SOLVER PREPARATION FOR RELIABLE AERODYNAMIC COMPUTATIONS OF MICRO UAV

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Abstract. The main goal of this paper is to introduce way of preparation to reliable aerodynamic computations. Starting with geometry creation, solver settings adjustment to validation of aerodynamic results obtained. The issue of the aerodynamic analysis is an untypical micro UAV. The delta wing aircraft, has propeller placed in a slot in the middle of the wing. Such an unusual configuration has interesting capabilities, not fully understand yet. Getting to know basic laws ruling the flow and future intent do develop new generation of the aircraft, requires reliable results, achieved from aerodynamic simulations. Earlier wind tunnel tests of the micro UAV provide data to which CFD computations can be compared. Test case from experiments, similar to the problem that will be solved in the future, increases reliability of computations. The closer the results are, between computations and experiment, the more engineer can rely on it. Geometry model prepared for the computations has to mimic real object very well, but some simplifications are indispensable, to avoid problems with grid generation. Way of geometry obtaining and impact of the simplifications made is shown. Also effect of the solver settings adjustment is demonstrated, on accuracy and speed of computations. The present work considers all mentioned issues, resulting in procedure for solver validation for reliable aerodynamic computations.

Keywords. aerodynamic computations, micro UAV, solver settings, solver results validation

1 Introduction

The mini UAV, called “Bee”, has delta wing configuration and a slot in the middle of the wing. In front of the wing Leading Edge eXtension (shortly LEX) was mounted, which is normally seen only in military fighters. The idea is, to take advantage of the vortex flow generated by LEX, at high angles of attack [1, 2, 3, 4]. Inversely than it is in military fighters, mini UAV-s should fly straight and stably for most of the mission time. It is hard to fulfill this requirement, due to high sensitivity of a small aircraft to any occurrence of disturbances. Very sensitive aircraft, responding to every disturbance causes many problems. Pictures taken from installed cameras without active stabilization will be blurred and measurements may contain large errors. The most common reason of disturbances are wind gusts [5]. To avoid rapid changes of position of an airplane, resulting from disturbances, good maneuverability capabilities and fast autopilot can actively dump the disturbances. Although, UAV flies straight, vertical gusts simulate aerodynamic conditions at high angles of attack. It means that features for that kind of flight are needed, such as LEX, to maintain good maneuverability [6]. Using LEX unable to mount propeller on the nose of the aircraft, because it would destroy vortex flow. On the other hand pusher propeller is dangerous for an operator who is hand lunching the mini UAV. Unfoldable propeller can also be damaged during transportation. The only place remaining for the propeller to be installed, is in the middle of the wing. Wind tunnel tests revealed, that working propeller in the slot of the wing, augments vortex flow [6]. Curve of CL(Alfa) is shifted approximately by $\Delta CL = 0.1$ to higher lift coefficients for all angles of attack. Additionally maximum useful angle of attack increases from 26deg to 32deg and maximum lift coefficient changes respectively from 1.1 to 1.4. Observed changes in characteristics of lift coefficient shows Fig 1. Of course some disadvantages also appear. Drag is increased and vibrations appear because of the propeller working in the slot. Because of the undoubtful advantages of the configuration, overcoming disadvantages, flyable prototype was built. Numerous successful test flights were conducted, confirming good flight characteristics [7].

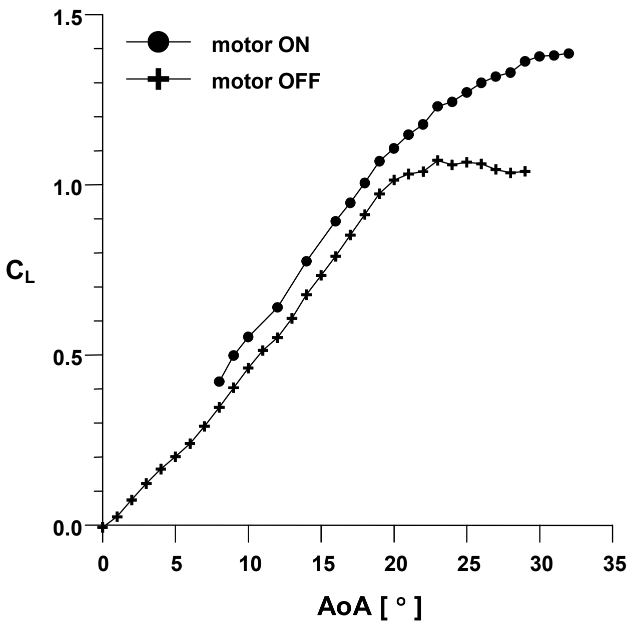

Fig. 1 Lift generated by the cranked delta wing MAV in motor ON and OFF modes, elevator in cruise position.

Next generation of the UAV is currently under development, in purpose to decrease disadvantaging effects and maintain advantageous effects. Optimizing the design, requires knowledge about behavior of the flow over the investigated aircraft. Reliable results from the aerodynamic simulation will be needed, what imposes obvious requirement on appropriate solver settings. The solver has to be calibrated, by comparing results from computations, with results from wind tunnel tests, conducted on the known airplane geometry.

2 Numerical geometry reconstruction

After aerodynamic wind tunnel tests of the "Bee", geometry was written in a data file, containing coordinates of cloud of points obtained from the surface of the aircraft. Coordinates were measured by the optical 3D scanner [8]. The data enabled preparation of geometry, for CFD analysis, in a 3D CAD system. Fig. 2a shows, the model of the aircraft on a wind tunnel test stand, whereas CAD geometry with cloud of points imposed, is presented in Fig. 2b. Created geometry was simplified to avoid problems with grid generation. Small elements like motor housing or servo levers were not taken into account during creation of CAD geometry. Surface of the wing was smoother in contrary to the real UAV, were wing ribs structure can be seen. Aft part of the real aircraft's wings was covered with a film giving savings on mass. Elastic membrane is sensitive to pressures forces acting on the wing and it would require aero elastic calculations to obtain the true geometry which was used in the wind tunnel. Although, the plane model had some manufacturing imperfections, dihedral differed 2mm depending on the left, or the right wing. It was assumed that it is ideally symmetrical. All this simplification saved memory requirements for grid storage and shortened significantly time needed for computations.

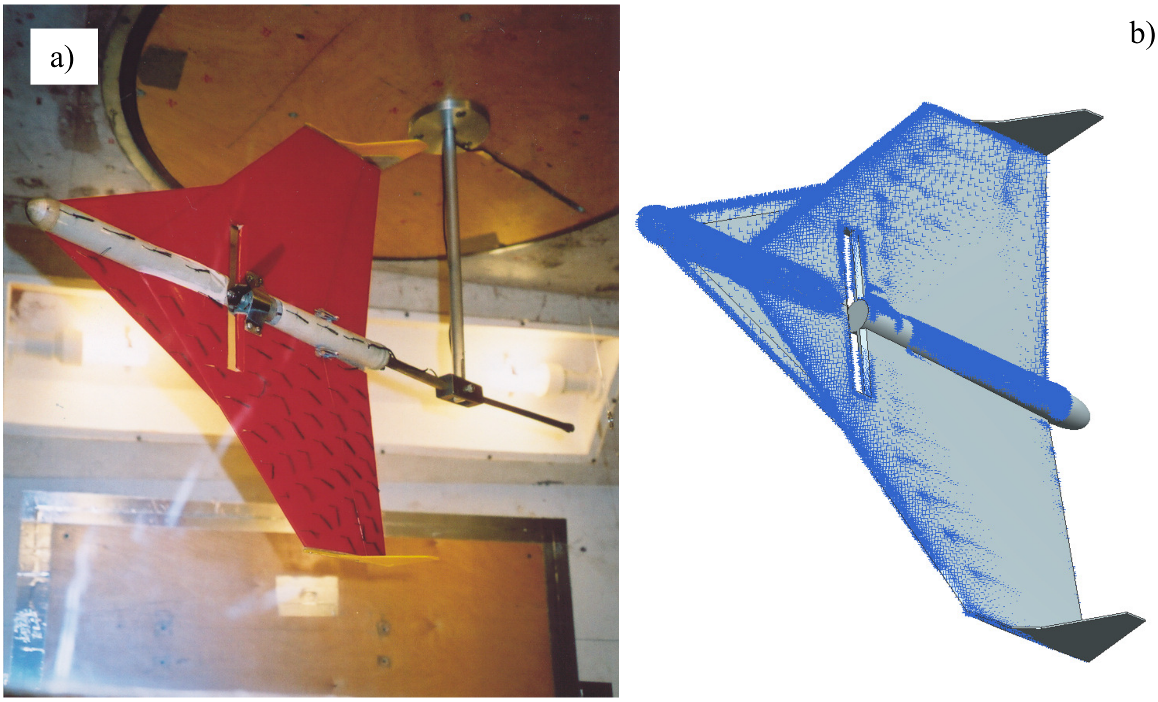

Fig. 2 “Bee” model in the wind tunnel and CAD geometry with cloud of points imposed.

3 Grid generation problems

Complicated geometry was reason of many problems, that had to be overcome, otherwise bad mesh could cause problems with convergence of the solution. During mesh generation highly skewed elements were generated on LEX faces, where LEX meets with fuselage Fig 3. Having fixed geometry from wind tunnel tests sweep angle of LEX nor corner smoothing could not be done and it had to stay that way. Fortunately it did not cause problems with convergence during solution. In future in design process, problem with connection of wing and LEX can be overcame by smoother geometry of the airplane.

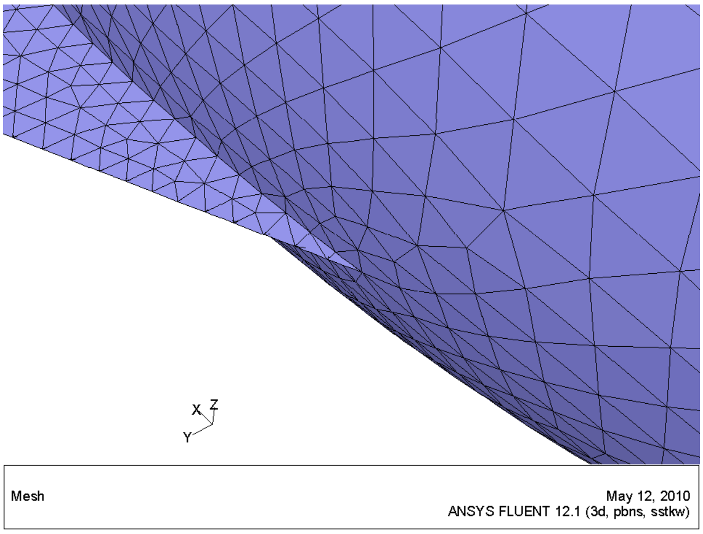

Fig. 3 Highly swept faces in the corner between LEX and fuselage.

Connection of wing and winglet at the leading edge generated similar problems, which was overcome, by using an interface in the inner plane of the winglet. Geometry of volume over the airplane and the winglet can be seen on Fig. 4.

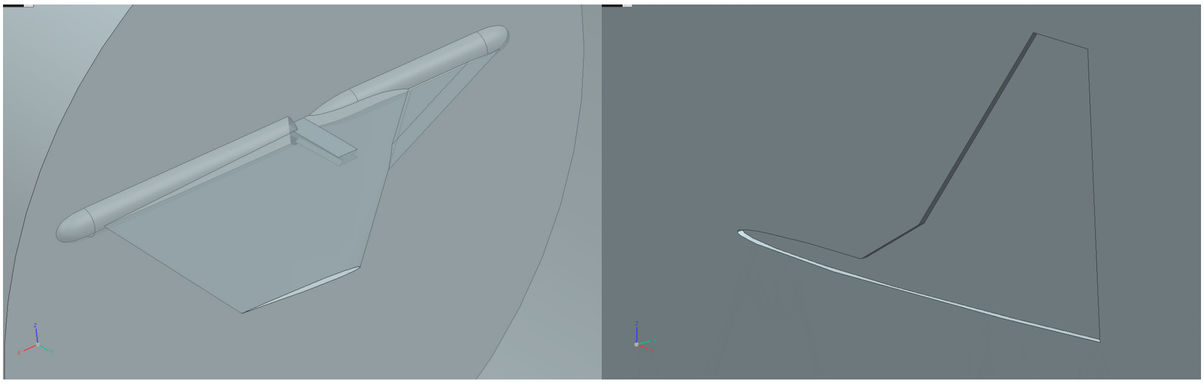

Fig. 4 Volume over plane and winglet.

Leading edge of the winglet, meeting with the wing surface, inclined at sharp angle indicated creation of highly skewed cell elements. Dividing volume in two and introducing an interface Fig. 5 omitted this undesirable mechanism of bad cell creation, because grid points from two volumes did not have to meet.

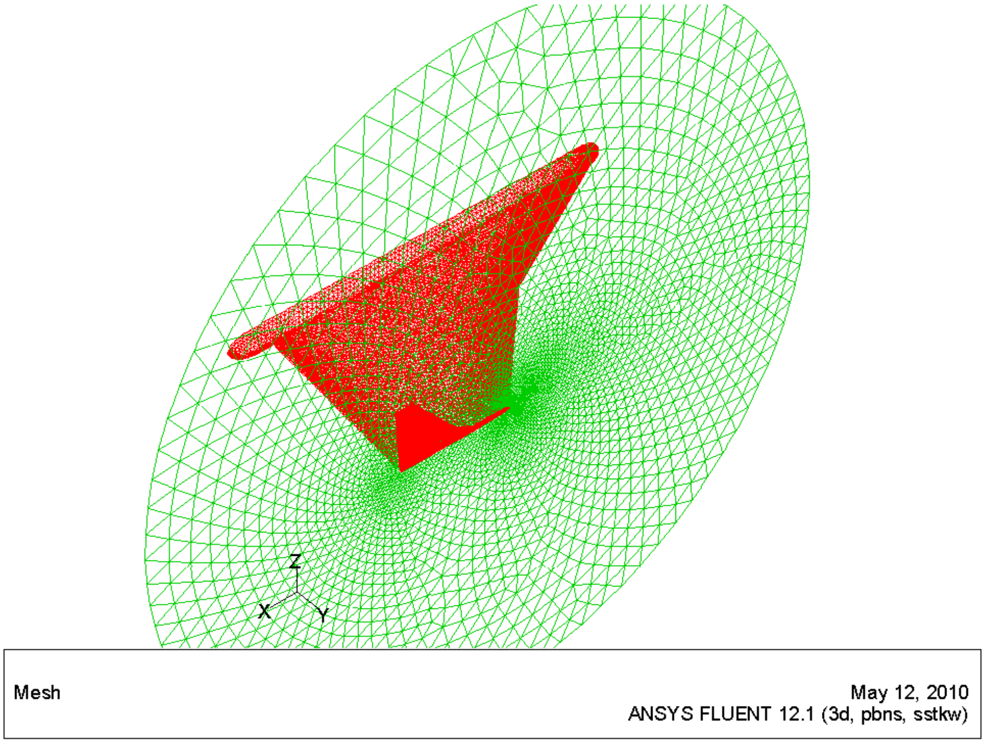

Fig. 5 Interface and "Bee" airplane grid.

Of course this approach blended distribution of pressure between wing and winglet, but it was not very important in this situation. Average estimations of aerodynamic coefficients were needed for the global solver settings, not the details of the flow. For assurance of good solutions details of the flow over the winglet are presented on Fig. 6.

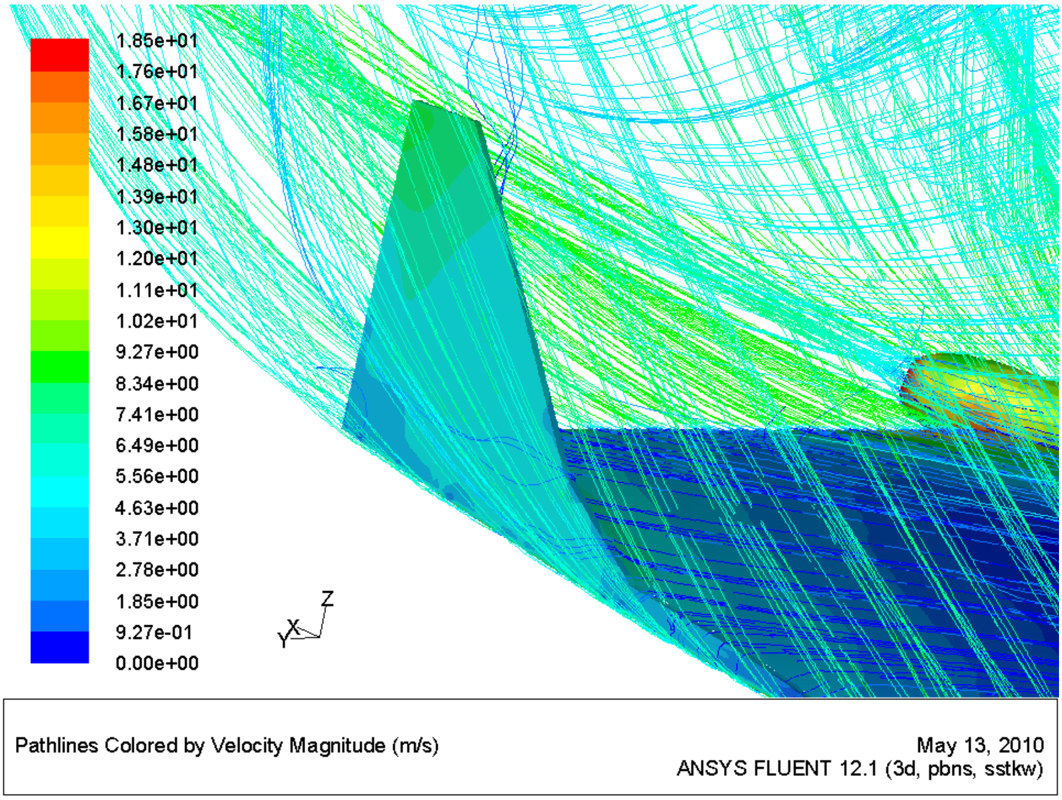

Fig. 6 Pressure distribution and pathlines over winglet geometry.

4 Solver settings adjustment

Computations of steady flow over 3D geometry were conducted. Geometry without propeller and slot in the wing was used to simplify part of the computations on solver adjustment and validation, since results from wind tunnel tests for such configuration of aircraft were available. k-ω-SST turbulence model [9] was used remembering, that analysis with rotating propeller are planed, where Reynolds number wary significantly.

At the beginning the true rate of Turbulence Intensity and Turbulent Length Scale was not known. That is why the problem was solved for a few configurations of mentioned values. Results from the computations were compared with the results from the wind tunnel tests [10]. Last squares error was estimated between curves of CL(AoA), CD(AoA), CM(AoA), with the equation:

$$LeastSquresError = 0.5 \cdot \left(\sum_{\alpha} \left(CL_{Tunel} - CL_{CFD} \right)^2 + \sum_{\alpha} \left(CX_{Tunel} - CX_{CFD} \right)^2 + \sum_{\alpha} \left(CM_{Tunel} - CM_{CFD} \right)^2 \right)$$

Error estimates for different Turbulence Intensity and Turbulent Length Scale are shown on Fig. 7. Minimum error for three cases is almost identical, that is why no more computations to minimize the error were done. For further computations 5% Turbulence Intensity and 0.001m Turbulent Scale Factor was assumed.

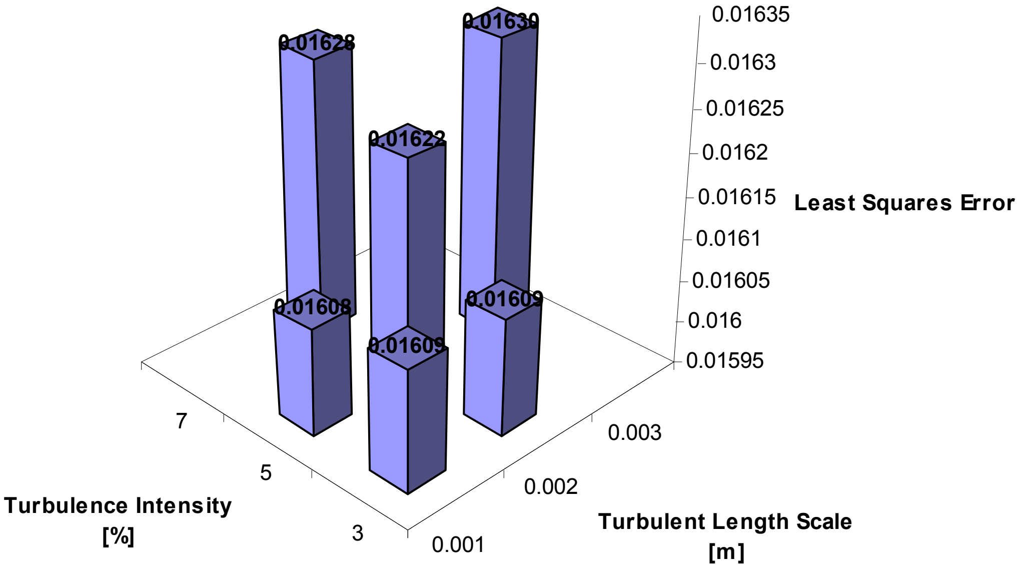

Fig. 7 Least squares error between curves of CL(AoA), CD(AoA), CM(AoA)

5 Results

5.1 k-ω-SST turbulence model

Computations were done for k-ω-SST turbulence model. Fig. 8 Shows comparison between aerodynamic characteristics of the micro UAV obtained from CFD computations and wind tunnel tests. Individual curves agree with acceptable accuracy. Curves seem to be little shifted, it may be due to imprecise definition of zero angle of attack or zero definition of neutral elevon position in the wind tunnel. Furthermore geometry for computations is simplified as mentioned in the chapter about numerical geometry reconstruction. All curves agree butter at lower angles of attack. Maximum lift coefficient computed by Fluent is smaller than achieved in aerodynamic tunnel, but it will help to stay on the safe side of the future design. Predicted momentum coefficient is higher for all angles of attack. Drag coefficient is significantly lower for high angles of attack, but it should not be a problem, because brushless motors have much more power then required for sustained flight and the aircraft will be flying on maximum angles of attack only for short periods of time. Fig. 9 shows vortex flow for 26deg angle of attack. Vortices are very regular creating cone surface. Winglets mounted on the tips of the wing, do not allow vortices created on part of the leading edge with less sweep, to affect the wing surface that would create more lift on wing tips. Changing position of vertical surfaces may have positive effect on aircraft stability and performance and should be considered during design.

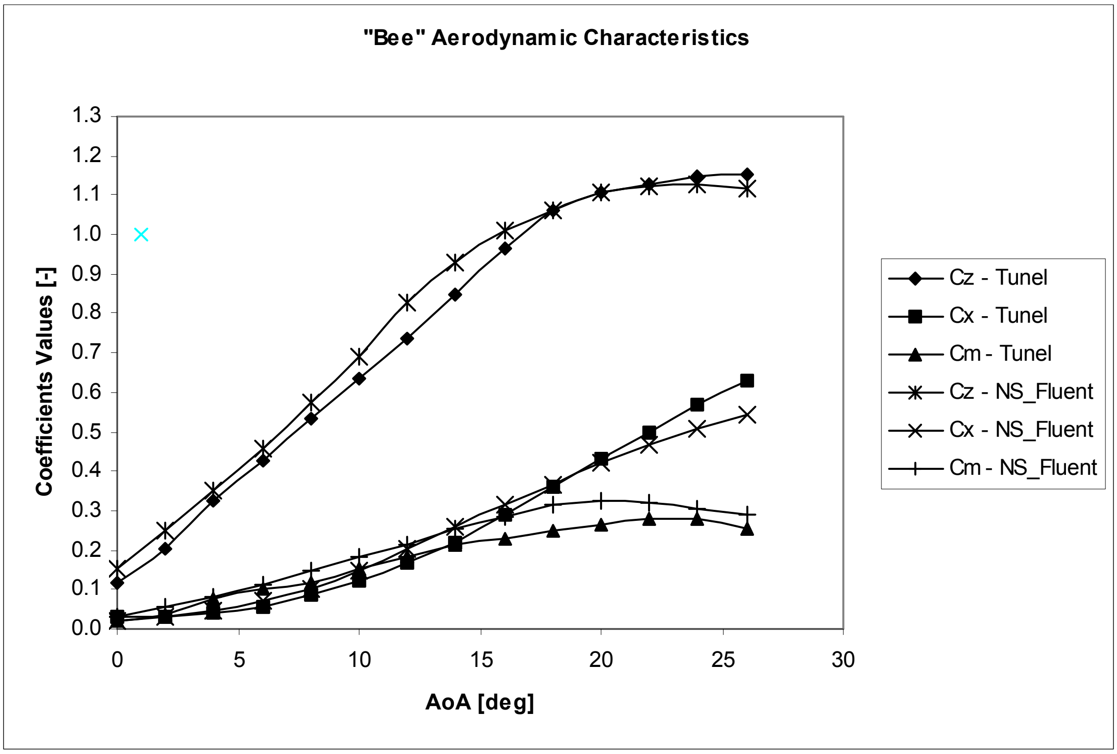

Fig. 8 Comparison of lift, drag and momentum curves.

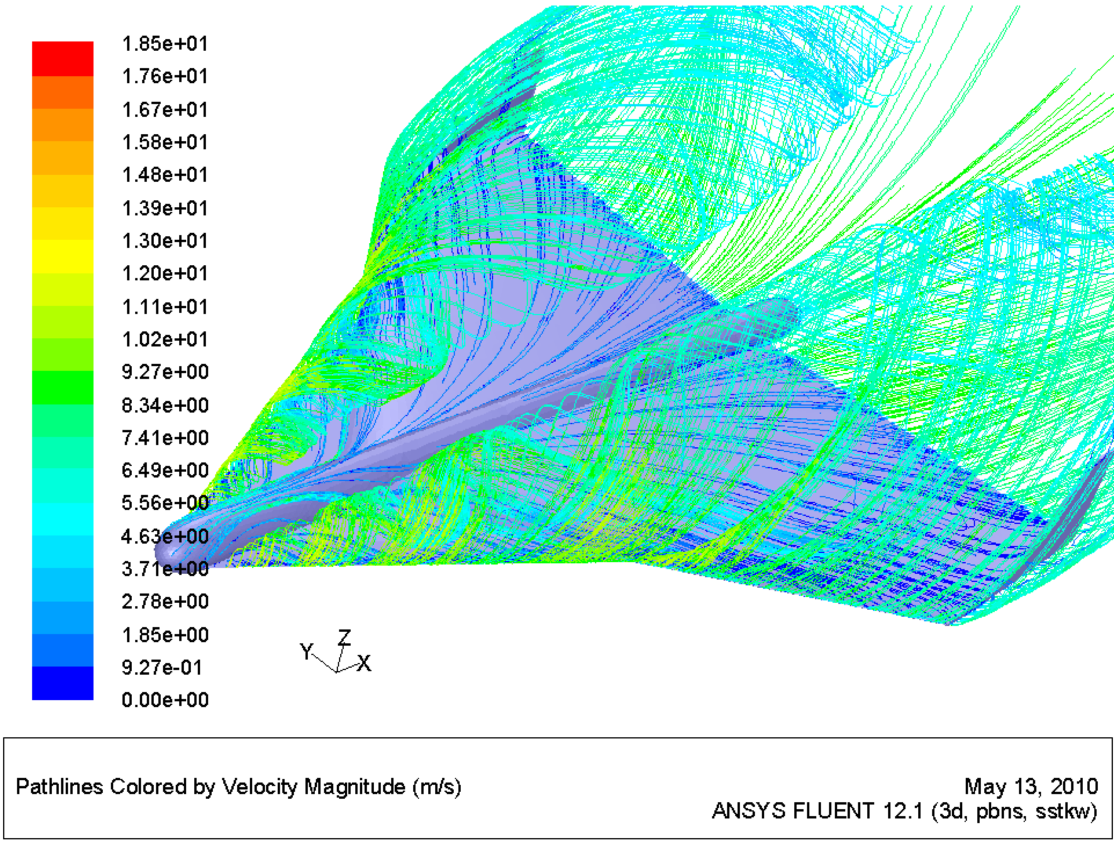

Fig. 9 Vortex flow over "Bee" UAV for AoA 26deg.

5.2 Inviscid model

Computations with inviscid model are much faster than with viscous models. In the design process, where many configurations of the aircraft will be tested, speed of aerodynamic analysis will have crucial meaning. It was hoped to take advantage of the inviscid model, saving on computation time. Results are shown on Fig. 10.

According to experiments described in [1] vortex flow is assumed to be insensitive of Reynolds number. This assumption seems to be right for low and medium angles of attack, because computed aerodynamic characteristics for k-ω-SST and for inviscid models are the same. Unfortunately the characteristics change abruptly for high angles of attack and do not fit to the wind tunnel data. Closer look on the flow for maximum angle of attack, reveals that vortices are destroyed Fig. 11. According to [11] fluid stability depends on viscosity, which slows down disturbances growth. This mechanism is not present in the inviscid analyses. It may occur that flow which is unstable in inviscid analyses, can be still stable in viscous analyses. Flow stability analysis should be conducted to gain certainty, but it is beyond the scope of this paper. Useful conclusion is that the inviscid flow model can not be used for aerodynamic analysis for high angles of attack.

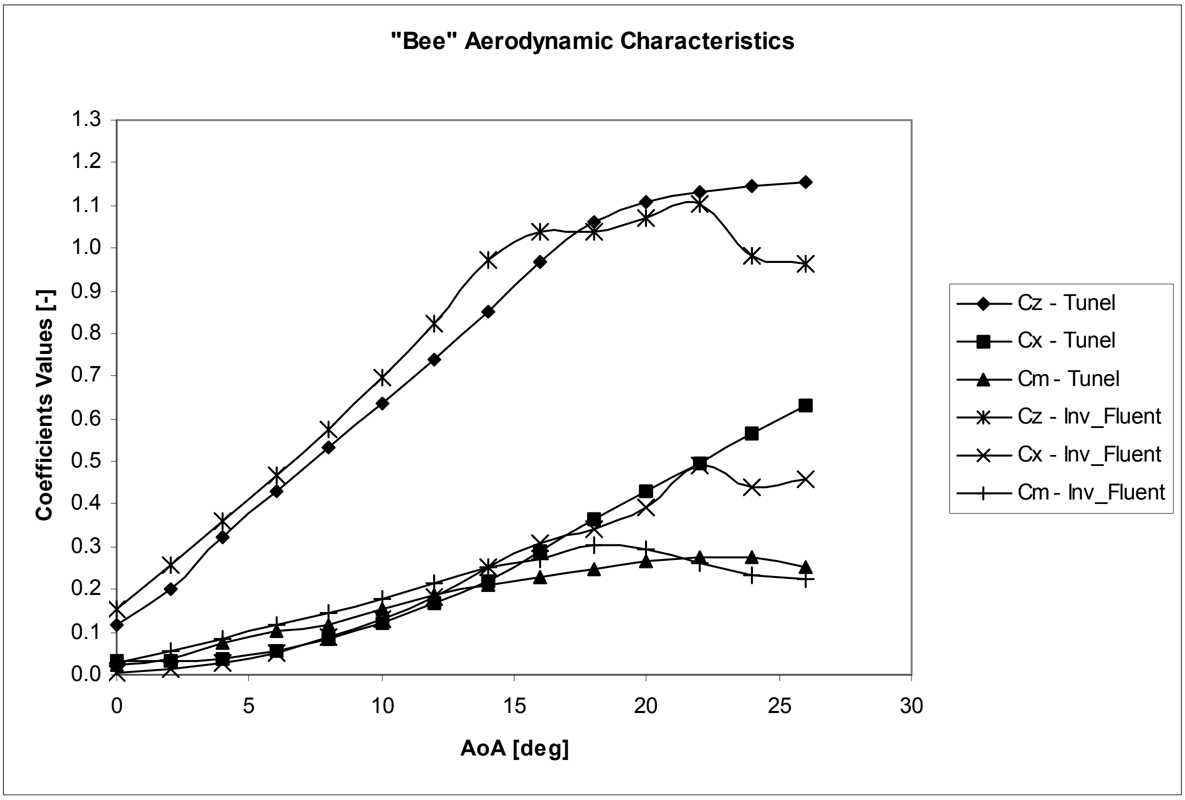

Fig. 10 Comparison of lift, drag and momentum curves in an inviscid model.

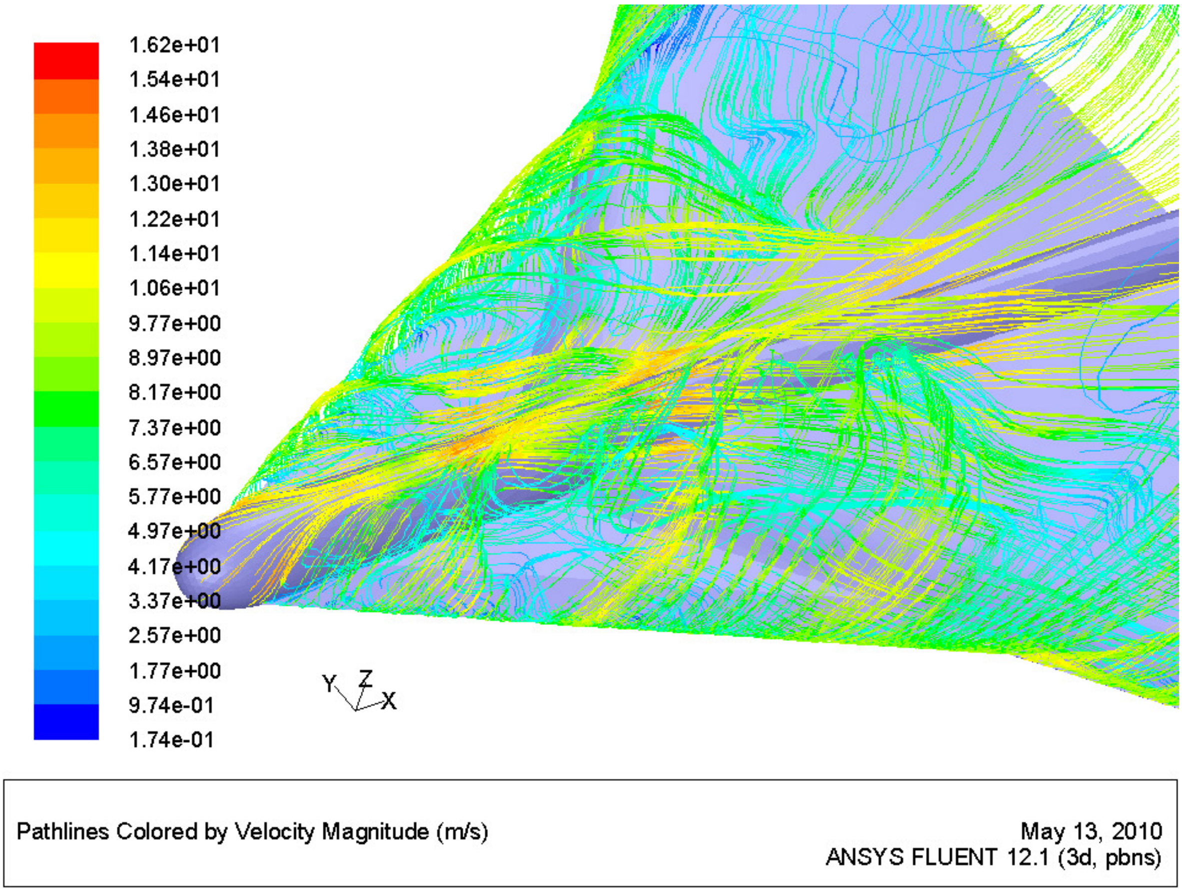

Fig. 11 Destroyed vortex flow over “Bee” UAV for AoA 26deg in an inviscid solver mode.

5 Conclusions

Outcome from the effort taken, are lessons learned about preparation of geometry and grid generation for future designs to avoid problems with convergence of the solutions. Turbulence Intensity and Turbulent Length Scale was estimated for computer analyses, based on comparison of existing aerodynamic data and numerical results. k-ω-SST turbulence model and inviscid model computations, compared with wind tunnel tests, revealed degree of agreement and drawbacks of the flow models. Viscous model provides good agreement with reality. Inviscid model, which computations are faster, can be used for reliable computations at low and moderate angles of attack, but it’s use for high angle of attack is unacceptable. All the work was necessary to provide reliable aerodynamic analysis needed for future design of next generation micro UAV.

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