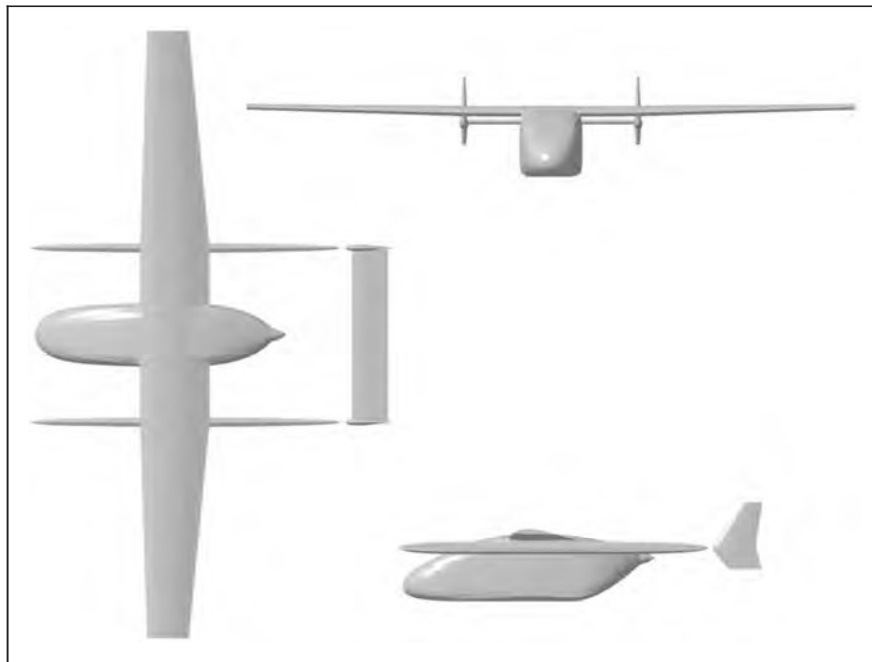


**Figure 16.** Simplified geometry used for stability derivatives computation.

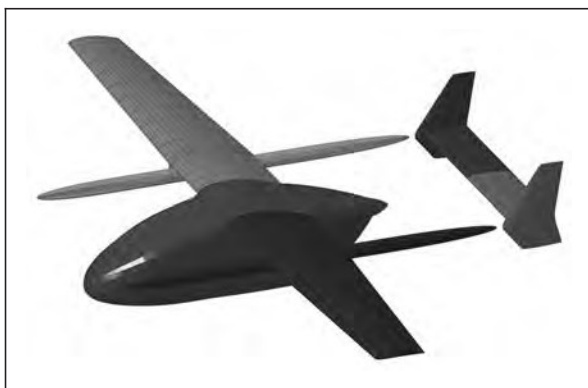


**Figure 17.** Computational mesh and division into main objects.

- Counteracting rotors torque reaction (the opposite torques from the rotors cancel each other out)

Selection of specific VTOL configuration has strong influence on aircraft geometry as sufficient clearance between propellers disks, and all the structural elements have to be provided. That is additionally complicated by the constraint of VTOL centre of thrust co-location with centre of gravity (CG) of the whole vehicle. If not satisfied, VTOL maneuvers would require more power and control would be more complicated.

The sizing of electric motors, propellers and batteries of VTOL propulsion results from the total thrust and time requirements of VTOL maneuvers. This can be expressed in terms of three factors:

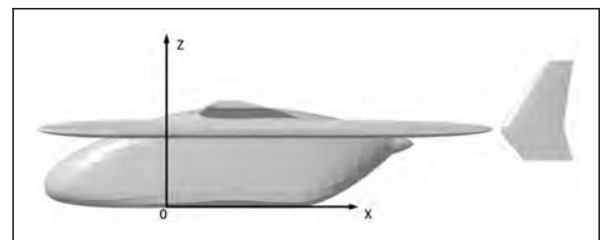


**Figure 18.** The coordinate system assumed to mass and inertia analysis.

- Maximum VTOL T/W ratio (maximum thrust of VTOL propulsion to MTOW) responsible for:
  - Dynamic stability in VTOL maneuvers
  - Vertical/climb speed
  - Mass of VTOL engines, speed controllers and propellers
- Average T/W ratio during VTOL and total time of VTOL maneuvers defines battery weight of VTOL system.

There is no reliable data available of these parameters regarding multicopters with surfaces like wings, elevators and rudders (or QuadPlanes). However, there are recommendations of T/W ratios for multicopters alone given in Table 2.

It was decided to estimate VTOL T/W and time factors from experimental test flights. The initial values were taken from recommendation for multicopter systems and adjusted by the results gathered from Wind Tunnel tests of VTOL testbed (described in the next section).

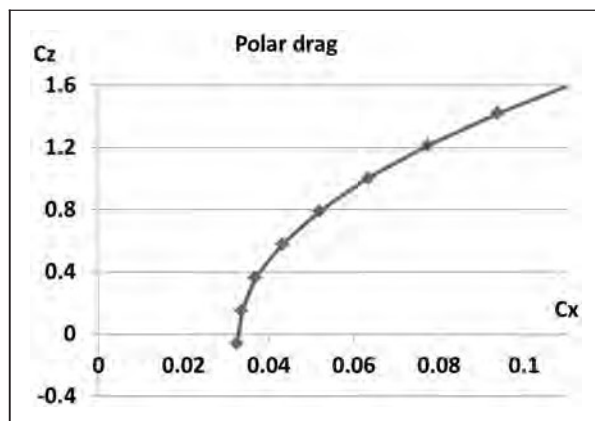


**Table 4.** Selected aerodynamic characteristics versus angle of attack.

| $\alpha$ (°) | $C_x$   | $C_z$    | $C_{xi}$ | $C_m$    | $\partial C_y / \partial \beta$ | $\partial C_l / \partial \beta$ | $\partial C_n / \partial \beta$ |
|--------------|---------|----------|----------|----------|---------------------------------|---------------------------------|---------------------------------|
| −6           | 0.03259 | −0.06162 | 0.00298  | 0.10429  | −0.30223                        | −0.06055                        | 0.03169                         |
| −4           | 0.03361 | 0.15091  | 0.03361  | 0.08325  | −0.3134                         | −0.06828                        | 0.0327                          |
| −2           | 0.03687 | 0.36428  | 0.03687  | 0.05779  | −0.32387                        | −0.07589                        | 0.03421                         |
| 0            | 0.04315 | 0.57783  | 0.04315  | 0.02803  | −0.33357                        | −0.0833                         | 0.03619                         |
| 2            | 0.05191 | 0.79091  | 0.05191  | −0.00589 | −0.34247                        | −0.09046                        | 0.03865                         |
| 4            | 0.06325 | 1.00286  | 0.06325  | −0.04379 | −0.35051                        | −0.09729                        | 0.04154                         |
| 6            | 0.07733 | 1.21301  | 0.07733  | −0.0855  | −0.35766                        | −0.10375                        | 0.04485                         |
| 8            | 0.09382 | 1.42072  | 0.09382  | −0.13081 | −0.36389                        | −0.10978                        | 0.04855                         |
| 10           | 0.11259 | 1.62535  | 0.11259  | −0.17951 | −0.36914                        | −0.11509                        | 0.05292                         |
| 12           | 0.13304 | 1.81901  | 0.13304  | −0.22838 | −0.37386                        | −0.11442                        | 0.06445                         |

**Table 5.** Damping derivatives.

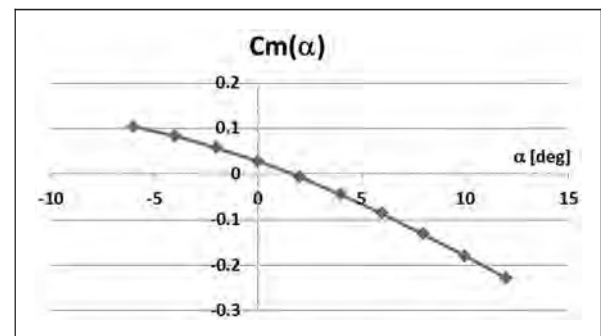
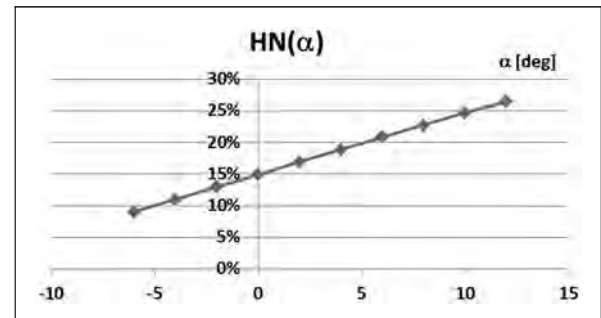
|                                   |          |
|-----------------------------------|----------|
| $\partial C_z / \partial \dot{q}$ | 10.5696  |
| $\partial C_m / \partial \dot{q}$ | −18.7523 |
| $\partial C_l / \partial \dot{p}$ | −0.60883 |
| $\partial C_n / \partial \dot{p}$ | −0.02529 |
| $\partial C_l / \partial \dot{r}$ | 0.00386  |
| $\partial C_n / \partial \dot{r}$ | −0.07636 |

**Figure 19.** Polar drag  $C_z(C_x)$ .

### Forward flight propulsion system

Fixed wing VTOL aircraft has very different take-off (T-O) characteristic from conventional aircraft. There is no T-O run and associated power requirements for roll friction, T-O distance limitation and obstacle clearance. Therefore, engine sizing has been based on required aircraft performance and forward flight drag analysis. There is one more important factor that has to be included, which is the drag of working VTOL propulsion system during transition phase from hovering to cruise velocity.

Forward flight drag estimation has been performed by use of CFD software (ANSYS FLUENT and

**Figure 20.** Pitching moment coefficient versus angle of attack.**Figure 21.** Static margin versus angle of attack.

MGAERO) and was based on derived aircraft geometry. To adjust for VTOL drag, a specially tailored testbed has been made and installed in recently constructed Warsaw University of Technology Wind Tunnel facility. It is equipped with two test sections and the floor turn table.

One tail-boom mockup with four fully functioning VTOL engines has been manufactured and attached to six components balance by the use of separation mast. The testbed has been equipped with a range of electric sensors and investigated in different wind velocities, engines power settings and tail-boom angles of attack. The state-of-the-art automated

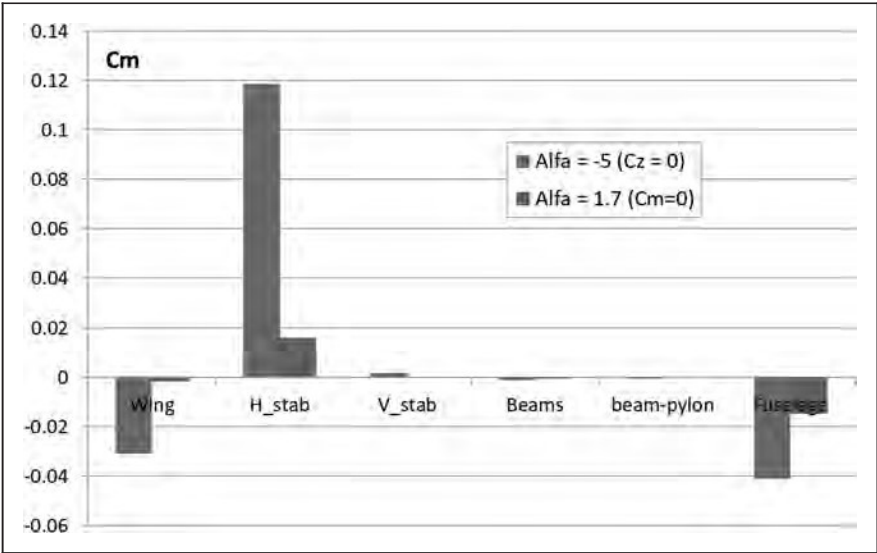


Figure 22. Pitching moment coefficient–contribution of the main aircraft components.

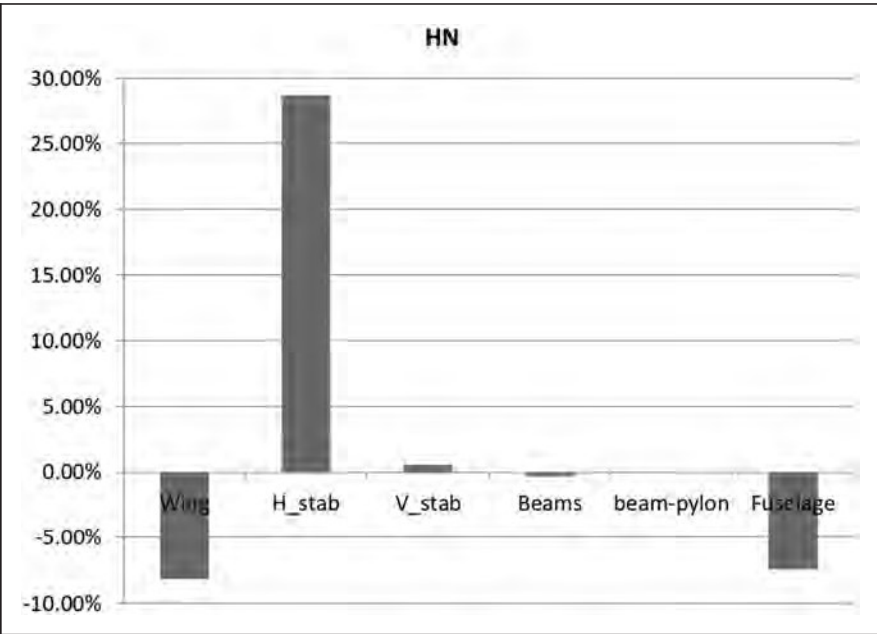


Figure 23. Static margin–contribution of the main aircraft components.

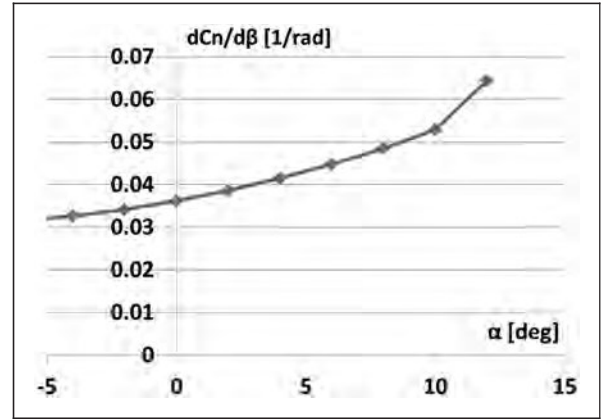


Figure 24. Yawing moment coefficient derivative with respect to sideslip angle.

control and data acquisition system allowed for pre-programmed sequence of experimental configurations together with forces, moments and electric parameters measurement and recording.

The autonomous landing system required wider field of view for sensors and lidars located in nose part of the fuselage; therefore, a pusher propeller configuration has been selected. The forward flight propulsion system is considered in two versions: small piston engine and electric motor. Acquired data from the Wind Tunnel experiment and CFD analysis allowed to accurately estimate the necessary power for forward flight for both versions of engines.

For both types of propulsion, the thrust to weight ratio (T/W) has to be determined within initial design

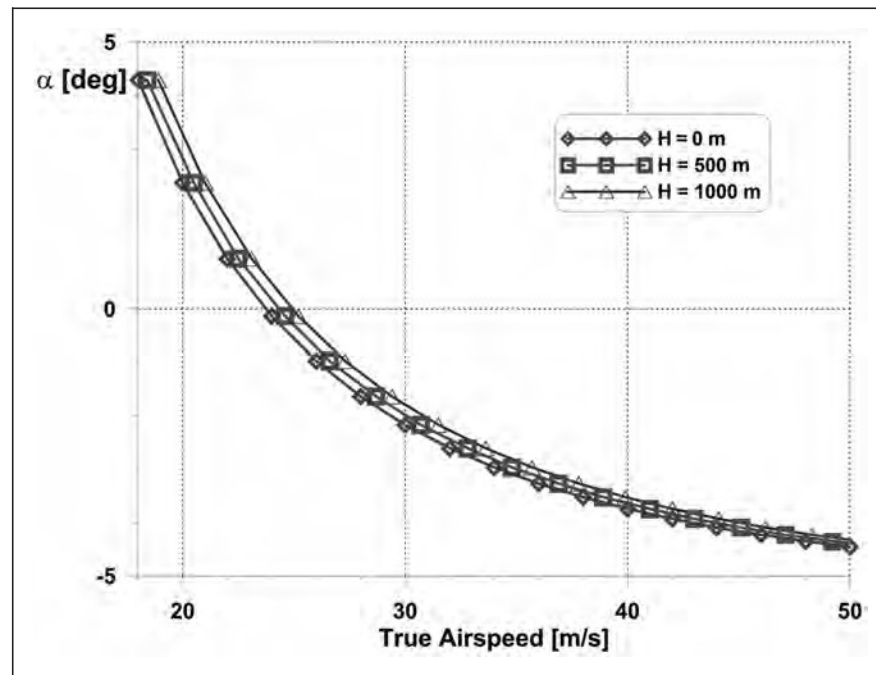


**Figure 25.** Angle of attack needed to trim versus airspeed and altitude.

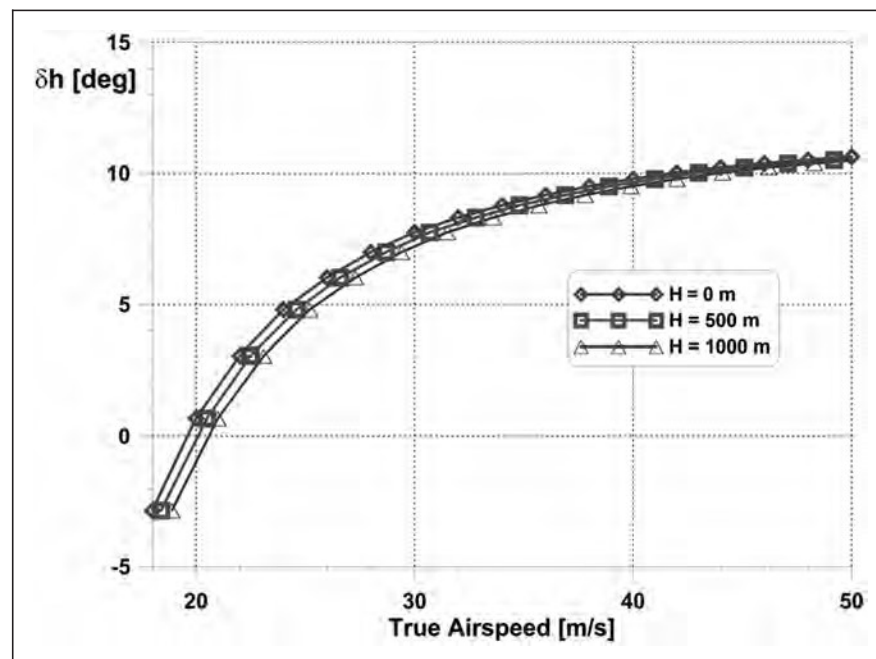


**Figure 26.** Elevator deflection needed to trim versus airspeed and altitude.

process. After setting-up basic design requirements, initial thrust and wing loading analysis were done for different flight conditions (Figure 7). Wing loading did not pronounce any design problems. The most crucial conditions for the thrust loading were takeoff, climb and maximum speed conditions. Since the aircraft is VTOL type, we assumed that it will always takeoff and land vertically, and the classical airplane takeoff condition can be relaxed. The other two restrictive requirements

were satisfied by limiting climb angle to  $8^\circ$  and maximum speed to 35 m/s, which are considered satisfactory for the designed aircraft.

### Aerodynamic analysis and optimization

The result of conceptual design stage was the quadcopter with four doubled electric motors with coaxial propellers for vertical flight and one engine/electric

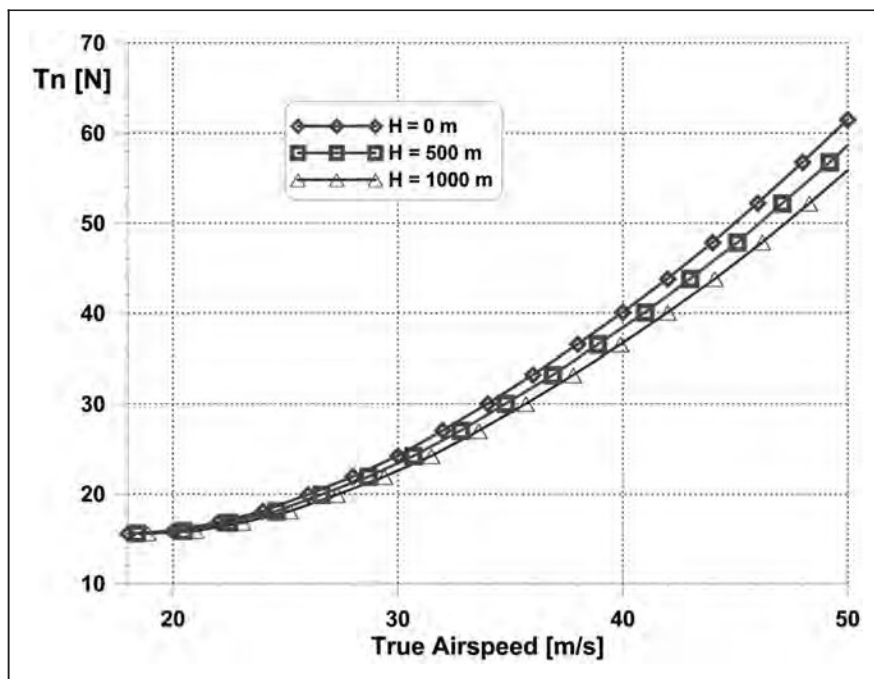


Figure 27. Thrust needed to trim versus airspeed and altitude.

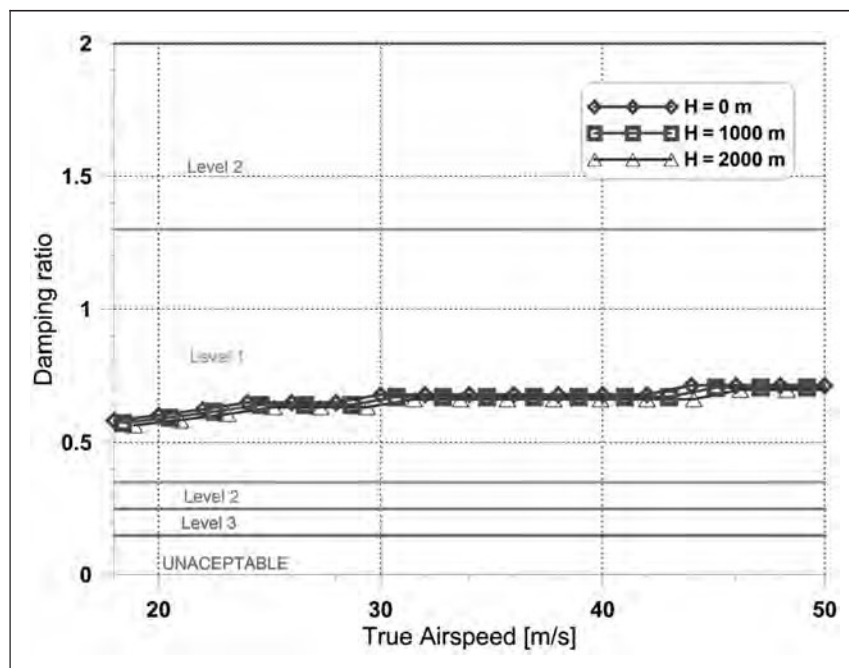


Figure 28. Damping ratio of Short Period oscillation against the background of MIL-F8785C specification.

motor in pusher configuration (Figure 8). The main geometry characteristics are given below:

- Wing area (S)  $1.2 \text{ m}^2$
- Wing span (b)  $3.5 \text{ m}$
- Mean aerodynamic chord (Ca)  $0.351 \text{ m}$
- Fuselage beams spacing  $1.0 \text{ m}$

After the initial analysis, the VTOL aircraft was optimized with medium fidelity analysis software,

which provided sufficient accuracy and limited computational cost. Similar procedures, although using more advanced methods (higher fidelity), are presented in Klymczyk and Goraj.<sup>11,12</sup> The numerical optimization included aerodynamic analysis, mass estimation and flight dynamics analysis. The aerodynamic analysis for the optimization was done with the use of the first-order panel method Panukl,<sup>13,14</sup> coupled with two-dimensional viscous analysis of an airfoil (Xfoil<sup>15</sup>) for better drag estimation. After the

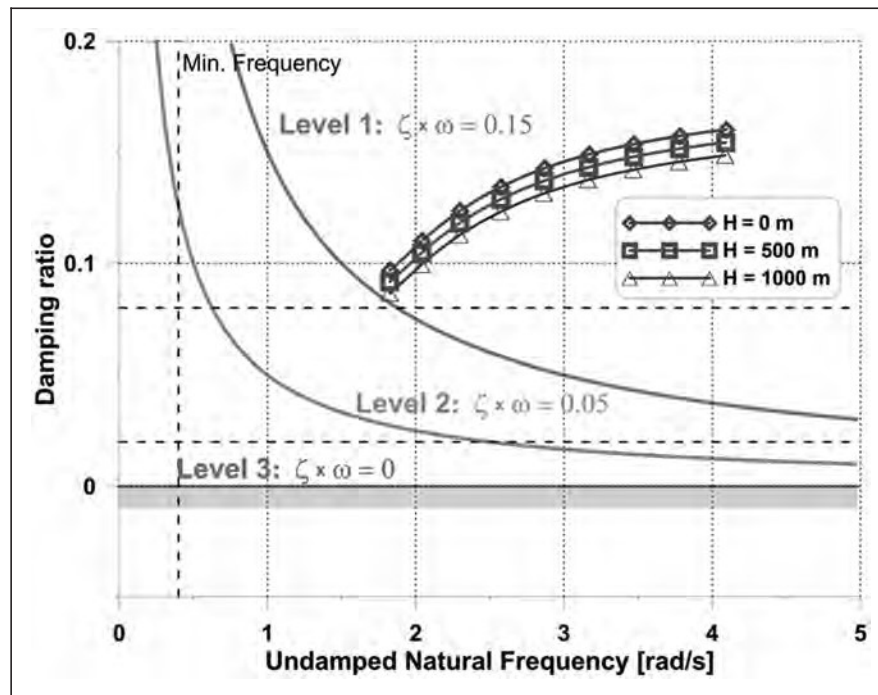


Figure 29. Dutch roll characteristics against the background of MIL-F8785C specification.

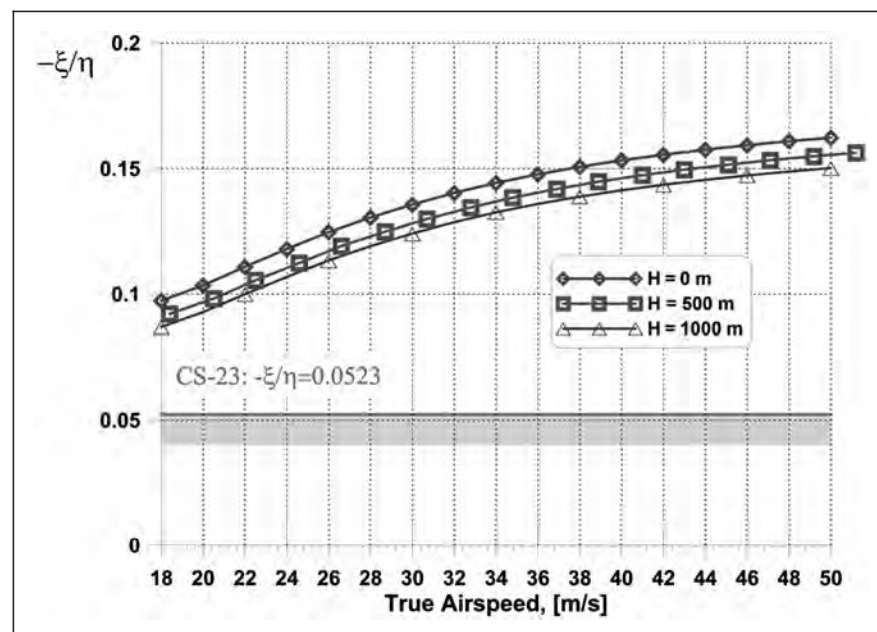


Figure 30. Dutch roll characteristics versus airspeed against the background of CS-23 criterion.

analysis, the airfoil S4062-095-87 was recommended, which has the best characteristics of aerodynamic efficiency and energy function affecting the range and duration of flight. In addition, the airfoil has a moderate pitching moment that will not increase the trim drag. Deterioration of aerodynamic characteristics as a result of dirt and finite trailing edge thickness is small and acceptable. The maximum lift coefficient is 1.6, and the stall characteristics is very mild.

To conduct dynamic stability analysis, mass properties were also needed. Total mass consisted of many components, few of the components were kept constant, and few were dependent on the aircraft's geometric optimization variables. From the initial analysis, it was clear that high total mass of the aircraft is favourable to operate with specified payload, but the design was limited in total mass of 25 kg. It was related to the flight law regulations, which are





**Figure 31.** PW-Chimera – internal layout.

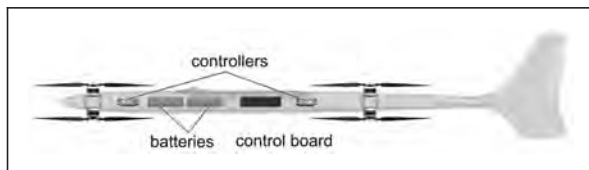


**Figure 32.** Fuselage beam internal layout.

much more restrictive if the mass to fly of an aircraft exceeds 25 kg. Flight dynamic stability analysis during optimization was done basing on the data obtained from the aerodynamic simulations and the mass model. The analysis was done by the use of SDSA software.<sup>16,17</sup> The procedure and the algorithm of the numerical optimization was described in detail by Mieloszyk and Goetzendorf-Grabowski.<sup>18</sup>

The main objective function of the optimization was flight endurance. Additionally, number of constraints had to be satisfied including dynamic stability constraints, which were defined according to the airworthiness regulation<sup>19</sup> and the MIL-STD norms.<sup>20</sup>

Obtained main characteristics of the aircraft after the optimization are shown in Table 3. The aircraft requires 374 W of power for flight in cruise conditions. It has lift to drag ratio 14.3 and is able to achieve flight endurance of about 1 h 40 min.

### Model

The main goal of aerodynamic analysis was to verify the geometry optimization result by the more advanced method of CFD. First of all, the model of the aircraft was prepared. The model includes all main parts of the aircraft. It was assumed that all analyses were made without the vertical and horizontal power units. Therefore, the propeller impact on the beam and the fuselage on the results were not included. Figure 9 presents the PW-CHIMERA aircraft model.

### Methodology

The numerical computation was made by the MGAERO<sup>21</sup> package, tested within the previous projects<sup>22</sup> and by the Ansys Fluent. First method is



**Figure 33.** The mock-up of PW-CHIMERA.

multigrid method which solved the Euler equation.<sup>23</sup> The method combines the low computational cost with accepted accuracy. Figure 10 presents the computational mesh prepared for aircraft model.

The second method was used only for the drag analysis of the fuselage. The fuselage shape is very complicated, so a more advanced method was necessary. To do this, the Ansys-Fluent<sup>24</sup> was used. The software used the RANS method.<sup>25</sup> The analysis was enhanced by GEKO turbulence model.<sup>26</sup> During computation, the pusher propeller impact was not included in the model.

### Computation

Computation by MGAERO was made for constant flight speed equal to 22 m/s and for different angle of attack  $\alpha$  and sideslip angle  $\beta$ . Additional analyses were made for the aircraft configuration without horizontal and vertical stabilizers. Moreover, the analysis was made for isolated tail with deflected control surfaces, elevator and ruder. Figure 11 presents the aircraft tail with highlighted control surfaces. The main goal was to obtain the control derivatives and the hinge moment coefficient for all control surfaces.

### Results

As a result of the aerodynamic computation, the  $C_p$  distribution on the aircraft's surface was acquired (presented in Figure 12). Next, the basic aerodynamic coefficients were computed (Figure 13).

Drag coefficient computed by the use of FLUENT was significantly greater than the value calculated by the use of MGAERO code. The FLUENT computation was highly time-consuming, but was necessary because of the complicated shape of the fuselage tail section. Figure 15 presents the drag polar for PW-CHIMERA aircraft. The graph presents the corrected drag coefficient.

### Conclusions

The optimized geometry of the aircraft was verified by the numerical aerodynamic analysis. The CFD analysis confirmed the correct optimization results. Moreover,

the results were used to perform stability analysis and to verify longitudinal and lateral control efficiency. The results were additionally utilised to evaluate the aircraft performance such as range and endurance.

### Stability analysis

The designed configuration was initially analysed from the point of view of aerodynamic, mass and flight qualities characteristics. The configuration was optimized for flight endurance. This chapter contains the results of aerodynamic and dynamic calculations of the final version of the aircraft, taking into account the previous analyses and the need to adapt to technological requirements.

The aerodynamic analysis including the stability derivatives was performed by the use of PANUKL package.<sup>13</sup> The geometry has been simplified to make the computation process more efficient while meeting accuracy requirements. The simplified model of analysed configuration is presented in Figure 16.

In order to analyse the impact of the main components of the aircraft on the pitching moment and stability characteristics, the computational model was divided into five main objects (wing, horizontal tail, vertical tail, fuselage and fuselage beams).

### Mass and inertia

Mass and inertia characteristics were determined assuming that most of the components are the point masses. Only the largest components of the aircraft were modelled using simple solids to map the inertia of the elements. A homogeneous bar model was adopted for panels (main wing, horizontal tail, vertical tail) and for fuselage beams. For the fuselage and cargo, a cuboid model was adopted. The gravity centre coordinates and inertia moments, in the assumed coordinate system (the origin is located at the bottom of the front main fuselage rib; Figure 18), are as follows

$$\begin{aligned} x_{CG} &= 0.2693, & z_{CG} &= 0.2212\text{m} \\ J_x &= 8.8488, & J_y &= 7.7573, & J_z &= 13.3231, \\ J_{xz} &= 1.7198\text{kgm}^2 \end{aligned}$$

### Aerodynamic characteristics

The following are the aerodynamic characteristics of the aircraft, together with aerodynamic derivatives. Reference area and dimensions are defined above. The angle of attack is measured relative to the  $OX$  axis of the adopted coordinate system (Figure 18). Aerodynamic moment components are calculated relative to the centre of gravity defined above. Derivatives relative to angular rates were determined assuming rotation

around the centre of gravity. Stability derivatives are dimensioned according to the rules:

- the relations between the components of forces and moments of aerodynamic forces and their dimensionless coefficients are defined as follows

$$X = 0.5\rho S V_A^2 C_x, \quad Y = 0.5\rho S V_A^2 C_y, \quad Z = 0.5\rho S V_A^2 C_z \quad (1)$$

$$\begin{aligned} L &= 0.5\rho S b V_A^2 C_l, & M &= 0.5\rho S C_a V_A^2 C_m, \\ N &= 0.5\rho S b V_A^2 C_n \end{aligned} \quad (2)$$

- dimensionless angular rates and accelerations with respect to which damping derivatives are defined and are expressed by relations:
  - in symmetrical case

$$\hat{q} = \frac{QC_a}{2V_A}, \quad \hat{\alpha} = \frac{\dot{\alpha}C_a}{2V_A} \quad (3)$$

- in unsymmetrical case

$$\hat{p} = \frac{Pb}{2V_A}, \quad \hat{r} = \frac{Rb}{2V_A}, \quad \hat{\beta} = \frac{\dot{\beta}b}{2V_A} \quad (4)$$

where

$X, Y, Z$  – components of aerodynamic force (N),  
 $L, M, N$  – components of aerodynamic moment (Nm),  
 $V_A$  – airspeed (m/s),  
 $C_x, C_y, C_z$  – dimensionless coefficients of aerodynamic forces,  
 $C_l, C_m, C_n$  – dimensionless coefficients of aerodynamic moments,  
 $\rho$  – air density ( $\text{kg/m}^3$ ),  
 $P, Q, R$  – angular rates (rad/s),  
 $\alpha$  – angle of attack,  
 $\beta$  – sideslip angle,  
 $\hat{p}, \hat{q}, \hat{r}$  – dimensionless angular rates,  
 $\hat{\alpha}$  – dimensionless time derivative of angle of attack  
 $\hat{\beta}$  – dimensionless time derivative of sideslip angle.

The computations were made with the PANUKL package in the incompressible range in the scope of angle of attack ( $-6, 12$ ). Aerodynamic characteristics versus angle of attack and lateral derivatives (with respect to sideslip angle) versus angle of attack are shown in Table 4. The values of damping derivatives are presented in Table 5.

Figure 19 presents the drag polar curve and Figure 20 presents the pitching moment coefficient versus angle of attack. Pitching coefficient decreases with the angle of attack which means that the aircraft is statically stable. Static margin versus angle of attack is presented in Figure 21.

Analysis of the variation of the pitching moment as a function of the angle of attack allows to state a satisfactory value of the static margin. The next charts show the contribution of particular parts of the aircraft (wing, fuselage, tail, etc.) in the characteristics of the pitching moment and static margin. As can be seen from the charts (Figures 22 and 23), the main share has main wing, fuselage and horizontal tail.

Figure 24 presents the derivative of yawing moment coefficient with respect to sideslip angle versus angle of attack. The positive value does mean that aircraft is statically directionally stable.

### Longitudinal trim and static margin

The first step to determine the dynamic stability characteristics is the equilibrium analysis. The following figures present the results of trim analysis: angle of attack, elevator deflection and thrust needed to satisfy steady level flight. The calculations were made by solving the nonlinear equations of motion with assumption of no accelerated flight, which means that the sum of forces and moments acting on the aircraft is equal to zero.

The results show that all parameters are in satisfactory range. Angle of attack (Figure 25) is in range  $(-5, 5)^\circ$  which corresponds with  $(-2, 8)^\circ$  for main wing – the wing incidence angle is  $3^\circ$ . The elevator deflection (Figure 26) is also in the range  $(-3, 10)$  which allows to have a reserve for control; however, it is advisable to set the horizontal tail at a small angle of incidence to reduce the elevator deflection close to zero for cruising speed (22–25 m/s). The necessary thrust (Figure 27) reaching 60 N for high speed corresponds to T/W ratio calculated within wing and thrust loading analysis (Figure 7).

### Dynamic stability analysis

The dynamic stability characteristics were determined using the SDSA package.<sup>17</sup> The necessary set of aerodynamic data was calculated by the PANUKL package.<sup>13</sup> All modes of motion show correct characteristics and are sufficiently stable. The charts below present the flying qualities of two oscillating modes that have to be strongly damped, i.e. Short Period and Dutch roll against the background of the stability criteria listed in the regulations CS-23<sup>19</sup> and military specification MIL-F8785C.<sup>20</sup>

The short period oscillations connect rapid changes of angle of attack with pitch rate. The period is usually very short. The requirements according to CS-23.181 say: ‘Any short period oscillation not including combined lateral-directional oscillations occurring between the stalling speed and the maximum allowable speed appropriate to the configuration of the aeroplane must be heavily damped...’. The results of computation show that the short period oscillations are well damped (Figure 28) in the whole range of airspeed and is classified in the first level of acceptance.

The requirements for Dutch roll are precisely defined in CS23<sup>19</sup> regulation and says: ‘Any combined lateral-directional oscillations (“Dutch roll”) occurring between the stalling speed and the maximum allowable speed appropriate to the configuration of the aeroplane must be damped to 1/10 amplitude in 7 cycles...’. Figure 30 shows that this requirement is fully satisfied. Also, the criterion by MIL specification<sup>20</sup> is well satisfied, and, as it is shown in Figure 29, it is located in the first acceptance level.

### Final layout

The analyses presented above allowed to design the final configuration and structure of the aircraft. The information of the structure and distribution of individual components and systems enabled verification of weight and mass distribution. The key issue was to agree on the position: the centre of gravity of the whole UAV, the centre of gravity of the payload and the centre of the lift force provided by the VTOL propulsion system. Therefore, the mounting points of VTOL motors form a square, in the centre of which is the centre of gravity of the entire UAV and payload. Other elements that can influence the location of the centre of gravity (as fuel for the engine) are close enough to it that they do not cause significant changes in the dynamic stability characteristics and do not significantly affect the asymmetry of VTOL engines thrust in hovering.

The fuselage nose contains a space of relatively big volume that allows to mount all onboard devices and parachute recovery system. The battery packs for the VTOL drive system were placed in the fuselage beams (Figures 31 and 32), in direct vicinity of electric motors, to reduce the total weight of the cables and the voltage losses. The selected batteries were used in the configuration of two packs connected in series in order to limit the cross-section of the fuselage beam and thus, aerodynamic drag. In addition, the placement of batteries in the fuselage beams at the initial design phase allows for a relatively easy change of their location in order to achieve the possible change in required CG location.

### Concluding remarks

Designing and developing this hybrid UAV is a major challenge because of the many problems that need to be addressed. The most important problems that were considered and solved are:

- Providing adequate payload – the trend analysis of the payload share in MTOW of available commercial and experimental QuadPlane shows that payload about 12% of MTOW is an ambitious but achievable target. For 25 kg MTOW, that target results in required 3 kg payload. The required volume of the payload space resulted in the aerodynamic design that satisfies minimum drag and gives a good efficiency of pusher propeller.



- The negative selection process of different fixed wing VOTL configuration has been presented, and revealed that QuadPlane is the best configuration for specified requirements. Moreover, two different QuadPlane engine configurations were compared, namely quadcopter versus coaxial octocopter and its effect on aircraft geometry.
- To design the effective control during transition phase, the detailed data of aerodynamic characteristics with VTOL engines running is required. As the result of preliminary project investigation, a dedicated stand was created for testing semi-span UAV models in the WUT wind tunnel. The stand is based on a six-element aerodynamic balance attached to a turntable in the tunnel. The state-of-the-art-automated Control and Data Acquisition System allows for the integration of the wind tunnel parameters and turntable position with measurement and model configuration actuators/drives.
- The optimization process and the stability analysis allow to receive the most effective configuration and aerodynamic design.

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