

**Micro Air Vehicle Design for Aerodynamic Performance and Flight Stability**

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## Abstract

This work is a computational study of the low Reynolds number aerodynamics for MAV applications. The emphasis of the research is to design the optimal MAV model that has a simple geometrical structure but superior in its performances. For a better understanding on the low Reynolds number flow structures, this work started with an investigation of the laminar separation bubble (LSB) on both a two dimensional aerofoil and a three dimensional wing planforms, including rectangular, trapezoidal, Zimmerman and inversed-Zimmerman wing planforms. The fuselage effects on aerodynamics were also studied, and it degraded the overall aerodynamic performance due to the aerodynamic interaction between the wing and fuselage. However, it improves the overall static longitudinal stability for all wing planforms. The aerodynamic comparisons show that the Zimmerman wing-fuselage model has a better static longitudinal stability than other models. The propeller slipstream effects for the Zimmerman wing-fuselage model was also carried out. The overall aerodynamics is improved. The swirl flow from the propeller, however, modifies the overall flow structure on both the upper and lower wing surfaces. The LSB forms on both the lower wing sides (a long bubble is formed on the down-going blade side, and a short bubble is found on the up-going blade side). Flow separation takes place at both sides of the fuselage. The longitudinal stability margin improves almost twice than that on the isolated wing-fuselage model. The static lateral stability shows that the MAV without the vertical stabilizer is statically laterally unstable, whereas the MAV with vertical stabilizer is statically laterally stable. Finally, the fluid-structure interaction effect on aerodynamics for the Zimmerman wing-fuselage model is also investigated, showing that the separation region on the upper wing surface is reduced significantly due to wing flexibility. This results in a significant upwards shift of the lift curve, with a much higher CLmax. The flexible wing also shows a longer working range for lift than the rigid wing model. In other words, the flexible wing model has the capability to carry more payloads.

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Nomenclature

|  |  |  |
| --- | --- | --- |
| AR | = | Aspect ratio, = b2/S |
| b | = | Wing span, m |
| Cd | = | 2D drag coefficient |
| CFx, CD | = | Drag coefficient, = D/qS or Fx/qS |
| CFy, CL | = | Lift coefficient, = L/qS, or Fy/qS |
| CFz | = | Side force coefficient, = Fz/qS |
| Cl | = | 2D lift coefficient |
| CL,max | = | Maximum lift coefficient |
| Cm | = | 2D pitching moment coefficient |
| CMx | = | Rolling moment coefficient, |
| CMy | = | Yawing moment coefficient, |
| CMz | = | Pitching moment coefficient at **, =** |
| Cr | = | Wing root chord, m |
| Ct | = | Wing tip chord, m |
| dt | = | Distance between propeller and leading edge, m |
| d1 | = | Maximum camber location, m |
| d2 | = | Reflex camber location, m |
| D, d | = | 3D, 2d drag force, N |
| Dia | = | Diameter of propeller, m |
| h1 | = | Maximum camber height, m |
| h2 | = | Maximum reflex camber height, m |
| J | = | Advanced ratio, |
| L, *l* | = | 3D, 2D lift force, N |
| Mx | = | Rolling moment, Nm |
| My | = | Yawing moment, Nm |
| Mz | = | Pitching moment, Nm |
| ns | = | Propeller rotational speed, , rev/s |
| P | = | Propeller power, = 2πnsQ, W |
| Q | = | Propeller torque, Nm |
| Rec | = | Mean aerodynamic chord Reynolds number |
| S | = | Wing Area, m2 |
| t | = | Wing thickness, m |
| T | = | Thrust, N |
| Ti | = | Turbulence intensity level |
| U∞ | = | Incoming freestream velocity, m/s |
| Ut | = | Tangential velocity, m/s |
|  | = | Angular velocity along X, Y, Z axis |
| VS | = | Vertical stabilizer |
| VR | = | Resultant velocity, m/s |
|  | = | MAV velocity vector, |
| *wp* | = | Propeller-induced vertical velocity, m/s |
| *ww* | = | Wing-induced vertical velocity, m/s |
|  | = | Distance between wing leading edge and the centre of gravity, m |
|  | = | Distance between wing leading edge and the aerodynamic centre, m |
|  |  |  |
| α | = | Angle of attack, ° |
| αeff | = | Effective angle of attack, ° |
| αi | = | Induced angle of attack, ° |
| ψ | = | Blade azimuth angle, ° |
| β | = | Side slip angle, ° |
| γ | = | Intermittency factor |
| ω | = | Propeller rotational speed, rad/s |
|  | = | MAV angular velocity vector, rad/s |

# Introduction

### Background and motivation

In the past 25 years, the rapid improvement of the new technologies had led to smaller and smaller electronic devices, (i.e. motor, sensors, mini-control system, etc.) and, as a result, the fast development of radio controlled unmanned air vehicles. There are strong interests for both civil and military applications, because of their relatively low cost and propensity for providing accurate surveillance information. The first document specifying the micro air vehicle was released by DARPA (Defense Advanced Research Projects Agency) in 1992.

The primary missions for micro air vehicles included surveillance, detection, communications, and rescue. Surveillance missions include video (day and night), infrared images of battlefields and urban areas. These real-time images can provide enough information to advice the team members to take the right actions, and hence to reduce the harm, e.g. hostage rescue and counter-force operations. The requirements for MAVs include a wide range of possible operational environments, such as urban, jungle, desert, mountain, and arctic. Furthermore, MAVs should be able to finish their mission in all weather conditions, especially in wind shear and gusts conditions [1]. Nowadays, several different types of MAVs have been developed, including fixed-wing, flapping-wing, and rotary-wing MAV. Fixed-wing MAVs would have much longer operation range and can fly at relative high altitude, whereas, flapping and rotary wing MAVs can perform indoor missions with slow flying speed. This study focuses on the fixed-wing MAV with aerodynamic design and stability analysis.

MAVs are usually flying at 8-16 m/s and the corresponding Reynolds number varies from 5×104 to 8×105 [2-4]. In this low Reynolds number rage, technical solutions are required to solve the problems, such as low Reynolds number aerodynamics, (which determines the aerofoil/wing planforms aerodynamic efficiency), propulsion system (propeller thrust level and power requirement), stabilities and controls (both longitudinal and lateral stability). Computational modeling, and wind tunnel work have generally pushed the small fixed wing MAV designs to a thin cambered, constant thickness, with a low aspect ratio. The low aspect ratio is due to the size constraint of the aircraft, and therefore maximizing the lift is a priority target for such low speed flying vehicles. A thin cambered plate aerofoil offers better aerodynamic performances as compared with the normal conventional aerofoils [5]. This is because the viscous boundary layer thickness (on the conventional aerofoil) increases significantly and resulting in a decrease in the lift curve slope in the linear range [6]. The advantage of using the reflex camber is to produce a nose-up pitching moment (because the wing or aerofoil usually shows a nose-down pitching moment and gives a negative contribution on the static longitudinal stability). Much work has been done on finding suitable aerofoils and the results showed that the cambered plate aerofoil offers much better aerodynamic performances than the conventional aerofoils [7]. Flat plate thin wing planforms were studied quite widely both numerically and experimentally [8-11]. However, little work had been done on the aerodynamic influences from the cambered thin wing planforms and the fuselage. The propeller slipstream effects on the low Reynolds number aerodynamics, however, were still not fully understood. For a better understanding on the low Reynolds number aerodynamics, the definition of low Reynolds number is discussed in the Section 1.2.

### Low Reynolds number aerodynamics

The small vehicles aerodynamics can be characterized by the Reynolds number into different regimes. For the conventional steady-state aerodynamics, Reynolds number is defined by the product of the aerofoil characteristic length and the freestream velocity divided the dynamic viscosity, , as shown the following relationship:

It is very useful to understand the Reynolds number characteristics of a body immersed in a fluid. Several authors have been looking into the details of different range of Reynolds number. A detailed Reynolds number regimes was given by Carmichael [5] in 1982, Lissaman [12] in 1983, Gad-el-Hak [13-14] in 1990, Mueller [15-16] in 1999, and Shyy *et al.*[17] in 2008. The Reynolds numbers at different operating regimes are summarized here.

* At extremely low Reynolds (1,000 < Re < 10,000), typical of small insects and small handmade aircraft, (such as dragonfly, house fly, or indoor model aeroplanes). The dragonfly has a saw tooth aerofoil which generates eddies to help the flow to keep attached [18-19]. The boundary layer is strongly laminar. The insects normally use the eddy-induced energy. Obviously, they are using an unconventional source of lift.
* At Reynolds number of 10,000 < Re < 30,000, a laminar boundary layer is formed around the lifting surface. The lift-to-drag ratios are the best obtainable at this Reynolds number region (due to their relatively low drag). On the other hand, the operating lift coefficient is restricted (typically CL, max is 0.5 or less) [5]. Trimming these aerofoils to achieve a higher lift coefficient will lead to the separate the laminar boundary layer without any reattachment.
* For Re between 30,000 and 70,000, usually for small MAVs, the laminar separation bubble may start to form in the boundary layer. However, a thick aerofoil (6% above) can have significant hysteresis effects due to the laminar separation with transition to turbulent flow [1]. Gottingen 801 aerofoil was tested by Tani [20] experimentally. He showed that at a Reynolds number around 42,000, separated boundary layer without reattachment was formed on the upper wing surface. The pressure distribution was relatively flat along the entire aerofoil. However, as the Reynolds number increases to 75,000, the pressure distribution exhibits a flat portion which indicates the presence of the laminar separation bubble. Usually, a short bubble is formed in this region.
* At 75,000 < Re < 200,000, the laminar separation bubble may still exist and the laminar portion of the shear layer extends as the Reynolds number increases. Namely, long bubbles may occur in this Reynolds number regime and the pressure distribution on the upper surface of an aerofoil has a smoother recovery to the unseparated turbulent boundary layer value. This gradual pressure rises, along with the reduced minimum pressure peak on the upper wing surface of the aerofoil [21].
* For Re > 200,000, the aerofoil performance improves significantly.

The lift-to-drag ratio is a measurement of the effectiveness of the aerofoil. CL/CD is very low at low Reynolds numbers and it improves significantly as the Reynolds number increases, in the range of Rec = 104-106 in Figure 1. In general, Below 104, belongs to the insects and low Reynolds number fliers exploiting unsteady effects to achieve remarkable aerodynamic performance [14, 22-23].



Figure Aerofoil performances as a function of chord Reynolds number (from McMasters and Henderson[24])

At the low Reynolds number region, the aerofoils usually have conflicts between one parameter to another and this increases the degree of complexity for designing a low Reynolds number aerofoil. Figure 2 shows the inter-reaction between separation/reattachment, lift enhancement, transition delay/advancement, and drag reduction, and the aerofoil aerodynamic performances. It is obvious that there must be compromises to achieve a particular design goal. As mentioned before, in the range of Reynolds number of 104-106, the aerofoil may be affected by the presence of the LSB which may influence the aerofoil lifting surface and lower the overall aerofoil performances. A turbulent boundary layer may have good resistance to the adverse pressure gradient, and hence resistance to separation. This can be a good achievement for an aerofoil to have better aerodynamic performances. However, the skin friction drag for a turbulent boundary layer can be an order of magnitude larger than the laminar boundary layer. In the case of delayed transition, low skin friction drag is achieved, but the laminar boundary layer, on the other hand, can only accept a lower adverse pressure gradient as compared to the turbulent boundary layer. As the incidence increases, the LSB moves towards leading edge and it shortens as the angles of attack increases even further. It then breaks down, either separating completely or forming a longer bubble. The form drag will increases dramatically and hence to reduce the overall aerofoil aerodynamic efficiency.

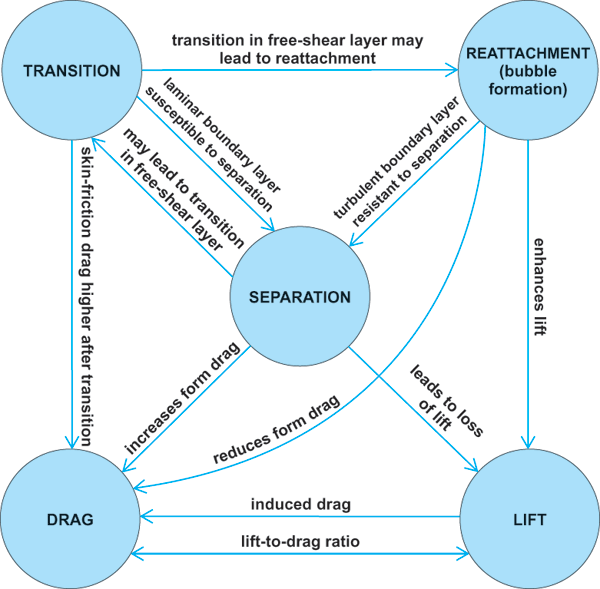


Figure Inter-relation between flow structure and forces (i.e. by Gad-el-Hak [13])

### Objectives

MAVs are notoriously difficult to fly. This is because it has lower gust wind resistance and highly maneuverability demand. The aerodynamics, one of the main challenges, is beset by several unfavorable flight problems:

1. Low Reynolds number aerodynamics: MAVs normally have the Re in the range between 104 and 105 [12]. Many complicated flow phenomena can occur inside the boundary layer, such as separation, transition, and reattachment, and can significantly affect on the lifting surfaces. The laminar separation bubble inside the boundary layer determines the boundary layer behaviour and the stalling characteristics of an aerofoil [14, 20, 25]. Therefore, a good understanding of low Reynolds aerodynamics boundary layer behaviour is required.
2. The low aspect ratio wing planform suffers stronger effect from the wingtip vortex than the high aspect ratio wings. This wingtip vortex can be easily energized with the aforementioned separation bubble which can lead to tip vortex destabilization [26-27]. Flat plate wing planforms have become one of the most representatives and it has been investigated by several different authors. Torres and Mueller studied various flat plate wing planforms at Re between 7×104 and 2×105 [9], and in 2011, Okamoto investigated various flat plate wing planforms at even lower Reynolds number of 1×104.
3. Propeller effects on low Reynolds aerodynamics: the presence of the wing behind the propeller has the effect to the wake geometry and hence modifies the overall performances. A good understanding on the interaction between the wing and the propeller is required and it requires to characterise both the propeller slipstream effect on the wing and the reciprocal influence of the wing presence on the propeller flowfield and performances [28]. Most of the investigations on propeller slipstream effect were focused on large aircraft. The propeller slipstream effect on the MAVs becomes the main challenge due to not only aerodynamics but also the overall stability (as additional forces and moments can be produced due to the interactions between the propeller and the wing planform).
4. Additional effects from the fluid-structure interaction on low Reynolds aerodynamics. MAVs with carbon fiber wings are definitely too heavy. However, MAV uses the membrane skin for the wing is apparently too soft. Different material behavious would generate different aerodynamic characteristics due to the wing structure flexibility. Therefore, fluid-structure interaction is important for flexible wings. Figure 3 shows the MAV prototype and the flight tests.
5. The stabilities analysis (both longitudinal and lateral stabilities) for the small flying vehicles is also required. The range of flyable CG locations is generally only a few millimeters long, which represents a strenuous weight management challenge [29].

|  |  |
| --- | --- |
| H:\Jason_chen\PHD2\PHD_year2\MAV\assembled_MAV\normal_propeller_assemble\mav4_fuse_assum_2013.jpg   1. MAV assembled model | C:\Users\jason\Pictures\2012-05-18\098.JPG   1. MAV Prototype |
| I:\academic\year 4\final_year_project\MEC_488_Individual_project\project_thesis\flying test\2009_04_11\coupling_occurs.bmp | I:\academic\year 4\final_year_project\MEC_488_Individual_project\project_thesis\flying test\2009_04_08\gust_wind2.bmp |

Figure MAV prototype

The problem, as mentioned above, can be investigated stage by stage as following: a) aerofoil design and analysis (cambered plate aerofoil contains both positive and negative cambers); b) wing planform study (trapezoid, Zimmerman, inversed Zimmerman, elliptical and circular wing planforms); c) propeller slipstream effect on the low Reynolds number aerodynamics; and d) the fluid-structure interaction effects on the aerodynamics (based on the Zimmerman wing-fuselage model).

### Dissertation outline

After the introduction in this chapter, Chapter 2 presents the mathematical models and numerical methods employed in this study. The following four chapters are arranged according to the different stages of the MAV investigation. Studies are from the low Reynolds number aerodynamics (including the leading edge separations on the wings) to propeller slipstream effects, and finally the fluid-structure interaction (FSI) for flexible MAV wing.

In each chapter, the related literature is reviewed at the beginning and the corresponding validation cases are presented. The findings for each chapter are summarized at the end. The overall conclusion is presented in Chapter 7.

The fluid-structure interaction effect, the final part of the thesis, is carried out for the Zimmerman wing-fuselage with and without the carbon fiber rods (the wing was made by the depron foam and the carbon fiber rods are attached underneath the wing to enhance the wing strength). Literature reviews on the fluid-structure interaction effects is also included, and the fluid-structure interaction methodology is also validated with a validation case.

# Governing Equation and Numerical Methods

Different discretization approaches are used for today’s CFD applications. There are in general in three different types: finite difference method (FD), finite volume method (FV), and finite element method (FE). The finite difference method is probably the oldest method for numerical solution of the partial differential equations. Such method usually needs structured grids, serving as local coordinates lines. The FV method, as on the other hand, combines the best from the finite element method (FEM, i.e. geometric flexibility), with the best of the finite difference method (FDM, i.e. the flexibility in defining the discrete flow field). It uses the integral form of the conservation equations, and the whole domain is divided into many sub-domains, i.e. control volumes. Conservation equations are applied to each control volume, and the node lies in the centre or the vertex of the cell where the variables are stored. The FV method is suitable for complex geometries, which means it does not have restrictions on grid types. The disadvantage of this method is that higher orders are more difficult to develop in three dimensional cases. The FE method, as the third type of method, is similar to the FV method. The physical domain is also reconstructed by discrete volumes (i.e. finite elements) that are generally unstructured. One of the important advantages is the ability to deal with arbitrary geometries. The FE method usually uses the unstructured mesh and the principle drawback is that the matrices of the linearized equations are not as well structured as those for structured grids. Therefore, it might have lower efficiency.

In general, the FD method can be very effective and simple to perform the higher-order schemes with a regular grid. The FE method has an ability to deal with arbitrary geometries and unstructured grid is usually used. The FV method is the simplest to understand and program, as all terms in the individual equation has its physical meanings. The FV methods can be used for any type of grids, which is why it is popular for CFD codes. This is the method used for the investigation in this thesis.

The control volume (i.e. the approach for FV method) deals with the flow in a certain spatial region. shows the control volume and the typical notation used for two dimensional grids: a) vertex-centre based, b) cell-centre based. For any conserved intensive property (i.e. mass, momentum, and energy). In Ansys Fluent, it stores discrete values of the scalar at the cell centre.

The numerical solution of the conservation equations with the FV approach can be written into the integral form given by,

|  |  |
| --- | --- |
|  | Eq. |

or,

C:\Users\jason\Desktop\vector_plot\control_volume_in_words.emf

The above equation represents that the rate of change of the amount of property in the control mass is the rate of change of the property within the control volume plus the net flux of it through the control volume boundary due to the fluid motion relative to the control boundary. In Eq. 1, denotes the boundaries of a control volume and a surface element, respectively, and is outward going unit normal. represents the vector quantities. For continuity equation, is ρ. is the convective flux tensor and is the diffusive flux tensor. stands for the sources locally inside the volume, and means surface source tensor.

C:\Users\jason\Desktop\vector_plot\control_volume_grid.emf

Figure Schematic of two-dimensional control volumes grids

### Governing equations

The equation of the fluid dynamics is the well-know Navier-Stokes equations that come directly from the conservation laws of mass, momentum, and energy. While finite difference methods are based on a discretization of the differential form of the conservation equations, the finite volume methods are based on a discretization of the integral forms of the conservation equations. The integral form of the mass, momentum conservation equation follows directly from the general equation (Eq. 1 ).

For continuity, Eq. 1 can be rearranged into:

|  |  |
| --- | --- |
|  | Eq. |

The first term in represents the time rate of change of total mass inside the finite control volume , and the second term denotes the net mass flow out of the control volume, and the positive mass flow corresponds to outflow and negative to inflow. By applying the Gauss’s divergence theorem to the convection term, the surface integral can be written into a volume integral, therefore, can be written into the differential form:

|  |  |
| --- | --- |
|  | Eq. |

For a given coordinate system, can be expressed using the divergence operator into the particular coordinate system. Hence, as an example, the can be reformulated into the Cartesian coordinate system:

|  |  |
| --- | --- |
|  | Eq. |

where in denote as the velocity components in x, y, and z direction. The derivation of also can be found from reference [30], and such equation in integral foam are also introduced from Versteeg[31], Cebeci [32], etc.

The equations for the momentum, consistent with the generic integral form of Eq. 1 are summarized as the following equations:

|  |  |
| --- | --- |
|  | Eq. |

The left hand side of the represents the time rate changing of momentum due to the unsteadiness of the flow properties inside the control volume, and the second term means the net flow of momentum out of the control volume across the surface S. The first term on the right hand side of the denotes the body forces, and the second, third terms are the sum of pressure forces, and the viscous forces acting on the flow as it across the control volumes, respectively. The surface forces are such as pressure, normal and shear stress, surface tension, etc. The body forces are such like gravity, centrifugal, and Coriolis forces, and electromagnetic forces.

The energy conservation is usually for the compressible cases, whereas the current work is mainly focusing on the incompressible flow. Therefore, the energy equation is not added.

### Navier-Stokes Equation: Vector-variable form

It is convenient to combine the continuity, momentum and energy equations into a compact vector-variable form. Rearrange the integral foam governing equation, Eq. 1, as following:

|  |  |
| --- | --- |
|  | Eq. |

where the vector of the conserved variables, and the fluxes are listing as following:

|  |  |
| --- | --- |
|  | Eq. |

The contravariant velocity, U, in is expressed as following:

The shear stress tensor (Stokes’s hypothesis) is introduced by

|  |  |
| --- | --- |
|  | Eq. |

The incompressibility condition simplifies the governing equation which has decoupled the energy equation with both continuity and momentum equations. There is no volume source due to body forces and volumetric heating, and hence the source term vanishes.

### Turbulence and transitional models

The three dimensional flows are highly unsteady and most of them are turbulent. Therefore, different treatments are required. Turbulent flows fluctuate on a broad range of length and time scales. This property makes direct numerical simulation of turbulent flows very difficult. The RANS and URANS (i.e. Unsteady/Reynolds-averaged Navier-Stokes) methods will be used for this investigation and LES (Large Eddy Simulation), DNS (Direct Numerical Simulation) will not be considered as they require very large simulation time. The RANS method is based on the time averaging and the flow is statically steady, all of the unsteadiness is averaged out (i.e. all unsteadiness is regarded as part of the turbulence). The non-linearity term must be modeled for the Navier-Stokes equation and for a statically steady flow (as can be seen in a), the time-averaged variable can be written as the following:

|  |  |
| --- | --- |
|  | Eq. |
| where | Eq. |

The t is the time and the T is the averaging interval.

For the unsteady flow field, the time averaging is not suitable anymore. Therefore, ensemble averaging is used (in b), shown as following:

|  |  |
| --- | --- |
|  | Eq. |

where N is the number of cycles(or number of ensembles involved) and the fluctuation of a turbulence flow will have impact on the value N. Hence, a relative large value for N is necessary.

C:\Users\jason\Desktop\vector_plot\Averaging_variable.emf

Figure the averaging methods [33]

Therefore, the averaged continuity and momentum equation for the incompressible flow with no body forces case can be written in tensor and Cartesian coordinates as:

|  |  |
| --- | --- |
|  | Eq. |
| where can be rewrite into tensor form,  and  and for incompressible flow, the term and hence the viscous-stress tensor is written as following:  or | Eq. |

In , the first two terms on the right hand side are the averaging term and the third term is the fluctuation term. This fluctuation term is also called the Reynolds stress term. Different forms for and Eq. 13 can be also found from ref. [32].

However, the additional terms were formed from the RANS. For a three-dimensional case, the six unknowns are: , , , , , and . The term is called Reynolds tensor stress, and term is the viscous tensor stress. Therefore, additional equations are required to solve the closure problem for the equations. The turbulence models should be regarded as engineering approximations rather than scientific laws. Two turbulence models are selected for this research project and the details of the models are described in Section 2.3.1 and 2.3.2.

#### K-ω -SST (shear stress transport) model

Menter’s k-ω-SST turbulence model [34] is used to close the RANS equations. This model is based on the original k-ω model, which was introduced by Wilcox [35]. The k-ω model is activated in the near wall region and the standard k-ɛ activated in the outer wake region and in the free shear layers. For clarity, the transition model, to be discussed in section , is showed here.

|  |  |
| --- | --- |
|  | Eq. |
|  | Eq. |
| where  , ,  and , ,  and , | Eq. |

represents the generation of turbulence kinetic energy due to mean velocity gradients, represents the generation of, and represent the dissipation of and due to turbulence, represents the cross-diffusion term, and , are user-defined sources terms.

The production of kinetic energy, *k, is*

|  |  |
| --- | --- |
|  | Eq. |

and the production of *ω is*

|  |  |
| --- | --- |
|  | Eq. |

The dissipation of k is given by:

|  |  |
| --- | --- |
| , , , ,  and , , | Eq. |

and the dissipation of ω is given by:

|  |  |
| --- | --- |
| , , , ,  and | Eq. |

The cross-diffusion term: is given by

|  |  |
| --- | --- |
|  | Eq. |

The SST model coefficients are blended between the inner and outer zones given by the expression:

|  |  |
| --- | --- |
|  | Eq. |

The blending function F1 is one at wall and zero far away from the wall, thus activating the Wilcox k-ω model in the near–wall region and the k-ε model for the outer zone. The blending function F1 is given by:

|  |  |
| --- | --- |
| , | Eq. |

where y is the distance to the next surface and is the positive portion of the cross-diffusion term, and

, , , ,

#### Menter’s transition model (four equations model)

The transition mechanisms, in general, include natural transition (i.e. Tollmien-Schlichting wave), separated flow transition, reverse transition (i.e. transition from turbulent to laminar or relaminarization), the bypass transition [36]. Namely, the natural transition occurs through different mechanics in different applications. Transition occurs due to the T-S (Tollmien-Schlichting) wave instability [37]. Viscosity destabilizes the T-S waves and the waves start to grow very slowly, and nonlinear three-dimensional disturbances have been formed due to the growing weak instability. After certain stages, the three dimensional disturbances transform into turbulent spots, and the transition from laminar to turbulent is completed after the turbulent spots are combined. The separated flow transition usually occurs at low Reynolds number due to the boundary layer does not have enough momentum to overcome an adverse pressure gradient with minimum disturbance. A large number of both numerical and experimental data show flow separation and reattachment in the transition region (i.e. Langtry [36], Lin [38], Abdelkader [39], Genc [40], Seyfert[41], Counsil [42], and Grabe [43]). It quite often occurs with a laminar separation bubble (LSB). The bypass transition, as the name suggests, the early stages of natural transition (stages of T-S waves, spanwise vorticity, and 3-D vortex spots) process are bypassed. It usually occurs at flows having high freestream turbulence levels [44]. The transition is one of the most complicated phenomenon in the flow, as this mechanism depends on several factors, such as freestream turbulence intensity, pressure gradient along the laminar boundary layer, curvature of the surface, surface roughness, freestream Mach number, acoustic disturbance, and structural vibration issues.

There are three main methods to predict transition [45]. The first approach is the method which utilizes the local linear stability theory. The basic assumption is that transition starts when a small disturbance is introduced at a critical Reynolds number and is amplified by a factor (a typical example of n value equal to 9, has a Re about 8000). It is relatively straight-forward for a two-dimensional case. However, for three-dimensional cases, it cannot work due to the streamline direction is not aligned with the grid which track the growth of the disturbance amplitude ratio along the streamline. Also, the high nonlinear effects would still limit the method, such as high turbulence intensity, roughness. The transition n factor is not universal and depends on the wind-tunnel freestream, acoustic environment and model surface smoothness. It is too complicated to work with Navier-Stokes equations, as the stability analysis typically based on the velocity profiles obtained from highly resolved boundary-layer codes that must be coupled to the pressure distribution from RANS CFD codes [46].

The second approach bases on the experiment correlation based on large number of experimental data. However the wind-tunnel environments can be crucial to the corrections for freestream turbulent intensity, pressure gradients, model surface roughness, acoustic, and vibration. It is also not compatible with the three-dimensional flows and unstructured/parallel CFD codes because it utilizes nonlocal transition criteria model [36].

The third approach is the low Reynolds number transition models [44, 47]. They typically correlate the transition momentum-thickness Reynolds number to local freestream conditions such as freestream turbulence intensity, and pressure gradient. The models are often accurate enough to capture the transition mechanism and it is also compatible with the commercial CFD codes.

The Menter’s transition model [48] is one of the low Reynolds number transition models used in this work for simulating the propeller slipstream effect on the micro air vehicle. The idea for the transition model is based on the momentum-thickness Reynolds number, which is proportional to the maximum strain-rate Reynolds number. It can be used to calculate the local property though each grid point in . The main requirement for this transition model is the local variables and gradients, as well as the wall distance.

|  |  |
| --- | --- |
| . | Eq. |

The model combines the Shear Stress Transport (SST) k-ω model with two additional transport equations: the turbulence intermittency, , to trigger the transition, and the transport momentum thickness Reynolds number , to capture the nonlocal influence of the turbulence intensity. The intermittency transport equation is shown as

|  |  |
| --- | --- |
|  | Eq. |

The transition sources are:

|  |  |
| --- | --- |
| ,  and | Eq. |

where is the vorticity magnitude, is the empirical correlation that controls the length of the transition region, and controls the transition onset location. Both are dimensionless for the intermittency equation in the boundary layer.

|  |  |
| --- | --- |
|  | Eq. |

is the critical Reynolds number where the intermittency first starts to increase in the boundary layer. The constants for the intermittency equations are:

|  |  |
| --- | --- |
|  | Eq. |

The equation for the transition momentum thickness Reynolds number is given by

|  |  |
| --- | --- |
|  | Eq. |

The source terms are defined as following:

|  |  |
| --- | --- |
|  | Eq. |

where t is a time scale that is present for dimensional reasons. The blending function is used to turn off the source term in the boundary layer and allow the transported scalar to diffuse from the freestream. is zero in the freestream and one in the boundary layer. It is defined as

|  |  |
| --- | --- |
| and | Eq. |

The constants are and . The boundary condition for at a wall is zero flux, and it should be calculated from the empirical correlation based on the inlet turbulence intensity for the inlet boundary condition. For proper behaviour of the transition model, the y+ value for the first cell above the wall has to be in the order of one as recommended, and more details can be seen in Menter’s papers [36, 48].

This model has been used by a number of researchers for low Re transitional flows. For example, a detailed study on two specific parameters ( are used in the intermittency equation for controlling the length of transition region and onset location of transition, respectively) is shown by Suluksna [49]. Benyahia [39] conducted a validation study for the model for low Re number flows. According to the comparison between the numerical and experimental data, it shows that the model accurately predicts the location and extent of the two-dimensional laminar separation bubble. Counsil [42] also studied the two dimensional aerofoils using the transition model. His results have shown that the transition model was accurate in the intermittent regions. A comparative study of four aerofoils at low Reynolds number using the different transition models are performed by Seyfert et al. [41]. They have pointed out that the application of is difficult if the turbulence intensity exceeds 0.1%, and for parallel computing, the model is more effective than the method, as the approach is based on the local quantities. Two three-dimensional cases were tested by Grabe and Krumbein [43], and the results have shown that the model is not reliable for three dimensional flows, especially for high Reynolds numbers which contains the compressibility effects. They pointed out that the possible reason is due to the characteristics of two-dimensional boundary layers has been used, of which the three-dimensional transition mechanisms are not take into account (such as the crossflow instability effects).

### Solver (segregated algorithm)

Two solvers are available in FLUENT: a) density-based solver, b) pressure-based solver. The density-based solver is usually for compressible flows and the pressure-based solver is for the incompressible flow calculations. An overview of the progress for the pressure-based solver is shown in Figure 6. For the current work, the incompressible solver will be used (i.e. pressure-based solver). The pressure based-based solver uses an algorithm where the governing equations are sequentially solved. To obtain a converged numerical solution, iterative process will be applied to the governing equations as they are non-linear and coupled. In the pressure based segregated algorithm, as shown in Figure 6 (left), the solution variables, such as, are solved one by one. This method requires less memory, since the discretized equations need only be stored in the memory one at a time. Fluent supplies several options: 1) The SIMPLE algorithm, it was originally introduced by Caretto [50] et al., in 1972. This algorithm uses the relationship between the velocity and pressure correction to enforce mass conservation and to obtain the pressure field. 2) The SIMPLEC algorithm [51] that has a similar procedure as compared with SIMPLE. The only difference is that it added a modified correction equation which has shown to accelerate the convergence in the problem. For the current investigation, the SIMPLE algorithm is applied for all the simulations. The iterative approach process is listed step by step as following:

1. Initialing the flow field by guessing the pressure values, as denoted by at all ‘pressure’ grid point (i.e. pressure are calculated at the solid grid point, see Figure 7), and from the momentum equations to obtain the velocity components: and values are associated with the proper ‘velocity’ grid point (i.e. velocities are calculated at open grid point, see Figure 7).
2. Solve for for the next time step.
3. Use to solve for at all the interior grid points.
4. Calculate at all internal grid points,.
5. Use the value that obtained in step 4 to solve the momentum equation again. For this, we designate obtained above as the new values of. Repeat steps 2 to 5 until convergence is achieved.

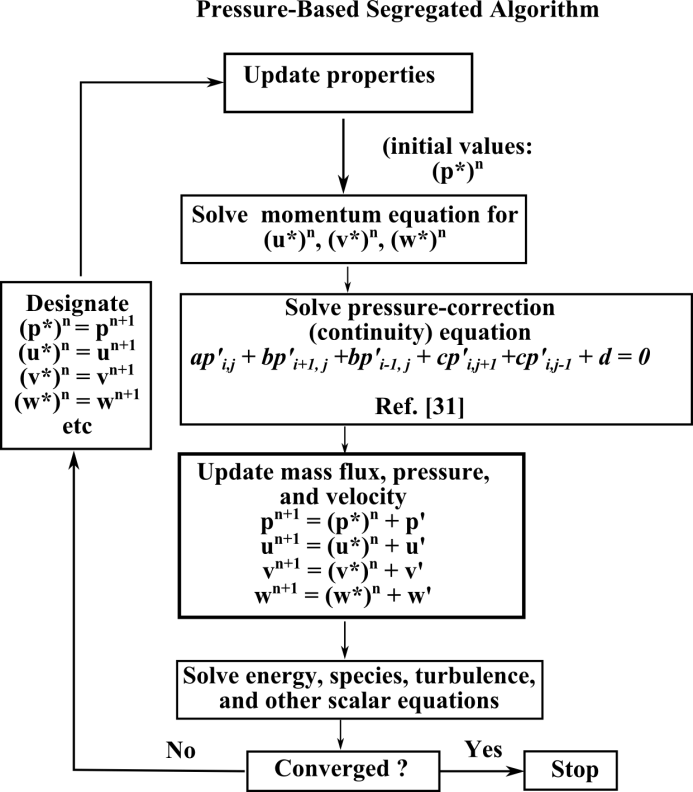


Figure Pressure-Based solution methods (Fluent-Incorporated. 2012, [52])

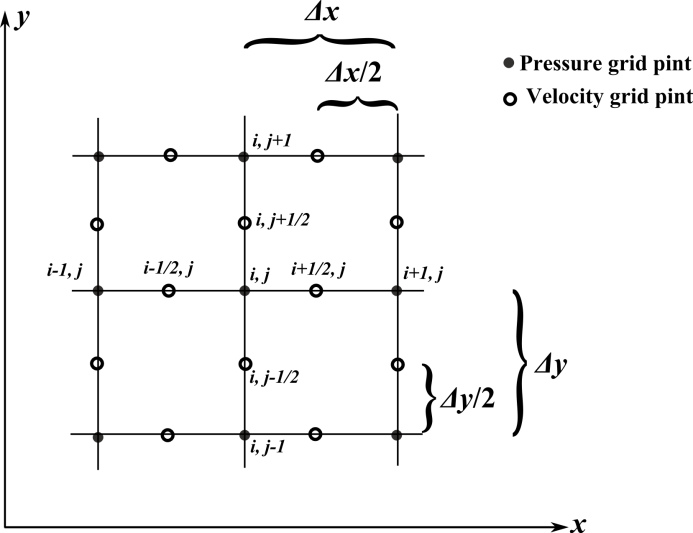


Figure Staggered grid [53]

### Discretization

#### Temporal Discretization

Ansys Fluent uses a control-volume-based technique to convert a general scalar transport equation to an algebraic equation that can be solved numerically. The integral form of general finite volume method, as shown in Eq. 1, can be written in the dual time stepping equation, see Eq. 32. In Fluent, the discrete values of the fluid properties are stored at all cell centres. However, the face values are required for the convection terms in Eq. 32 and need to be interpolated from the cell centre values. This is accomplished using an upwind scheme. Fluent has several different schemes available, such as first-order upwind, second-order upwind, power law, and QUICK[52]. The diffusion terms in Eq. 32 are central-differenced and are always second-order accurate.

|  |  |
| --- | --- |
|  | Eq. |

where is physical time and is pseudo time. Re-arranging the dual time stepping equation, Eq. 32, as shown by:

|  |  |
| --- | --- |
|  | Eq. |

Theoretically, the pseudo term on the left hand side of Eq. 33 should approach to zero at each time step before the physical time marches. On the right hand side, a steady state condition should be satisfied before marching the physical time step.

Temporal discretization is realized using the implicit second-order backward Euler scheme with an iterative procedure. The nonlinear coefficients are updated with each inner loop while the outer loop advances the solution in time. The separate spatial and temporal discretization of the governing equation, Eq. 6, leads for each control volume, to

|  |  |
| --- | --- |
|  | Eq. |

where incorporates the volume, denotes the residual (i.e. the complete spatial discretization including the source term), is the mass Matrix, and index I means the particular control volume. Eq. 34 has to be iterated in time to obtain a steady-state solution (), or to reproduce the time history of an unsteady flow. For unsteady flow, the 3-point backwards Euler scheme with second-order accurate temporal discretization is used, as shown in Eq. 35

|  |  |
| --- | --- |
|  | Eq. |

In , denotes the global physical time step. n+1 is the value at next time level, i.e. t+∆t, n is the value at the current time level, i.e. t, and n-1 is the value at the previous time level, i.e. t-∆t.

#### Spatial Discretization

Based on the grid, control volumes are defined in order to evaluate the integrals of the convection and viscous fluxes as well as the source term. The time derivative of the conservation variables can be written as:

|  |  |
| --- | --- |
|  | Eq. |

The surface integral on the right hand side of Eq. 36 is approximated by a sum of the fluxes crossing the faces of the control volume. This approximation is called spatial discretization. It is usually supposed that the flux is constant along the individual face and that it is evaluated at the middle of the surface. The source term inside the control volume is assumed to be constant. Therefore, for a particular control volume, , Eq. 36 can be written as:

|  |  |
| --- | --- |
|  | Eq. |

The in the Eq. 37 represents the control volume in computational space, are the number of faces around a control volume ( for 3d case). The variable stands for the area of the face. The term inside the square bracket on the right hand side of Eq. 37 is usually called the residual which can be written as:

|  |  |
| --- | --- |
| and | Eq. |

where for a 3D cell with a control volume,, denotes the midpoint of the control volume face (i.e. for a 3D cell, number of faces ), in Eq. 38 is the area of the face , and is the location in space of some point of the control volume .

The cell-centre scheme, as shown in Figure 4 (b), has been used in Fluent, and the control volumes are identical to the grid cells and the flow variables are associated with their centroids [54]. Thus, the cell face is the face between cell and in Figure 4 (b). Therefore, the convective fluxes through the face () reads:

|  |  |
| --- | --- |
|  | Eq. |

where is the artificial dissipation which is added to the central fluxes for stability [55]

The discretization of the viscous flux in Eq. 7 is often calculated from cell interface. The central difference scheme is used for calculating the variables on the cell and. Therefore, the variable on the interface as denoted by can be written as:

|  |  |
| --- | --- |
|  | Eq. |

where are all the flow variables that are stored at the centre of the control volume (e.g. ) and the gradient (such as in the -direction). Xia and Qin [56] investigated and gave the method to evaluate variable gradients in a cell, and to estimation of variables on a face by high order flux reconstruction.

### Fluid-structure interaction theoretical formulation

For flexible wing, the fluid flow and the structure can not be solved separately. The tow domains are needs to be coupled along their common boundaries. Both forces and moments induced by the flow will introduce to the structure domain and the fluid dynamics was coupled to the kinematic and kinetic variables along the structure walls. In order to decouple the physical models and to solve them in a partitioned approach an iterative procedure was introduced as shown in . Fluid and structure were solved alternatively.

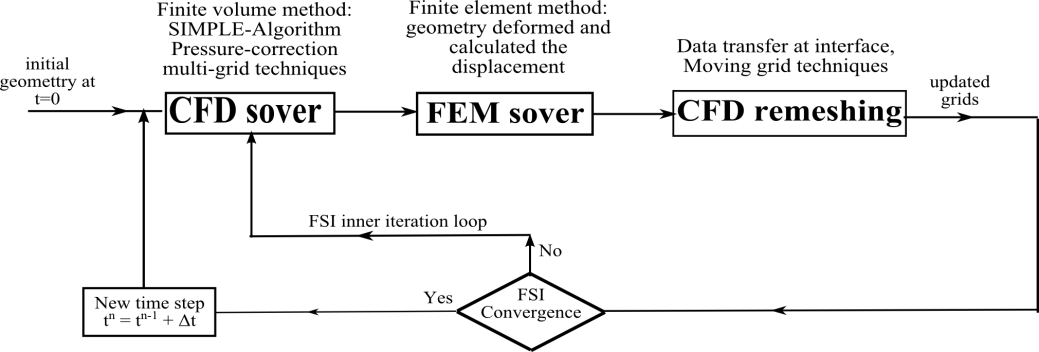


Figure Schematic of the fluid-structure interactions in a partition approach

The Lagrangian algorithm shows that each individual node of the computational mesh follows the associated material particle during motion, and mainly used for the structural mechanics. The Lagrange description allows an easy tracking of free surfaces and interfaces between two different materials. However, the disadvantage of using the Lagrange algorithm is that it needs re-mesh frequently for the large distortions of computation domains.

The Eulerian algorithm is widely used in the fluid dynamics, in which the computational mesh is fixed and the continuum moves with respect to the grid. In the Eulerian algorithm, large distortions in the continuum motion can be handled with relative ease. However, high resolution/precision is required for the interface definitions.

The combination of both Lagrange algorithm and Eulerian algorithm is called the arbitrary Lagrange Eulerian description (i.e. ALE). In the ALE description, the nodes of the computational mesh may be moved with the continuum in normal Lagrange method, or be held fixed in Eulerian method, or be moved in some arbitrary specified rezoning capability. The capability of the ALE is that greater distortions of the continuum can be handled than the pure Lagrangian scheme, and with more resolution than that offered by the pure Eulerian method.

The equilibrium equation to be solved between the solid domain and fluid domain is the conservation of momentum and the conservation of mass, momentum, and energy.

Before introducing both structural and fluid solvers/domains, the general equations for continuum mechanics for both solid and fluid domain will be detailed. This makes an easier understanding on the whole fluid-structure interaction system as because the only difference between their solvers is the constitutive relationships. The equations are cast in an ALE which provides a very general framework that captures the Eulerian, Lagrangian, or an arbitrary frame of reference, Dutsch in 2002 [57], Campbell [58] and Gomes in 2011 [59], and Facci [60] in 2013.

For fluid-structure interaction, the N-S equation can be rearranged for ALE format [54]. For the time-dependent integral form for a moving or deforming control volume with a surface element , can be written as:

|  |  |
| --- | --- |
|  | Eq. |

where vector is the same as shown in . The vector of the convective fluxes becomes on dynamic grids , and is given in , is the contravariant velocity at the face of the control volume, and hence,

|  |  |
| --- | --- |
|  | Eq. |

where represents the grid velocity, and denote the components of the outwards facing unit vector of the surface . Therefore, the convective flux can be rearranged as:

|  |  |
| --- | --- |
|  | Eq. |

where denotes the contravariant velocity relative to the motion of the grid. The vector fluxes and the source term remains the same as shown in and .

Beside the conservation of mass, momentum, and energy, the Geometric Conservation Law (GCL) must be applied in order to avoid errors duo to the deformation of the control volumes, which was originally introduced by Thomas and Lombard [61].

#### Finite element method for structure domain

The Ansys commercial finite element code (ANSYS, 2012) is used as the nonlinear structure solver for this research. The method uses the principle of virtual work combined with the Rayleigh-Ritz solution for the finite element analysis. Therefore, the general form of the equations, in matrix notation can be written as:

|  |  |
| --- | --- |
|  | Eq. |

where , , and are the generalized mass, damping, and stiffness matrices respectively. , , and are the nodal acceleration vector, nodal velocity vector, and nodal displacement, respectively. The term at the right hand-side of , , is the force loading vector, which is responsible for linking the unsteady aerodynamics and inertial loads with the structure dynamics. shows that there are distinct terms representing the structures, aerodynamics, and dynamics displacements. This gives the flexibility in choosing different models for any particular system. There are three different FSI (i.e. fluid-structure interaction) models as mentioned by Kamatoki [62]: fully coupled model, loosely coupled model, and closely coupled model. To resolve , several different algorithms can be employed to obtain the solution. The central difference method is used for the explicit transient analyses which has mentioned by Hallquist [63]. The Newmark method [64-65], as another method, is used for implicit transient analysis, which uses finite difference expansions in the time interval .

In this work, the Newmark scheme [64-65] has been used for temporal discretisation for structure domain. This popular algorithm in structure dynamics is a truncated Taylor series collocation algorithm with quadratic expansion. A minimum second order is required for the problem. A typical one dimension Newmark scheme is showing here:

|  |  |
| --- | --- |
|  | Eq. |
|  | Eq. |

where are the factors which controls the stability and accuracy for the equation. The scheme is implicit for and is uncondionally stable for where. For and, i.e. a constant-average-acceleration or trapezoidal scheme, it has zero dissipation for all choices of time step. Linearising Eq. 44 about time tn to obtain the dynamic solution at time tn+1 gives:

|  |  |
| --- | --- |
|  | Eq. |

where is the external force at tn+1 which in the context of fluid structure interaction is composed of the traction boundary condition imposed by the fluid and any other external applied loads. is the internal force at tn and is composed of the internal stresses within the structure.

Substituting the Newmark equation Eq. 45 and Eq. 46 into Eq. 47, the new equation can be written as:

|  |  |
| --- | --- |
|  | Eq. |

where

|  |  |
| --- | --- |
|  | Eq. |
|  | Eq. |
|  | Eq. |

The changing of the displacement can be obtained by solving Eq. 48, rearranging Eq. 45 to obtain the updated nodal acceleration, velocity and displacement, as shown here:

|  |  |
| --- | --- |
|  | Eq. |
|  | Eq. |
|  | Eq. |

Therefore, the approximation of the current time step’s acceleration, velocity, and the displacement can be obtained from Eq. 52 to Eq. 54, and the updated solution can be achieved based on the convergence criteria.

The updated equilibrium equation is now rearranged from Eq. 48 and Eq. 49:

|  |  |
| --- | --- |
|  | Eq. |
| Where | Eq. |

The convergence criteria is based on the displacement Euclidean norm, as shown in Eq. 57

|  |  |
| --- | --- |
|  | Eq. |

where is the Euclidean norm increased displacement for the *i*th equilibrium iteration, and *xmax*is the maximum total displacement up to and including the current iteration. is the tolerance and typically set at a default value of 10-3.

#### Moving grid method

In both Chapter 5 and Chapter 6, the moving grid technique will be applied on solving the dynamic problems: the rotating/sliding mesh for the propeller-wing aerodynamic interaction case and the dynamic mesh method will be applied for the fluid-structure interaction.

**Sliding moving mesh for propeller simulation**

The sliding mesh model is a special case of general dynamic mesh motion wherein the nodes move rigidly in a given dynamic mesh zone. Additionally, multiple cells zones are connected with each other through non-conformal interfaces (i.e. the non-conformal mesh would let the solid and fluid equations be solved independently from each other with their respective grids, and re-meshing is not necessary[66]). As mesh motion is updated in time, the non-conformal interfaces are likewise updated to reflect the new positions each zone. It is important to note that the mesh motion must be prescribed such that zones linked through non-conformal interfaces remain in contact with each other as this would let the fluid to be able to flow from one mesh to the other. To compute the flux across the non-conformal boundary, it must first compute the intersection between the interface zones that comprise the boundary. The resulting intersection produces an interior zone where the two interface zones overlap, as shown in Figure 9 b and c. In Figure 9 (c), the interfaces are composed of faces A-B, B-C, and D-E, and the intersection of these zones produces the faces a-d, d-b, b-e, and e-c. The overlapping face can be seen in Figure 9 (a) as marked in solid red line and all the faces are grouped to form an interior zone. To compute the flux cross the interface into cell III, face D-E is ignored and instead faces d-b and b-e are used to bring information into cell III from cell I and II.

With respect to dynamic mesh, and for simplicity purposes, the integral form of conservations, as shown in Eq. 41, can be written for a general scalar, , on an arbitrary control volume, , whose boundary is moving:

|  |  |
| --- | --- |
|  | Eq. |

where ρ is the fluid density, is the flow velocity vector, is the mesh velocity of moving mesh, is the diffusion coefficient, and is the source term of In the above equation, Eq. 58, is used to represent the boundary of control volume, .

For the sliding mesh, the mesh motion in the sliding mesh formulation is rigid; all cells retain their original shape and volume. As a result, the time rate of change of the cell volume is zero. Hence, using the first-order back-wards difference formula, the time derivative term in Eq. 58 is:

|  |  |
| --- | --- |
|  | Eq. |
|  |
|  |

where is the mesh velocity of the moving mesh, is the number of faces on the control volume, and is the face area vector.

The sliding mesh movement is known a prior and thus the position of the new coordinates can be analytically computed, as shown in Eq. 60.

|  |  |
| --- | --- |
|  | Eq. |

where is the current node position in Cartesian coordinates, is the updated node location at the next physical time instance, is a vector describing the translation of the nodal coordinates between time steps, and in three dimensions, the rotation matrix, , is defined in Eq. 61.

|  |  |
| --- | --- |
|  | Eq. |

with being equal to the change in angular position of the nodal coordinates about the specific rotation centre between tn+1 and tn (i.e. the rotation centre is shown in Figure 9 a). The rotation direction for this matrix is based on the right-handed rule which first rotates about x-axis, then y-axis, and finally z-axis. However, the propeller is only rotating about the x-axis. Thus the Eq. 61 can be simplified to:

|  |  |
| --- | --- |
|  | Eq. |

The general form of Eq. 60 can be applied for the multiple types of motion, including constant rotational or translation rates, pitching, or plunging. The value of and are computed at each physical time step and Eq. 61 is applied at each node of the mesh at interface boundary, as shown in Figure 9 (c).

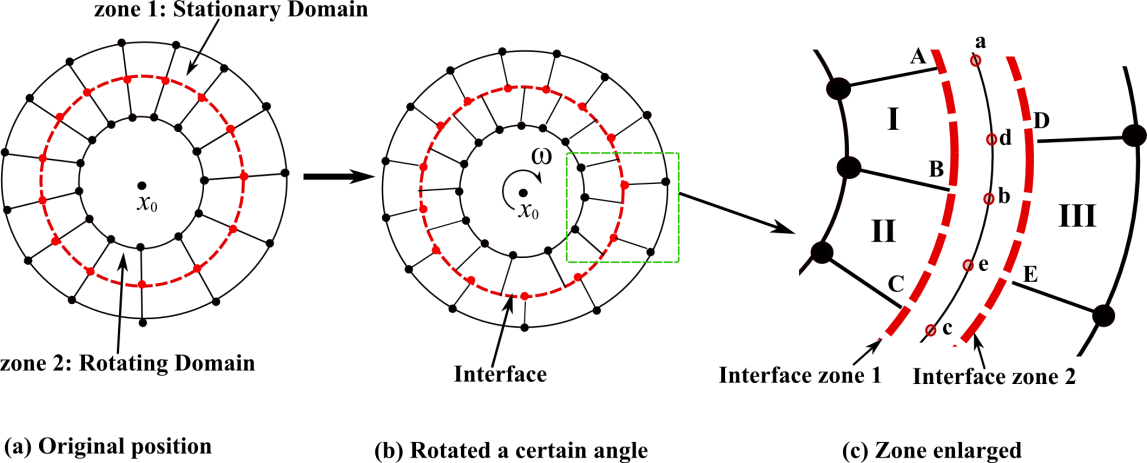


Figure Two dimensional non-conformal mesh interface

**Dynamic moving mesh for FSI**

The fluid-structure interaction, as shown in Chapter 6, requires such moving grid methods for updating the geometry under the aerodynamic loading. Therefore, the boundary condition for the wing will be the moving boundaries, and the dynamic mesh methodology is applied to solve the problem. The dynamic mesh (or moving grids) also can be used for geometry optimization [67-69].

For dynamic meshes, there are two different types of mesh deformations: the algebraic method and the pseudo-structural method [70]. One of the most widely used methods is the spring analogy methodology which belongs to pseudo-structural methods which was originally introduced by Batina [71]. It was designed for the unstructured mesh which has an advantage for treating complex geometries. In this approach, the spring functions are applied to each mesh edge and the deformation of the boundary translates into a deformation of the spring network which updates the geometry to its equilibrium position. The advantage is that re-meshing is not required. However, the disadvantage of using spring analogy is that for highly skewed mesh, negative volume elements often occur which causes the whole program crashes. A further extended version was proposed for structured mesh by Robinson [72]. Another similar approach to make mesh movement is the hybrid mesh with dynamic layering at near wall region to add or remove layers of cells adjacent to a moving boundary, based on the height of the layer adjacent to the moving surface [73]. Overlapping grids method, as another dynamic mesh methodology, allows the grid to move/slide at the overlapping region [74]. The advantage of this approach is that the body-fitted grids do not deform during the body motion. However, the interpolation algorithm (e.g Lagrange interpolation algorithm has been used in ref. [74]) needs to be performing for each overlapping position of the grids.

The spring analogy (or spring-based smoothing) is used for the present investigation. The edges between any two mesh nodes are idealized as a network of interconnected springs. Also Hook’s law is also applied and the force on a mesh node can be shown in Eq. 63:

|  |  |
| --- | --- |
|  | Eq. |

where and are the displacements of the node *i* and its neighbour *j*, is the number of the neighbour nodes connected to node *i*, and is the spring constant(or stiffness) between node *i* and its neighbour *j*. the spring constant for the edge connecting nodes *i* and *j* is defined as:

|  |  |
| --- | --- |
|  | Eq. |

An iterative equation is also required to solve the net force on the connecting nodes to approach its equilibrium position, see Eq. 65:

|  |  |
| --- | --- |
|  | Eq. |

The displacements can be calculated from the initial boundary condition and after deformation (i.e. from updated positions), Eq. 65 are solved using a Jacobi sweep on all interior nodes. At convergence, the positions are updated as following:

|  |  |
| --- | --- |
|  | Eq. |

where *n+1* and *n* are used to denote the position at the next time step and the current time step, respectively.

#### Coupling methodology for fluid and structure solvers

The boundary between fluid and structure is denoted as fluid-structure interface as shown in Figure 10. The term permanent interaction means that both fluid and structure materials are in contact during the whole transient period. On the other hand, no free-surface or Lagrangian fluid-fluid interfaces would be allowed in the permanent interaction cases. In the current work, only the permanent fluid-structure interaction will be discussed.

A contact pressure force, p, is transmitted between the interface of structure and fluid domains, as shown in Figure 10 (b). For a clear understanding of the force interaction between two domains, it is shown separately, because the interaction force is known at each node at each time step, see Figure 11 (a). Therefore, the essential no-slip boundary condition on the interface requires the following:

|  |  |
| --- | --- |
|  | Eq. |
|  | Eq. |

Eq. 67 means the continuity of velocities across the interface and Eq. 68 implies the continuity of displacements. In other words, the fluid nodes and structure nodes have been constrained to remain a contiguous movement, so that all nodes on the sliding interface remain permanently aligned. This is achieved by prescribing the grid velocity v of fluid at the interface to be equal to the material velocity vS of the adjacent structural nodes, as shown in Figure 11 (b), where n is the unit normal on the interface and subscript F and S indicates either the fluid or solid domain.

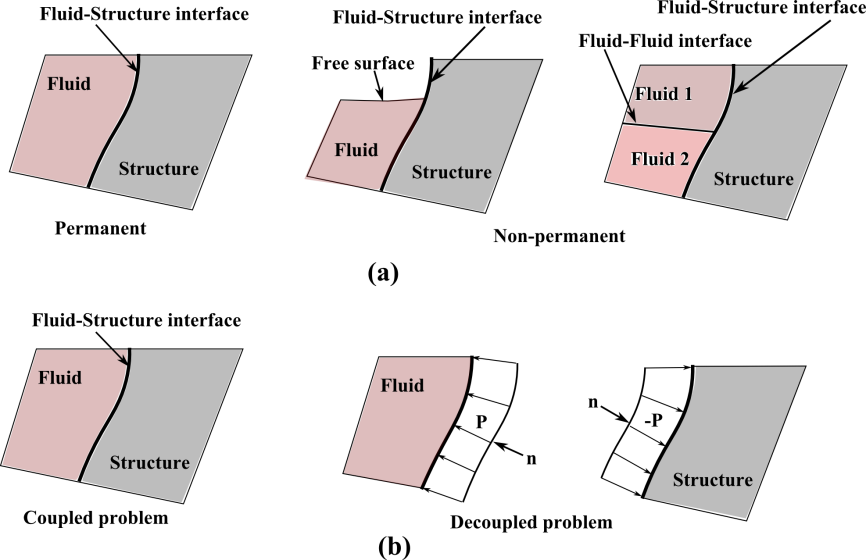


Figure Fluid-structure interaction types: (a) Permanent and Non-permanent, (b) Decoupled problem

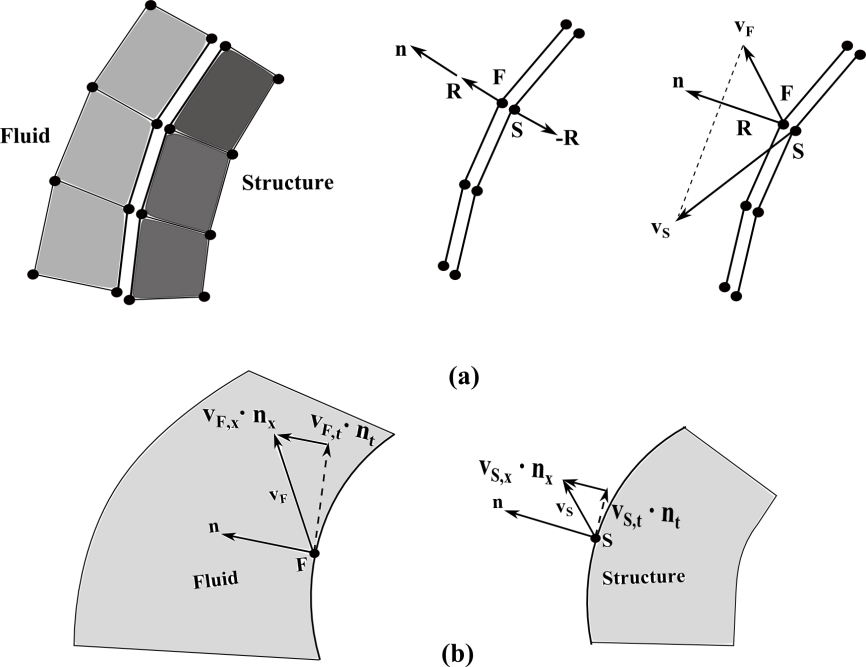


Figure Vector forces on the continuous interface [75]

The discretization of the problem domain is based on a block-structure technique. Fluid and solid parts are assigned to different blocks. The interaction of the fluid and the structure at a mesh interface causes the pressure to exert a force applied to the structure and the structural motions produce an effective “fluid load”. The coupling can be described roughly as following: the structure deformation is caused by the external forces applied by the fluid at the fluid-structure interface. The structure displacements, at the other end, define the geometry and the geometry changes the fluid domain. In a discrete finite element setting, this problem can be described as a coupled three field problem, including the fluid flow, the motion of the fluid mesh and the structural dynamics. The following shows the governing equation:

|  |  |
| --- | --- |
|  | Eq. |

In Eq. 69, the subscript f, s represent the fluid and solid respectively, [M] is the element mass matrix, [C] is the element structural damping matrix, [K] is the element stiffness matrix, is the displacement vector, is the sum of the element nodal force, is the applied pressure vector, and equals the vector of displacements of a general point. The matrices in Eq. 69 are composed of the following:

|  |  |
| --- | --- |
|  | Eq. |
|  |
|  |
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|  |

The subscript and represent the volume of the element and the area over which pressure acts, and [N] is the matrix of the shape function for displacement, [B] is the strain-displacement matrix based on the element shape functions. The matrix [D] is defined as the elasticity or elastic stiffness matrix, as shows in Eq. 71:

|  |  |
| --- | --- |
|  | Eq. |

The coupling methodology of fluid-structure interaction, as mentioned before, can be divided into three different types [62]: a) fully coupled, b) closely coupled, and c) loosely coupled systems. The fully coupled method (or named as monolithic approach by Hou [66]) has a governing equation which combines both structural and fluid equation of motion, and it solves and integrates in time simultaneously. However, this would lead to the computational cost being very expensive, especially for large three-dimensional cases. It is more commonly used with two-dimensional problems. The closely coupled model (or named as partitioned approach by Hou [66]) is one of the popular methods in solving the fluid-structure-interaction problem. Two separated/different solvers are used for structural and fluid domains and the data exchange are designed in one module which makes the whole coupling system tightly. The data exchange is such as surface loading from CFD surface grids mapped onto the structural dynamics grids; this will update the deformed geometry (i.e. change displacement) which maps the structural grids back to the CFD surface grids. However, this mapping-grid method (i.e. see Figure 12) is usually called as a moving boundary and for the complex geometries it is quite often to make the whole coupling system to terminate as the negative volumes are taking place. The loosely coupled models, on the other hand, have separated solvers for fluid and structural domains and the information (i.e. force loading) exchanges after the convergence has been completed. However, it has the advantage of choosing different/separated solvers for structural and fluid domains, and the defects are losing the accuracy for the data exchanging as the data are updated when the convergence is finished.

The emphasis of the above methods is on the coordination of data transfer and consistency between the existing fluid and structure codes. Namely, the fluidic and structural computation will be performed in a sequential manner to achieve a multidisciplinary solution. The challenges one might encounter at the fluid-structure interfaces are to maintain proper data transfer between the disciplines and to reach the convergence efficiently. To increase the accuracy of the data transfer between interfaces, we uses a congruence mesh topology for both fluid and structural domains, as shown in Figure 11. Grid points are mapped from the fluid domain to the surfaces of the structural domain. In other words, the mismatch grids points and gaps can be reduced on the interface of the associated meshes.

In the present study, the closely coupled method is used and as the incidence increases the deformed geometry (or mesh topology for dynamic structure) will be used as the initial geometry for the higher angle of attack cases. For example, for incidence at about 15 deg, results at angle of attack 12deg will be applied as the initial value. This would reduce the overall computational time. The steady fluid-structure interaction of the depron-foam contacted with carbon fibre rods is simulated and the sequence can be found as following:

1. The incompressible three-dimensional Navier-Stokes equations coupled with a transition model are used to solve the fluid domain.
2. The pressure forces are loaded onto the dynamic structural grids by interpolation.
3. The dynamic structure is then solved and its displacement will map back to the CFD moving boundaries.
4. Updating the CFD mesh using the spring analogy.
5. Repeat steps from 1 to 4 until convergence (the maximum displacement converge criteria of 0.2% is used).

The steps from 1-5 are shown in Figure 8. In the present investigation, the implicit coupling approach is used, for which both the structural and fluid solvers exchange information more than once (i.e. there are 10 coupling steps per unit time step) per coupled time step (i.e. sub-iteration loop). Wood et al. [76] showed that the FSI solution based on sequential computation of fluid and structural dynamics becomes unstable, if there are no sub-iteration steps between fluid and structural solvers. One additional sub-iteration would be able to reduce two order of magnitude of numerical error, and more sub-iteration will meet better convergence without a substantial increase in computational time. The number of outer-loop iterations (or number of time steps) is also determined by convergence criteria, and usually 100 iterations are required. At low incidences less iteration are used and as incidence increases the number of iteration increases.

There are several time scales associated with the FSI problem due to the interdisciplinary nature of the problem itself. In general, there are three different time scales[62]: diffusion time scale, convective time scale, and the time scale due to the structural vibration. The first two scales are working with the fluid domain/solver, and the third one is associated with the structure, the formulations are defined as following:

|  |  |
| --- | --- |
|  | Eq. |

where is the local mesh size, is the local characteristic speed, is the structure frequency, and is the diffusion coefficient. The physical time scale as normally used in the dual-time stepping problems, and for many simulations, a reasonable time estimated *t* is easy to make based on the characteristic geometry length L, and the mean velocity U, or . The physical time scales that are too large are characterized by bouncy convergence or results that do not converge. The time scales that are too small normally offer a very slow, steady convergence. For unsteady case, a proper time scale should consider in order to satisfy the stability and accuracy issues of using different solvers. There are no further limitations on time scales if both fluid and structural solvers use the implicit scheme. However, time stepping size becomes an important factor if different solvers are used for structural and fluid solver, such as Newmark method (i.e. implicit scheme) for structural solver and PISO (i.e. semi-implicit scheme) is selected for fluid solver. Consequently, the stability condition is only treated for the fluid solver, the CFL (or Courant) number is applied to improve the stability, defined by:

|  |  |
| --- | --- |
|  | Eq. |

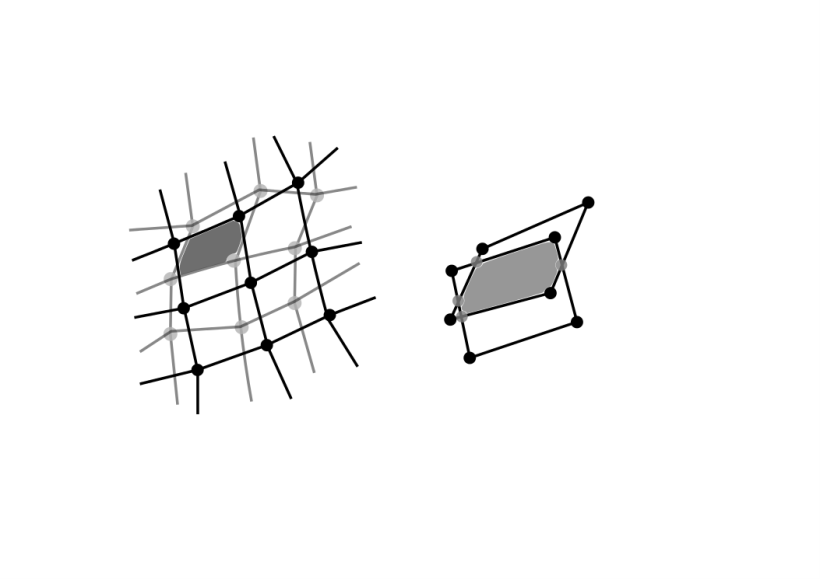


Figure 12 Mapping-grid method

### Boundary conditions for Navier-Stokes equations

The physical boundary condition, another important issue, needs to be applied correctly for different boundaries or cases. This is because the nature of the equations and their domains of dependence and zones of influence have implications for boundary conditions. Figure 13 (a-b) shows a typical example of the MAV CFD domain with different boundaries condition applied. The velocity-inlet boundary condition is applied on the front, top, bottom, and the side faces. The pressure-outlet is applied on the back-face, symmetry is for the centre plane and the no-slip wall boundary condition is applied on the micro air vehicle surfaces. and Eq. 75 show the velocity and pressure boundary conditions for the MAV surfaces.

|  |  |
| --- | --- |
| at the surface | Eq. |

The contravariant velocity () is zero at the MAV surfaces. Consequently, the vector of convective fluxes reduces to the pressure term alone, i.e,

|  |  |
| --- | --- |
|  | Eq. |

with being the wall pressure.

At the inlet, the velocity is given () and the pressure is set as the atmosphere pressure.

At the symmetry plane, there is no flux across the boundary. This is equivalent to the requirement that the velocity to the symmetry boundary is zero. Furthermore, gradients in and Eq. 77 have to vanish:

|  |  |
| --- | --- |
|  | Eq. |
|  | Eq. |

where stands for a scalar variable. and denote a vector tangential or normal to the symmetry boundary, respectively.

|  |  |
| --- | --- |
| C:\Users\jason\Desktop\export.jpg   1. Outer domain | C:\Users\jason\Desktop\export2.jpg   1. Wall boundary |

Figure boundary conditions for a CFD domain

# The Laminar Separation Bubble (LSB)

The rapid development in materials, micro fabrication, electronic devices, sensors, controllers, and actuators has led to the convergence of the micro air vehicle (MAV) designs to smaller and smaller sizes. As MAVs are easily manufactured, light, portable and low cost, there are strong interests for both civil and military applications. However, the challenges of the MAV aerodynamic design has also become more prominent, including better understanding of three dimensional low Reynolds number aerodynamics and a large number of design parameters, such as the planform, aerofoil profile, camber, aspect ratio, wing sweep, anhedral or dihedral, and wing warping effect as mentioned by Pines and Bohorquez [23].

A commercial airliner wing typically has a chord Reynolds number of 107 to 108. At the other end, small MAVs have typical low Rec around 104 and 105. Due to this significant difference, the aerodynamic design principles for large aircraft with high Reynolds numbers are not directly applicable to such small vehicles and the conventional aerofoil aerodynamic performances are degraded dramatically when they are used for these small low speed vehicles due to the much stronger viscous effects. For MAVs operating at a low Reynolds number, laminar flows dominate assuming very small adverse pressure gradient without separation. As the incidence increases, flow separation, transition and reattachment appear, which affect strongly the flow structure and hence the lifting surface performance. Lian *et al.*[25] and Swanson *et al.* [77] indicated that transition can occurs for Rec ≥ 104 and therefore a transition model is required in this flow condition.

Laminar separation bubbles have been studied on two-dimensional geometries for many years, but the phenomenon is still not fully understood. The main reason is that any disturbances would affect the low Reynolds number boundary layer characteristics, and thus relocate the separation, transition, and reattachment positions [23]. The location of transition is clearly important for the development of low Reynolds number aerofoils and natural laminar flow (NLF) aerofoils as discussed in Obara and Holmes [78]. Accurate experimental data for integrated force coefficients (lift and drag) are also more difficult and challenging to obtain for low Reynolds number aerofoil flows due to the low speed and small dimension. Furthermore, for velocity profile measurement, Batill and Mueller [79] pointed out that to minimize wake effects to the wing the Reynolds number of a hot-wire should be less than 20 (hot wire should have a limited diameter).

Only a few experimental studies have been conducted on laminar separation bubbles with three dimensional characteristics, while experiments on two dimensional wing sections with end plates have been investigated by many researchers to simulate the two dimensional wing flows. The main reason is because the laminar separation bubble is very sensitive to the experimental conditions, such as freestream turbulence intensity level [80-81], experiment devices installation, *etc.* The bubble may also become unsteady which will make the flow structure even more complex. One such experiment with end plates was performed by Gaster [82]. He investigated the structure of a bubble and concluded that the structure of the bubble depended on the value of the Reynolds number of the separating boundary layer and a parameter () based on the pressure rise over the region occupied by the bubble. From an experiment without the endplates, Batill’s [79] showed that the bubble is highly erratic and three dimensional along the wingspan on a two dimensional aerofoil cross section upper surface at an incidence of 8° with Reynolds number of 55,000. Horton [83] mentioned that any three-dimensional flow such as that due to tip effects or sweep-back will lead to situations in which the bursting process takes place at different values of Reynolds number or incidences at different spanwise positions. An experimental study on three dimensional bubbles was conducted by Bastedo [84]. His experiment was on a Wortmann FX63-137 aerofoil section with an aspect ratio of two. The bubble was observed to be three dimensional towards the wingtip due to the tip effect. Tezuka *et al.* [85] conducted wind tunnel tests on a cambered plate aerofoil with endplates. The surface oil flow results showed that LSB formed at moderate angles of attack. The length of pressure plateau increased, and the chordwise position of the pressure recovery point moved downstream as incidence increased. Namely, the bubble length extended as incidence increased.

On the numerical side, Wilcox [35] proposed to account for transition within the k-ω two equation turbulence model, showed reasonably accurate description on several transitional boundary layers. Rumsey [86] pointed out the importance of appropriate transition modeling for low Reynolds number flows.

In the present study, the phenomenon of the laminar separation bubble on a three dimensional cambered thin wing at Reynolds number of 60,000 with various angles of attack is examined in comparison with the corresponding 2D situation. The aim is to gain some further insight into the development of the three dimensional laminar separation bubble and its spanwise variation due to interaction with the wingtip vortex, hence its effect on the overall aerodynamics performance of the wing.

### Geometry specification

Both a 2D cambered plate aerofoil and a 3D rectangular wing with the same aerofoil are studied in the present work according to the wind tunnel experiment by Mueller [15] and Pelletier and Mueller [8] . For the wind tunnel tests, the aerofoil chord was 100.1mm and the wing has a span of 304.7mm. The freestream Reynolds number based on the chord length was 60,000. For the 2D tests, two endplates were used to confine the wing-tip flow. For the 3D tests, an endplate was applied on one end of the wing and the other end was left free. In our numerical study of this 3D wing case, a symmetry condition is applied at the centre of the whole wing. This represents a 3D wing with an aspect ratio of 6.088.

|  |  |
| --- | --- |
| J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\1st_time\ICEM\cambered_airfoil.emf  2D aerofoil geometry, ref. [8] | J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\2D_mesh.jpg  2D mesh at leading edge regime |

Figure 14 Two-dimensional geometry [15]and2D mesh at leading edge regime

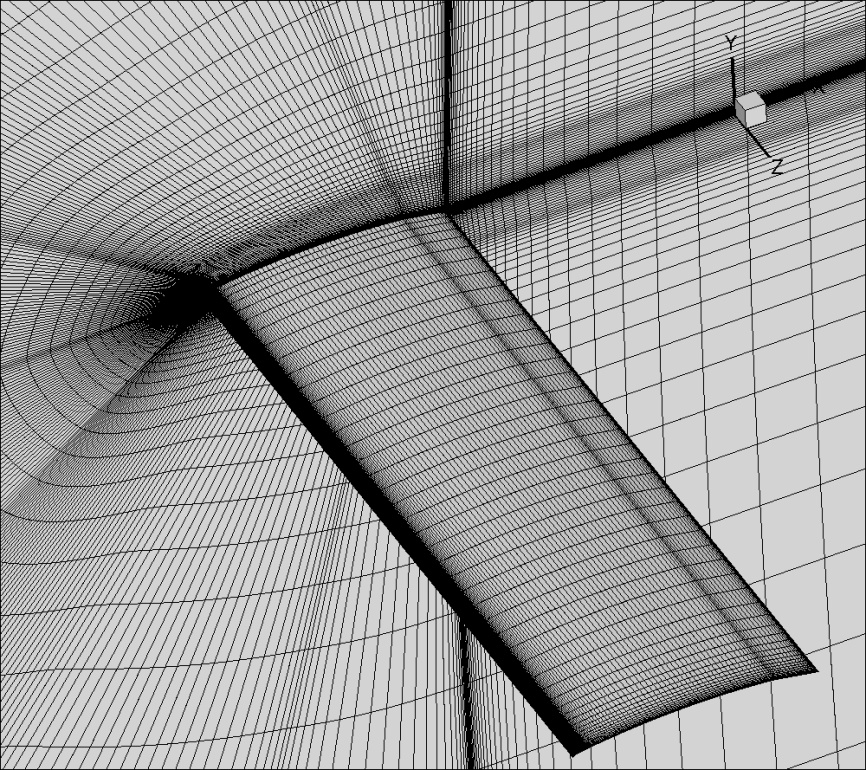


Figure 15 3D mesh

The aerofoil profile, as shown in a), has a thickness-to-chord ratio of 1.93%, and the model was designed to have a 5-to-1 elliptical leading edge and a 3-deg tapered trailing edge. The camber line has a circular arc shape with 4%c camber.

The ANSYS ICEM mesh program was used to generate the structured meshes for the present numerical study. b) shows the mesh topology for the 2D aerofoil, and Figure 15 is the mesh topology for the 3D wing.

The uncertainties are also provided in Pelliter and Mueller’s paper [8]. Experiment results, and the maximum forces that can be measured with the balance without engaging the second set of flexures are 14.7N for lift and drag and 226N·cm for the pitching moment, whereas the minimum measurable loads are approximately 0.01N for the lift and drag and 0.05N·cm for the moment. The actual error for the force coefficients at different incidences were provided by Pelliter and Mueller, which are used in the following presentation of the experimental results. The freestream turbulence intensity of 0.05% was stated for all experiments and, therefore, it will be used for this investigation.

### Mesh sensitivity analysis

Due to the computational demand, the mesh-sensitivity study is limited to the 2D case for α = 5° including the phenomena of laminar separation, transition and turbulent reattachment. The present computational simulations use a C-type mesh around the 2D cambered plate, as shown in (a) and (b). To minimize the farfield boundary condition effects, the domain is set at 25c upstream, 35c downstream, and the upper and lower boundaries are placed at 25c, all from the aerofoil leading edge.

The mesh sensitivity is shown in .The coarsest mesh has 190 grid points around the aerofoil and 95 in the normal direction. Close to the wall, there are about 30 grid points within the boundary layer, and in the turbulent region, the y+ value of the first cell distance is ensured to be in order of 1.The stretching ratio for the mesh is less equal than 1.2.

shows the sensitivity of lift, drag, LBS separation, transition, and reattachment points. For the meshes studied, the wall normal direction resolution of 95 seems to be sufficiently good. Higher sensitivity is shown for the mesh resolution in the steamwise direction around the aerofoil, requiring the finest resolution of 470. From this mesh sensitivity study, Mesh 3, the grid with 470×95 points, is chosen for all computations reported here, as it gives sufficiently accurate results in the lift coefficient (~O(10-3)), drag coefficient (~O(10-4)) and separation, transition and reattachment locations in comparison with the finest mesh.

Table Grid-sensitivity at α 5° (NR = Not reported)

|  |  |  |  |  |  |  |
| --- | --- | --- | --- | --- | --- | --- |
| Mesh | Grid size | Cl | Cd | XS/c (%) | XTr/c (%) | XR/c (%) |
| 1 | 190×95 | 0.806 | 0.0459 | 0.92 | 15.2 | 24.5 |
| 2 | 350×95 | 0.798 | 0.0469 | 0.99 | 13.4 | 19.9 |
| 3 | 470×95 | 0.815 | 0.0476 | 0.99 | 13.6 | 19.9 |
| 4 | 190×150 | 0.799 | 0.0477 | 0.92 | 15.2 | 24.5 |
| 5 | 350×150 | 0.815 | 0.0476 | 0.94 | 13.6 | 19.9 |
| 6 | 470×150 | 0.816 | 0.0478 | 0.94 | 13.6 | 20.1 |
|  |  |  |  |  |  |  |
| Experiment |  | 0.84±0.035 | 0.0504±0.035 | NR | NR | NR |

### LSB on a 2D cambered plate aerofoil

For comparison with 3D LSB, a series of 2D simulations were carried out for the same aerofoil. The lift and drag coefficients at different incidences are plotted in Figure 16 and Figure 17respectively for 2D flow simulations in comparison with the experimental data. The numerical results show reasonably good agreement in lift and drag with the experiment measurements until the aerofoil stalls at about 14° angle of attack. The drag values are slightly under-predicted outside the experimental uncertainty range at higher incidences. Note that the 2D experimental data were obtained by implementing two endplates on the 2D sectional wing. The discrepancies could be due to the difference in the experimental model and the numerical model, in particular, the endplate effects and the gap between the wing and the endplates. Pelletier and Mueller [87] found that the endplate can lead to an increase in CD and a reduction in CL at low Reynolds numbers.

By a closer examination of the lift curve, a sudden jump (change of the slope) can be observed at an incidence about 5° in both the simulation and the measurement by Pelletier and Mueller [8]. This phenomenon was not discussed or explained before. By associating this with the development of the LSB, we found that this lift enhancement is due to the sudden enlargement of the laminar separation bubble as the incidence is increased near this angle. The corresponding pressure distribution and skin friction coefficient on the suction side, found in (a-d), confirm this correlation. The expansion of the LSB causes the pressure plateau (suction) to enlarge on the upper surface, which in turn creates extra lift on the aerofoil.

In general, the 2D flow structure for this particular cambered thin aerofoil can be classified into three major different types corresponding to incidences. To help understanding, Figure 19 shows the flow streamline structure variation with incidences.

At α = 0°, the flow is attached to about 83%c (i.e. as shown in )with a small portion of trailing edge separation on the upper wing surface. Note that a long LSB (about 40%c) forms at the lower surface at 0° incidence for this cambered plate profile. The lower surface LSB disappears at α = 1°.

In the low incidence range, α = 1° to 4°, the flow structure is dominated by trailing edge separation, which enlarges gradually as the incidence increases. Note that a very small separation region is present near the leading edge for the 4° case, which is insignificant to aerodynamic performance.

At medium incidences, α = 5° to 8°, the leading edge separation bubble dominates the flow structure, which is accompanied by the disappearance of the trailing edge separation. LSB starts to form on the upper surface as the angle of attack increases to 5°, which is also reflected in the formation of the pressure plateau in Figure 18 (a) and the negative skin friction region in Figure 18 (b). Note that the suction plateau drops as the transition starts (around 10%c) within the LSB. As the incidence increases, the LSB increases both its length and thickness accompanied by the increase of the suction plateau. The maximum suction plateau is achieved at about 7°. The higher the plateau value the shorter the plateau length because as incidence increases the transition location moves forward slightly towards the leading edge.

For higher incidences, α = 9°-13°, the streamlines on the upper surface show both LSB and trailing edge separation. The extended leading edge bubble met up with the trailing edge recirculation to form a long and thin separated region, which occupies the whole upper wing surface with two distinct recirculations at the leading and trailing edges. Further increase of the incidence thickens the separated region, enlarge the trailing edge recirculation significantly and eventually a single very large recirculation on the upper surface forms. The pressure suction plateau reduces substantially for higher incidences. Note that the divergence of the lift curve between the numerical and experimental beyond 13° indicates the incapability of the current numerical model for stalled aerofoil flow.

From the above discussion, except the peculiar case of 0°, the low Reynolds number flow around the 2D thin aerofoil can be categorized into three distinct flow structures according to the incidence range: (i) trailing edge separation; (ii) leading edge separation bubble, and (iii) large separation including both, evolving to massive separation.

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Figure 16 2D lift coefficient, Cl

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Figure 17 2D drag coefficient, C*d*

|  |  |
| --- | --- |
| J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\Cp3_TSST_Re6_2-12U.emf   1. Cp: upper surface | J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\Cf3_TSST_Re6_2-12U.emf   1. Cfx: upper surface |

Figure 18 2D cambered plate aerofoil: a) Cp on suction side, b) Cfx on suction side

|  |  |  |
| --- | --- | --- |
| J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\steady2\post_process_pressure_outlet\0deg.jpg  α = 0° | J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\steady2\post_process_pressure_outlet\1deg.jpg  α = 1° | J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\steady2\post_process_pressure_outlet\2deg.jpg  α = 2° |
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| J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\steady2\post_process_pressure_outlet\9deg.jpg  α = 9° | J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\steady2\post_process_pressure_outlet\10deg.jpg  α = 10° | J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\steady2\post_process_pressure_outlet\11deg.jpg  α = 11° |
| J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\steady2\post_process_pressure_outlet\12deg.jpg  α = 12° | J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\steady2\post_process_pressure_outlet\13deg.jpg  α = 13° |  |

Figure 19 Streamlines around 2D aerofoil at various incidences

### LSB on 3D cambered thin wing

The geometry planform view, shown in Figure 20, illustrates the various definitions for the 3D study. The wing section profile is the same as that in the 2D study. The freestream incoming direction is from the top, as indicated by Ux in the following presentation. The pressure and skin friction distributions on the upper wing surface is extracted from three different spanwise locations (i.e. wing root, wing tip and z/b = 0.6). A C-H mesh topology is used for the 3D wing as shown in Figure 15 and the wingtip is modeled to be flat. According to the mesh sensitivity study for the 2D cases, the mesh resolution is chosen as 470 grid points wrapping around the aerofoil, 95 in the normal direction, and 50 in the spanwise direction. The flow structure are studied and analyzed from 0° to 12° incidences. The wall and farfield boundary conditions are similar with those for the 2D cases as mentioned before. A symmetrical boundary condition is applied at the wing root, simulating a wing with an aspect ratio of 6.088. Due to the 3D effects, the comparison with 2D is carried out at the symmetry plane.

|  |
| --- |
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Figure 3D Wing planform

### Aerodynamic performance

The aerodynamic results of the 3D camber-wing (Figure 21 and Figure 22) show reasonably good agreements with the experimental data. Very good agreement is shown for the lift coefficient before the predicted wing stall at 11°. The divergence of the lift curves between prediction and experiment occurred earlier as compared to the 2D case. The drag values are within the experimental uncertainty range but show a tendency of under-prediction for very low and high incidences. This could be because the drag value measured at the centre of the wing from the experiment which also includes interaction due to the end plate installed at wing root (In the experiment, an endplate (or wind tunnel wall) is placed at the wing root to eliminate the wing tip effects).

Interestingly, again, a sudden jump of the CL occurs in both the experimental data and prediction at an incidence of 6°, as shown in Figure 21, which is one degree later than the 2D case. This phenomenon indicates that the LSB is enlarged at this particular incidence with an associated pressure suction plateau, enhancing the lift. Due to the 3D effects on the finite wing, this lift enhancement feature of the LSB is delayed.

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Figure CL for 3D cambered thin wing

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Figure CD for 3D cambered thin wing

#### Flow structure at low incidences (0°-4°)

The flow structures at low incidences are shown in Figure 23 and Figure 24. The pressure coefficient (Cp) and skin friction distribution (Cfx) at three different spanwise locations and different incidences are extracted along with the 2D distribution. The surface flow patterns (skin friction lines) are also shown to identify separation and reattachment lines.

The flow structure on the upper surface indicates that although the patterns are dominantly 2D away from the near tip region, some variations in the pressure and skin friction profiles and the trailing edge separation location along the span can be easily identified. The trailing edge recirculation reduces towards the tip due to the interaction with the wingtip vortex.

Similar to the 2D case, at 0° a bubble forms on the lower surface, as shown in Figure 23 (d). Compared with the upper surface, the spanwise variation is much more significant with the presence of the lower surface LSB, as shown in Cp and Cfx in Figure 23 (a-b). From the skin friction plot and stream lines at the symmetry plane, the 3D LSB is much smaller with spanwise variation. The transition locations calculated from 2D and 3D are also quite different from each other, with transition points occurring at 35%c and 25%c for 2D and 3D, respectively.

At 4°, there is no LSB for the 3D wing while a tiny LSB can be observed to form for the 2D aerofoil, as shown in at 4° and Figure 24 (a-d). There is also a significant difference in the trailing edge separation, 83%c at the wing root for the 3D case, and 65%c for the 2D aerofoil, found in Cfx distribution in Figure 24 (b). The surface flow pattern in Figure 24 (c) indicates that the spanwise flow becomes stronger in comparison with that for lower incidence cases. The tip effect pushes the trailing edge separation line backward towards the trailing edge. This trailing edge separation line trend is opposite to that for a thicker aerofoil case (NACA0012) reported by Huang [88] in his experiment results with Reynolds number from 20,000 to 100,000, where the wingtip effect causes the separation line to curve towards the leading edge.

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| J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\2d_3d_compare_0deg_2.emf   1. Cp distribution | J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\2d_3d_compare_0degCfx_2.emf   1. Skin friction coefficient distribution |
| J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\3D\0deg.png   1. Surface flow pattern on upper wing (left: tip; right: root) | J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\3D\slice_0deg.png   1. Streamline pattern at wing root |

Figure Pressure, skin friction and streamline patterns at α = 0°

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| J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\2d_3d_compare_4deg_2.emf   1. Cp distribution | J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\2d_3d_compare_4degCfx_2.emf   1. skin friction coefficient distribution |
| K:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\3D\4deg.png   1. Surface flow pattern on upper wing(left: tip; right: root) | J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\3D\slice_4deg.png   1. Streamline pattern at wing root |

Figure Pressure, skin friction and streamline patterns at α = 4°

As the incidence increases to 5°, the 3D LSB starts to form. The small LSB forms close to the leading edge, with a length about 5%c and roughly one third in the spanwise direction (Lbc/b =0.3) from symmetrical plane (Figure 25 a-d). However, this LSB does not change lift much without forming a significant suction plateau, Figure 25 (a-b). A fully attached flow takes place near the leading edge region further towards the wingtip area. The recirculation region appears at the trailing edge and it starts approximately at x/c = 0.73 at root chord and moved towards trailing edge near the wingtip. As incidence increases, stronger spanwise flow of the trailing edge separation seems to occur (Figure 25 c). This is responsible for bending the streamline towards the symmetrical plane at higher incidences. For the 2D case, however, a more substantial bubble forms at leading edge with a length of 20%c. The lift slope, in Figure 16, increased dramatically. Trailing edge recirculation is found at roughly 85%c, but it occurs at 73%c for the 3D wing. The flow separated from the side edges which form vortex sheets and curl up over the wing into a vortex ( Figure 25c), and this foci moves towards the wingtip as the incidence increases (α from 6° to 10° ).

At 6°, LSB increases and the trailing edge separation is pushed back. Strong 3D features are observed for LSB and TSB. A junction flow appears at the centre of the wing area due to the stronger interference between LSB, wingtip vortex, and trailing edge recirculation (Figure 26 c). The LSB length increases the chordwise length as incidence rises, opposite to the tendency for conventional aerofoils. At this particular incidence, α = 6°, the low-α trailing edge recirculation flow regime changes to the high-α LSB-dominated flow regime. As compared with 2D, both LSBs have a similar length of 20%c. However, the 2D bubble has a much higher pressure suction plateau value (-1.6) than the 3D bubble at wing root (-1.1), contributing to higher lift generation. The 2D bubble has an earlier transition location of 10%c, whereas the 3D bubble has a delayed transition at 16%c, as shown in Figure 26 (a-b). Close to the tip, a strong spanwise flow component is induced by the tip vortex, modifying the chordwise pressure distribution dramatically from that for the two dimensional case.

Figure 26 (e) presents the critical/singular points on the upper wing surface. They are saddle points (as indicated by s) at z/b = 0.88 and z/b = 0.45, and attracting focus point (focus of separation, and as indicated by F) at z/b = 0.52. As defined by Perry [89], critical/singular points are those points at which the magnitude of the vector vanishes, and these points may be characterized according to the behaviour of nearby tangent curves. According to Tobak [90] these critical/singular points include two different classifications: nodes and saddle points. Nodes may be further subdivided into two subclasses: nodal points and foci (of attachment or separation). The focus invariably appears on the surface in company with a saddle point. Beside this, a separation line normally comes with the saddle point, as discussed in Lighthill [91].

Three bifurcation lines are also clearly marked on the upper wing surface, where the flow separates, reattaches and separates again, near the leading and trailing edges. These are marked as BL- and BL+ respectively ( Figure 26 e).

At incidence of 7°, the maximum chordwise length (Lbc/c) of the LSB shrinks to less than 0.2 in contrast to the previous incidence. However, it extends spanwise to almost the full wingspan and the LSB shape is much more uniform in the spanwise direction away from the tip. The pressure and skin friction distribution in Figure 27(a, b) also show the almost uniform bubble across the wingspan, and the bubble length is approximately 20%c. However, the pressure plateau level decreases towards the wingtip. The bubble on the 2D aerofoil has a higher pressure plateau value and the bubble length is also 4%c longer than the 3D bubble at the wing root. The transition location is around 10%c for both cases. The surface streamline pattern in Figure 27(c) indicates that the separation line has moved towards the leading edge and the reattachment line has relocated roughly to x/c = 0.2. The distance between the saddle point and the focus of separation point becomes shorter, and the saddle point moved inboard due the stronger wingtip effect at higher angles of attack and the attracting focus point moves outboard.

The more spanwise uniform LSB with a taper shape at wingtip section appears near the leading edge as the incidence increases to 8°. The flow separated at the leading edge (Figure 28 c), and the LSB has a chordwise length of 20% chord, occupying the full spanwise length. An almost uniform pressure plateau with reduced level along the span (Figure 28 a), which indicates a uniform LSB with wingtip effect accounted. This is evident from the skin friction plot, Figure 28 (b), which shows a uniform bubble with a chordwise length around 20%c. The same pressure plateau is found for both 2D and 3D cases (Figure 28 a). However, the 3D skin friction is much higher than that for 2D. At this angle of attack, both saddle points and the attracting focus have relocated to the wingtip area, which may be because the influence of the tip vortex is reduced as the incidence increases. The LSB separation, reattachment positions on the upper wing surface and the transition locations are listed in

Table 2 and .

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| J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\2d_3d_compare_5deg_2.emf   1. Cp distribution | J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\2d_3d_compare_5degCfx_2.emf   1. Skin friction distribution |
| K:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\3D\5deg.png   1. surface flow pattern on upper wing (left: tip; right: root) | |
| J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\3D\slice_4deg.pngJ:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\3D\slice_5degH.png   1. Streamline pattern at wing root (Leading edge separation bubble starts to form) | |

Figure Pressure, skin friction and streamline patterns at α = 5°

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| J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\2d_3d_compare_6deg_2.emf   1. Cp distribution | J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\2d_3d_compare_6degCfx_2.emf   1. Skin friction distribution |
| K:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\3D\6deg.png   1. surface flow pattern on upper wing (left: tip; right: root) | |
| J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\3D\slice_6deg.pngJ:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\3D\slice_6degT.png   1. Streamline pattern at wing root (trailing edge separation is reduced) | |
| *K:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\3D\6deg_topology3.jpg*   1. Flow topology at 6° (left: tip; right: root) | |

Figure Pressure, skin friction and streamline patterns at α = 6°

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| J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\2d_3d_compare_7deg_2.emf   1. C p distribution | J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\2d_3d_compare_7degCfx_2.emf   1. Skin friction distribution |
| K:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\3D\7deg.png   1. Surface flow pattern on upper wing (left: tip; right: root) | J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\3D\slice_7deg.png   1. Streamline pattern at wing root |

Figure Pressure, skin friction and streamline patterns at α = 7°

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| J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\2d_3d_compare_8deg_2.emf   1. Cp distribution | J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\2d_3d_compare_8degCfx_2.emf   1. Skin friction distribution |
| K:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\3D\8deg.png   1. Surface flow pattern on upper wing (left: tip; right: root) | J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\3D\slice_8deg.png   1. Streamline pattern at wing root |

Figure Pressure, skin friction and streamline patterns at α = 8°

Table separation, transition, and reattachment location of the LSB

(NR = not reported, LE = leading edge)

|  |  |  |  |  |  |  |
| --- | --- | --- | --- | --- | --- | --- |
| Results from spanwise section at wing root  Re = 60,000 | | | | | | |
| AoA (°) | XS/c (%) | | XT/c (%) | | XR/c (%) | Lbc/c (%) |
| 0 | | Bubble formed on lower wing surface | | | | |
| 4 | | | | No LSB is formed | | |
| 5 | 5 | | 5.8 | | 9 | 5.0 |
| 6 | 0.9 | | 18.7 | | 23.2 | 22.3 |
| 7 | 0.5 | | 11.9 | | 17.5 | 17.0 |
| 8 | LE | | 10.4 | | 20.1 | 20.1 |
| 9 | LE | | 10.4 | | 24.9 | 24.9 |
| 10 | LE | | 11.9 | | 38.1 | 38.1 |
| 11 | LE | | 12.9 | | 99.7 | 99.7 |
| 12 | LE | | 12.9 | | 99.8 | 99.8 |

Table 3 LSB maximum spanwise length with various incidences at Re of 60,000

(FS = full span)

|  |  |  |
| --- | --- | --- |
|  | AoA (°) | Lbz/z (%) |
| Re 60,000 | 4 | None |
| 5 | 30 |
| 6 | 82 |
| 7 | 95 |
| 8 | FS |
|  | 9 | FS |

#### Surface flow patterns at high incidences (9°-12°)

The LSBs at higher incidences (from 9°-12°) are shown in Figure 29(a-d). The bubble length extends as the angles of attack increases. As the incidence increases further, the leading edge bubble merges with the trailing edge separation to form a long bubble which covers the whole upper wing surface chord. For the 3D wing, this happens progressively from inboard towards the wingtip. Both the leading edge bubble and the trailing edge recirculation regions expand. The LSB varies spanwise clearly and the relevant pressure plateau can be found in Figure 29 (a) and Figure 30 (a) for incidences of 9° and 10° respectively. The corresponding separation, transition, reattachment points can be found in . The pressure plateau has found to be much lower at higher angles of attack, as shown in Figure 31(a) and Figure 32 (a). However, this pressure plateau is higher than the 2D case, except at the wing tip region.

A further expansion of LSB in its chordwise length happens at an incidence of 10°, with the bubble length increasing to its maximum value of 38% of chord, which is almost twice as longer as the bubble present at lower incidences. This bubble is a long bubble, affecting the lifting surface significantly, which also grows rapidly with increasing incidence until it extends over the entire chord.

At incidences of 11° and 12°, large recirculation occupies on most of the upper wing surface, as shown in Figure 31 (c) and Figure 32 (c). At incidence of 11°, the reversed flow occupies the whole chord at wing root, which can also be found from the skin friction distribution in Figure 31 (b). Towards the wing tip, the LSB and trailing separation bifurcate with attached flow in between, shown in Figure 31 (c). The transition point stays at the roughly the same location along the wing span around 10% of the chord. As the incidence increases to 12°, shown in Figure 32 (c), the merged flow occupies the whole chord at the inboard area, and the bubble has a spanwise length of nearly two thirds of the wing span. Figure 32(e) shows the 3D flow structure on the upper wing surface, which clearly indicates the spanwise variations. The strong leading-edge-tip corner induced vortex feeds into the inboard flow recirculation.

Care needs to be exercised in the above analysis in the high incidence range. The comparison with the experimental data indicates the incapability of the numerical modeling in predicting the wing stall correctly beyond 11°. Therefore the merge of the LSB and the trailing edge separation can also be used to set a limit for the validity of the numerical modeling.

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| J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\2d_3d_compare_9deg_2.emf   1. Cp distribution | J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\2d_3d_compare_9degCfx_2.emf   1. Skin friction distribution |
| K:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\3D\9deg.png   1. Surface flow pattern on upper wing (left: tip; right: root) | J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\3D\slice_9deg.png   1. Streamline pattern at wing root |

Figure Pressure, skin friction and streamline patterns at α = 9°

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| J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\2d_3d_compare_10deg_2.emf   1. Cp distribution | J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\2d_3d_compare_10degCfx_2.emf   1. Skin friction distribution |
| K:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\3D\10deg.png   1. Surface flow pattern on up wing (left: tip; right: root) | |
| J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\3D\slice_10deg.pngJ:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\3D\slice_10degT.png   1. Streamline pattern at wing root (trailing edge separation increases again) | |

Figure Pressure, skin friction and streamline patterns at α = 10°

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| J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\2d_3d_compare_11deg_2.emf   1. Cp distribution | J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\2d_3d_compare_11degCfx_2.emf   1. Skin friction distribution |
| K:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\3D\11deg.png   1. Surface flow pattern on upper wing (left: tip; right: root) | J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\3D\slice_11deg.png   1. Streamline at wing root |
| K:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\3D\3d_CP_2_SS_11deg3.eps   1. 3D streamlines (bottom left: tip; top right: root) | |

Figure Pressure, skin friction and streamline pattern at α = 11°

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| J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\2d_3d_compare_12deg_2.emf   1. Cp distribution | J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\2d_3d_compare_12degCfx_2.emf   1. Skin friction distribution |
| K:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\3D\12deg.png   1. Surface flow pattern on upper wing | J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\3D\slice_12deg.png   1. Streamline at wing root |
| K:\Jason_chen\PHD2\PHD_year2\MAV\validation\case2\model\2nd_time\fluent-results\3D\Re6_3d_CP_2_SS_12deg.eps   1. 3D flow structure (bottom left: tip; top right: root) | |

Figure Pressure, skin friction and streamline pattern at α = 12°

### Summary

A numerical investigation of the three dimensional laminar separation bubble development and its spanwise variation with wingtip effect on a cambered thin wing at a Reynolds numbers of 60,000 has been carried out. LSB in both 2D and 3D are studied and discussed. The comparison between prediction and experiment indicates good modeling accuracy before stall conditions. Near and beyond stall with merged LSB with trailing edge separation, the reliability of the modeling becomes questionable.

By comparing results at the 3D symmetry plane with those from 2D, a delayed LSB formation is found on the 3D wing. This 3D LSB expands chordwise downstream and spanwise outboard gradually with increasing incidence. The bubble is tapered at the outboard end and eventually merging with the leading-edge-tip corner vortex. The strongest influence of the LSB is obtained at moderate angles of attack, which occurs later for the 3D wing. The sudden jump in the lift curve slope is shown to be associated with the expansion of the LSB for both the 2D and 3D cases, at different incidences. The 3D effects vary with incidences and the strongest 3D effects are found at either a moderate incidence at 6° or high incidences beyond 10°.

Trailing edge recirculation dominates on the upper wing surface at incidences between 0° and 4° for this particular Reynolds number. Both 3D and 2D cases have a LSB formed on the lower wing surface close to the leading edge at 0°. Both cases showed that trailing edge recirculation region expands and the separation point moved towards the leading edge with increasing incidence. A clear wingtip effect has been observed on the suction surface for the 3D case.

Between incidences 5° and 10°, both leading edge LSB and trailing edge recirculation present on the suction surface at the same time. The bubble length varies as the incidence changes. A short LSB with approximately 5% chord forms near the leading edge area at incidence of 5°, and its spanwise length is about 30% chord. However, the bubble did not affect the lifting surface much. As the incidence increased to 6°, the LSB increased its chord length up to 23%c, and the corresponding increase in lift slope CLα for both experimental data and numerical results is shown. This indicated that the LSB starts functioning significantly at this particular incidence. In general, the LSB extends in both chordwise and spanwise, as the angles of attack increases. The separation location moves towards the leading edge, and the reattachment points to the trailing edge as incidence increases. Nevertheless, the transition location varies in a very small range, roughly located between 10% and 12% for all the moderate angles of attack, except at incidence of 6° for which the transition point shifts to 18.7% of the chord. Generally, the pressure plateau peak occurs at the junction flow area and this junction points moved towards the wingtip as the incidence increases. The surface flow pattern shows the strongest 3D effects at 6° with the focus point near half way between the wing root and the wingtip. At 7° and 8°, the LSB is shown to be relatively more uniform along the span.

At 9° and 10°, the leading edge bubble and trailing edge recirculation extend and merge to form a large bubble on the upper wing surface progressively from inboard as incidence increases. As incidence increases further, the large 3D reversed flow occurs on the upper wing surface at the root and a strong leading-edge-tip corner induced 3D vortex flow dominates in the wingtip area. This tip vortex occupies much of the outer wing and feeds into the inboard recirculation. The pressure plateau has similar value along the wing span for the incidence of 11°, and the transition location is almost fixed at a value of 13%c. At these large incidences, the modeling is more questionable due the large discrepancies in lift between the prediction and the experiment.

Generally, due to the strong three dimensional LSB effects shown for the rectangular wing with a moderate aspect ratio (AR=6) at low Reynolds numbers, direct application of two dimensional profiles to MAV design has be to carried out with much caution.

# Wing Planform and Fuselage Aerodynamics

Key geometric parameters in micro air vehicle design include aerofoil section (thickness, cambers, etc.) and wing planform (aspect ratio, taper ratio, etc.) as mentioned by Pines and Bohorquez [23]. Along with the effects of these geometric parameters, operating at a low Reynolds number range with strong viscous effects, flow separation and reattachment, vortices, boundary layer transition, can all significantly affect the wing aerodynamics [13-14].

Due to the low Reynolds number effects and practical issues, thin aerofoil profiles are preferred for MAV designs. Pelletier [8] studied different thin/cambered aerofoil for a rectangular wing planform with Reranging from 60,000 to 200,000. The wind tunnel results showed that 4% cambered wing (without reflex camber) offers better aerodynamic characteristics than flat-plate wing for given Reynolds number and aspect ratio. The flow on the cambered wing remained attached for a higher incidence than that for the flat plate wing. These thin plate wings do not experience abrupt stall and the lift force often reached a plateau and then remained relatively constant. However, there were no particular reasons provided to explain this phenomenon. Swanson et al. [77] studied planform and camber effect on the low Reynolds number aerodynamics computationally. He showed that the tip vortex is the dominant flow and it forms the highly-three-dimensional low-velocity region at high incidences. Tezuka et al. [85] also investigated the surface flow structure on a 4% cambered plate at Reynolds number of 93,000. The laminar separation bubble was clearly showed by the oil flow technique on the upper wing surface and the pressure distribution with a plateau followed the suction peak near the leading edge is also appeared.

The reflex camber is usually designed for MAVs to perform in a longitudinal stable flight as a flying wing. The combined positive and reflex camber effects on MAV aerodynamic characteristics were studied by Reid *et al.*[92] and Null *et al.*[4] individually. Reid *et al.* focused on a rectangular wing (AR=2) planform with a positive camber varies from 1%c to 9%c located at 25%c, and a fixed reflex camber of 1%c located at 85%c. The surface oil flow visualization was taken in the investigation. The laminar separation bubble was found to be a key phenomenon, depending on the camber, Reynolds number, and incidence. At a given incidence and Reynolds number, the separation point moves downstream as camber increases, and the flow fails to reattach after separation as the camber is above 6%. The force measurement showed that the (CL/CD) max depends strongly on both the positive camber at 25%c and the Reynolds number. Unfortunately, there were no further studies on influences of the reflex camber and its location. Null *et al.*[4] studied a circular wing planform with various cambers (both positive and reflex cambers) at Re of 50,000, 75,000, and 100,000, respectively. The positive cambers varied from 3%c to 12%c. For the high speed flight ( Re = 100,000), the 3% camber gave the best (CL/CD) max value of 6.5, and for the low speed, 6% and 9% camber offer better (CL/CD) max values.

On the effect of wing planforms at low Re, however, most the work focused on the flat plate geometries without camber. Torres [9] investigated flat plate wing planforms with different aspect ratios at low Reynolds numbers. The results indicated that, for AR > 1.5, the elliptical planform is more aerodynamically efficient than the other planforms studied. Beside this, as the aspect ratio increased, the stall angle occurred earlier and a relatively constant lift appeared after stall angle as the incidence increased further, but the lift slope C*Lα* becomes more linear than lower aspect ratio wings. More recently, Okamoto [93] studied a series of flat plate wing planforms at an even lower Reynolds number of 10,000. A strong planform effects was found for low-aspect-ratio MAV wings. Zuo [94] also investigated experimentally flat plate wing planforms with different leading edge sweep angles. The wing has a low aspect ratio at a relatively higher Reynolds number of 273,000. From surface oil flow visualization and force measurements, it was found that for sweep angle Λ<35°, CL drops abruptly after stall but CL decreases more gradually after stall as the sweep angle increases beyond 50°. The large vortex flow on the upper wing surface varied with the incidence and the detachment of the vortices was associated with the abrupt drop of CL. The sweep angle was found to alter the stall behaviour significantly. Zhang [95] performed similar investigation on sweep angle/taper ratio effects at a Reynolds number of 342,000. 3D leading edge separation bubbles were observed at low incidences, varying with the sweep angles. He confirmed the sweep effect on the wing stall type and pointed out that the reason may be due to the different leading edge vortex structures. Elliptical wing planform was also studied by Jian [26] by both numerical and experimental. He pointed out that at incidence larger than 11 degree the flow becomes bilateral asymmetric. It is mainly due to the tip vortices’ destabilization. As the angle of attack increased to 33degree, a large separated vortex stays above the wing, and forming a stationary vortex. Delta wings at low Reynolds number also studied by several investigators, such as Ol, et al. [96], and Wang [97-98], etc (i.e. investigated on the leading edge vortex structure at low Reynolds number).

From the literature, as far as the MAV planform study is concerned, investigation on planform effects has so far been limited to flat plate wings. In the present work, various MAV planforms are investigated for thin cambered wings, with both positive and reflex cambers, at a low Reynolds number. The study is divided into three parts: (1) a validation case is performed and results are compared with the experiment data, and this can be found in 4.1.1 and 4.1.2; (2) the aerodynamic performance of camber plate wings of different planforms is simulated, analyzed, and the results are compared; and 3D flow structures at both design and stall conditions are investigated and discussed, can be found in 4.3; (3) The wing-fuselage aerodynamic interaction are discussed in 4.4.

### Validation cases

In this section, two validation cases are introduced, a) flat plat Zimmerman wing planform, and b) circular cambered thin wing planform. The CFD results are compared with the wind tunnel results and the results are analyzed and discussed.

#### Flat plate Zimmerman-wing-planform

The flat plate Zimmerman wing planform was investigated by Torres and Mueller experimentally [9]. It has a zero camber (i.e. flat plate) and the aspect ratio is two (AR = 2). The aerodynamic mean chord is 0.1725m which gives a corresponding Reynolds number of 100,000. The model was constructed from aluminium, and it has a thickness-to-chord ratio of 1.96% and 5-to-1 elliptical leading and trailing edges. It is noticeable that the aerodynamic centre (i.e. ) is the distance between the leading edge and the 25% point of the mean aerodynamic chord, the wing planform is shown in Figure 33.

To minimise the boundary condition effects, two different types of boundary settings are tested and the results are compared with the experiment data. The models are listed in and named as: model 1, model 2 and model 3. Two different types of mesh topologies (C-H type and O-type mesh) are used to investigate the grid effects, shown in . Model 1, in (a), has a C-H type mesh topology and the relevant boundary conditions are: velocity-inlet, pressure-outlet and a non-slip wall is applied on the wing. Model 2, however, has used an O-grid type mesh, which only velocity-inlet boundary condition is applied on the entire outer domain and the same boundary condition (non-slip wall) is applied for the wing. Wing tunnel boundaries are applied for Model 3. Therefore non-slip wall boundary condition is used. For the rest boundaries (inlet, outlet, and the wing), they are applied with the same boundary conditions as those have applied on Model 1 and 2. To minimize the farfield boundary condition effects, Model 1 is set 25c, and 35c for both up and downstream, respectively. Both the upper and lower boundaries are set 25c away from the wing surface . Model 2, however, has a spherical outer domain which is 35c away from the leading edge. The wind tunnel domain, for Model 3, has a square cross-section area of 2 ft×2 ft (610mm×610mm) and the test section has a length of 1820mm, see Ref. [9] for the rest details. All models are simulated only half geometry and therefore a symmetrical boundary condition is used.

Due to the computational demand, the mesh-sensitivity studies are performed for the three-dimensional case for α = 4°. Mesh topologies have mentioned before and showed in . Grid points were concentrated near the wing planform edge in order to capture the vortical flow. Therefore, the description of wake structure is limited to the region of the downstream, and the study of farfield wake is not included in this investigation. The mesh sensitivity results are listed in . The baseline mesh has 330 grid points around the aerofoil, 95 points in the normal direction and 160 points in the spanwise direction, (). Close to the wall, there are about 60 grid points within the boundary layer, and in the turbulent region. The y+ value of the first cell distance is ensured to be in order of 1. The stretching ratio for the mesh is less equal than 1.2. Based on the mesh sensitivity studies summarized in . The wall normal direction resolution of 95 seems to be sufficiently good. The boundary conditions study, in , has shown that the Model 3 (wind tunnel settings) gives a better lift coefficient as compared with the other data. However, Model 1 and Model 2 have quite reasonable results at lower incidences and under-predicted values are obtained at high incidences. On the drag value side, values are quite close to the experiment data, and under-predicted are found at high incidences. This may due to several possible reasons, such as effects from the model surface roughness, transition locations (or free-stream turbulent intensity level), data acquisition methods (i.e. especially at high incidences) and also wind tunnel wall correction, which can be fluctuated up to 10% as mentioned by Mueller [9]. Mesh 1 (with wind tunnel settings), the grid with 330×95×160 points, is chosen for all computations reported here, as it gives sufficiently accurate results in the lift, drag coefficients in comparison with the finest mesh.

|  |  |
| --- | --- |
| J:\Jason_chen\PHD2\PHD_year2\MAV\validation\PHD3\case3\AR2_zime\ICEM\C_H_grid.jpg   1. C-H-type mesh | J:\Jason_chen\PHD2\PHD_year2\MAV\validation\PHD3\case3\AR2_zime\ICEM\O_grid.jpg   1. O-type mesh |

Figure Three-dimensional wing planform with computational mesh topologies (baseline grid)

Table Grid-sensitivity analysis for α = 4°

|  |  |  |  |  |  |
| --- | --- | --- | --- | --- | --- |
|  | Case | Grid Size | | CL | CD |
| Model 1  (C-H-type mesh) | 1 | 330×95×160 | | 0.1756 | 0.02042 |
| 2 | 400×100×180 | | 0.1744 | 0.01913 |
| 3 | 470×110×200 | | 0.1775 | 0.02068 |
|  |  |  | |  |  |
| Model 2  (O-type mesh) | 1 | 330×95×160 | | 0.1719 | 0.01821 |
| 2 | 400×100×180 | | 0.1721 | 0.01761 |
| 3 | 470×110×200 | | 0.1822 | 0.02123 |
|  |  |  | |  |  |
| Model 3  (C-H-type mesh) | 1 | 330×95×160 | | 0.1887 | 0.01756 |
| 2 | 400×100×180 | | 0.1899 | 0.01890 |
| 3 | 470×110×200 | | 0.1900 | 0.01877 |
| Experiment [9] |  |  | | 0.1906 ± 0.02 | 0.0220± 0.003 |
| G:\JasonChen\PhD_work\validation\PHD3\case3\matlab\post_process\zim_validation_CL.emf   1. CL | | | G:\JasonChen\PhD_work\validation\PHD3\case3\matlab\post_process\Zim_validation_CD.emf   1. CD | | |

Figure Aerodynamic coefficients

#### Cambered circular-wing-planform

The following section is divided into two subsections: (a) geometry description; and (b) computational setup, including both mesh topologies and boundary conditions. shows the geometry for the validation case and the MAV model has a circular wing planform. This wing was investigated by Null [4] experimentally. The flow conditions are: freestream velocity of 10m/s, the aerodynamic chord of 0.206mm is used, and this gives a Reynolds number of 100,000. The given experiment uncertainties are ±0.07N for lift, ±0.06N for drag, and ±0.073Nm for the pitching moment. The aerofoil has a positive camber of 6%c and a reflex camber of 2.3%c, located at 24%c and 87%c respectively. The model has a wing area of 0.0387m2 with a mean chord length of 206mm, and a thickness of 0.508mm. All the parameters are listed in .

Table Validation cases parameters

|  |  |
| --- | --- |
| Validation Case- Circular wing MAV | |
| Wing area, S (m2) | 0.0387 |
| Chord length, c (m) | 0.206 |
| Positive Camber, (m) | 0.0124 |
| Reflex Camber, (m) | 4.75e-3 |
| Wing Thickness, t, (m) | 5.08e-4 |
| Positive camber location, (m) | 0.0495 |
| Reflex camber location, (m) | 0.1794 |
| Wing span, b, (m) | 0.2286 |
|  |  |

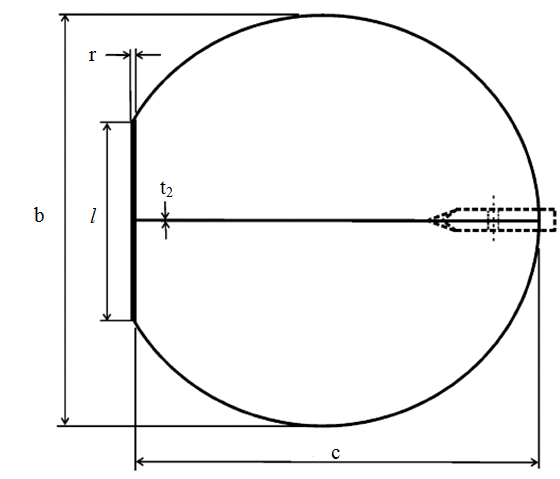


Figure MAV with circular wing planform [4]

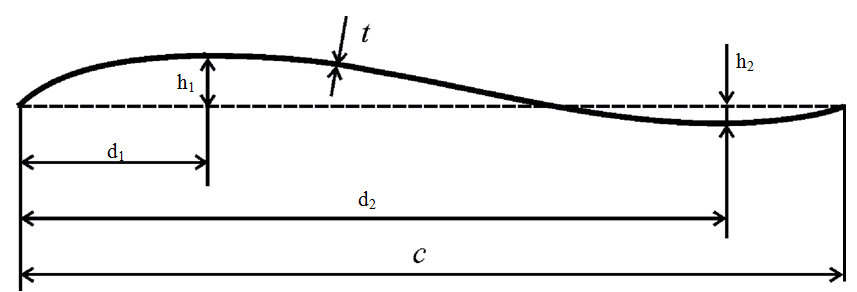


Figure Aerofoil used this study

After the validation, different planforms with a thickness of 0.9% are investigated. The camber line shown in [4] indicates the parameters in the camber line design. Based on previous studies [4, 77, 85, 92], a positive camber of 5.8% located at 25% of the mean chord and a reflex camber of 1% located 86% are chosen for all the following planform studies. The reflex camber is introduced in the design for longitudinal stability for these flying wing configurations without the tail plane although it also reduces the overall aerodynamic performance. The flow conditions are: freestream of 10m/s, the mean aerodynamic chord of 0.2214m is designed, and this is giving a Reynolds number of 150,000.

In the simulations, fully structured meshes are generated by the ICEM mesh generation program, as shown in Figure 37. All models have a circular leading edge and a sharp trailing edge. A structured C-H topology is chosen for the calculations. The freestream Reynolds number based on the chord length is set to 100,000. A symmetric boundary condition is applied at the wing root plane. The no-slip wall boundary condition is enforced on the wing surface and the farfield condition is set for the farfield boundary. To minimize the effect of the far-field boundary on accuracy of the solutions near the geometry, the outer domains were placed at 12c in the upstream, upward, and downward directions, 15c for the downstream, and 10c in the span direction.

Table 6 lists the results based on the grid sensitivity analysis for the circular wing, at two incidences of 0° and 10°. From this mesh sensitivity study, Mesh 2, the grid with 402×70×75 points, is chosen for all computations reported later, as it gives reasonably accurate results in the force coefficients. shows the aerodynamic coefficients versus the angle of attack, showing good agreement with the experiment data below 25° and under predictions are found in the stall region. For the resolution of the turbulent boundary layer, the y+ value needs to be in the order of O(1), which are satisfied as shown in .

|  |  |
| --- | --- |
| J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case1\mesh independent study\circular_Leading_edge3\mesh1.png | J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case1\mesh independent study\circular_Leading_edge3\mesh3.png |
| J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case1\mesh independent study\circular_Leading_edge3\sym_mesh.png | J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case1\mesh independent study\circular_Leading_edge3\sym_mesh1.png |

Figure 37 Mesh topology for validation case

Table Grid-sensitivity analysis for α = 0° and 10°

|  |  |  |  |  |
| --- | --- | --- | --- | --- |
| Case (size) | α (°) | Grid size | CL | CD |
| 1 (2million) | 0 | 276×55×58 | 0.108 | 0.047 |
| 10 | 276×55×58 | 0.585 | 0.119 |
| 2 (4 million) | 0 | 402×70×75 | 0.106 | 0.047 |
| 10 | 402×70×75 | 0.604 | 0.125 |
| 3 (6 million) | 0 | 478×90×86 | 0.109 | 0.052 |
| 10 | 478×90×86 | 0.599 | 0.125 |
|  |  |  | **CL** | **CD** |
| Experiment | 0 |  | 0.097±0.12 | 0.036±0.05 |
|  | 10 |  | 0.639±0.12 | 0.129±0.05 |

|  |  |
| --- | --- |
| J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case1\CL_3D_color.emf   1. CL | J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case1\CD_3D_color.emf   1. CD |

Figure Aerodynamic coefficients for mesh set 402×70×75

J:\Jason_chen\PHD2\PHD_year2\MAV\validation\case1\Yplus10.emf

Figure 39 y+ value

### Geometry description and computational mesh

The different planforms with a thickness of 0.9% are investigated and results were compared. The camber line shown in Figure 36 [4] indicates the parameters in the camber line design. Based on previous studies [4, 77, 85, 92], a positive camber of 5.8% located at 25% of the mean chord and a reflex camber of 1% located 86% are chosen for all the following planform studies. The reflex camber is introduced in the design for longitudinal stability for these flying wing configurations without the tail plane although it also reduces the overall aerodynamic performance. The flow conditions are: freestream of 10m/s, the mean aerodynamic chord of 0.2214m is designed, and this is giving a Reynolds number of 150,000. The sketch of the thin aerofoil can be seen in Figure 36 and specifications are listed in .

The aerofoil, as mention before in the validation case 1-Figure 36, is a simple cambered aerofoil and specification can be found in . The positive camber is designed to have a better aerodynamic performance, whereas, the reflex camber is designed to maintain stable level flight, resulting in decreased flight times. Not only the static stability but also the dynamic stabilities (for control handling) are important for the design. Torres and Mueller [9] showed that the lift centre location is calculated from the normal force and pitching moment taken from the /4 location of each wing. The location of lift centre is very close to /4 point at low angles of attack. However, it shifts towards the trailing edge as the incidence increases. The reason behind this is due to the effect from both the increasing trailing edge separation and the strengthening wing tip vortices. In this investigation, the mean aerodynamic chord,, as defined in Eq. 78 is used and Figure 41 (a) shows the location of the mean aerodynamic chord.

|  |  |
| --- | --- |
|  | Eq. |

Table 7 Thin plate aerofoil camber parameters

|  |  |  |  |  |  |  |
| --- | --- | --- | --- | --- | --- | --- |
|  | d1/ | d2/ | h1/ | h2/ | t/ |  |
|  | 25% | 86% | 5.8% | 1% | 0.9% | 0.2214m |

To study the cambered thin plate wings, four distinct geometries are selected: rectangular, tapered wings with swept leading edges, Zimmerman, inversed Zimmerman, as shown in Figure 40 and Figure 41, respectively. The choices of these planforms are based on some currently often used MAV planform designs. For example, the Wasp MAV by AeroVironment [99] and UGMAV from University of Ghent [100] took a low aspect ratio tapered wing planform design. The MITE MAVs from the Naval Research Labarotory [101] and Hornet MAVs of AeroVironment [1] were based on a simple low aspect ratio rectangular design. Tapered wings with high leading edge sweep were adopted for Zagi MAV [1] and Zimmerman planforms were used in the Dragonfly MAV by University of Arizona [102], Florida MAVs [103], and the thrust-vectored MAV by the University of Sheffield [104-105]. No detailed comparison of the aerodynamic performance at low Reynolds numbers for these different planforms can be found in the literature, giving the reason for the present study. Figure 40 (f) shows an H-type mesh topology for current wing planforms.

shows the specifications of the different MAV planforms, all having the same wing area of 0.0895m2, the same aspect ratio of 2.12 and the mean aerodynamic chord of 0.221 m. The tapered wings also have the same taper ratio of 0.44. The root chords are 0.25m for all wings except for the rectangular wing.

Table wing planform specifications

CL, required is the required lift coefficient for the design condition

|  |  |  |  |  |
| --- | --- | --- | --- | --- |
| Design condition: CL, required = 0.35 | | | | |
| Wing planform | Cr (m) | Ct (m) | Λ (°) | b (m) |
| Rectangular | 0.221 | 0.221 | 0° | 0.4 |
| LE Sweep 0° | 0.250 | 0.193 | 0° | 0.4 |
| LE Sweep 9° | 0.250 | 0.193 | 9° | 0.4 |
| LE Sweep 18° | 0.250 | 0.193 | 18° | 0.4 |
| LE Sweep 30° | 0.250 | 0.193 | 30° | 0.4 |
| In-Zimmerman | 0.250 |  |  | 0.44 |
| Zimmerman | 0.250 |  |  | 0.44 |

|  |  |  |
| --- | --- | --- |
| K:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\without_fuselage\Fluent_No_fuselage\MAV1\fluent\new\MAV1_geometry.jpg   1. Rectangular | J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\without_fuselage\Icem\mav2_sweep1_no_fuse\sweep1.jpg   1. 0° leading edge sweep | J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\without_fuselage\Icem\mav2_sweep3_no_fuse\sweep3.jpg   1. 9° leading edge sweep |
| J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\without_fuselage\Icem\mav2_sweep2_no_fuse\sweep2.jpg   1. 18° leading edge sweep | J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\without_fuselage\Fluent_No_fuselage\MAV2\sweep4_nofuse\Sweep4_geometry.jpg   1. 30° leading edge sweep | J:\jasonchen\PhD_work\MAV\MAV1_MAV6\without_fuselage\3ddp_results\rectangular_mesh.jpg   1. H-type mesh |

Figure Trapizoidal planform geometries

|  |  |
| --- | --- |
| J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\without_fuselage\Icem\mav3_invzim_no_fuse\inv_zim.jpg  U∞   1. Inverse-Zimmerman | J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\without_fuselage\Icem\mav4_zim_no_fuse\zim.jpg   1. Zimmerman |

Figure Zimmerman and Inversed-Zimmerman planform geometry

### Wing planform effects on aerodynamic performance

The design condition for testing the various planforms is based on the requirement for our flying MAV prototype, as shown in Figure 3 with a thrust vector propeller cruising at 20 m/s and a Zimmerman wing planform. This MAV has been flight-tested with the fuselage, the propeller in a tractor configuration and a vertical stabilizer. Note, in the current paper, only the wing itself is studied without the influence of the other components. shows the aerodynamic efficiencies of the various planforms defined in the last section at the design lift condition of CL=0.35. The solution was run iteratively to satisfy the design lift condition and in the process, the required incidences for the different wings were found. As can be observed from the table, the design lift is achieved at relative small incidences for this particular positive and reflex camber design.

Table 9 Aerodynamic coefficients at design conditions

|  |  |  |  |  |  |
| --- | --- | --- | --- | --- | --- |
|  | CD, dc | CM, dc | CL/CD, dc | α°dc | Lift slope: |
| Rectangular | 0.0578 | 0.0682 | 6.052 | 1.43 | 0.0531 |
| Tapered 0° LE | 0.0567 | 0.0671 | 6.172 | 1.57 | 0.0531 |
| Tapered 9° LE | 0.0571 | 0.0669 | 6.134 | 1.59 | 0.0550 |
| Tapered 18° LE | 0.0568 | 0.0687 | 6.165 | 1.45 | 0.0535 |
| Tapered 30°LE | 0.0504 | 0.0652 | 6.944 | 1.69 | 0.0567 |
| Inv-Zimmerman | 0.0626 | 0.0725 | 5.869 | 0.45 | 0.0573 |
| Zimmerman | 0.0468 | 0.0439 | 7.475 | -1.03 | 0.0567 |

All the tapered wings, with the same taper ratio of 0.44, have better aerodynamic performance than the rectangular wing. Among them, the one with the largest leading edge sweep produces a significantly better lift-drag ratio. The inversed Zimmerman wing is the poorest among the candidates while the Zimmerman wing gives the best aerodynamic performance at the design condition. A series of angles of attack from -6° to 27° are simulated at every three degrees.

#### Tapered wing with different leading edge sweep

The aerodynamic performance plots of trapezoidal wings in -(a) shows that all wing planforms have the similar lift slope CLα in the near linear parts of the curves at low incidences, despite the differences of the aerodynamic performance are shown in Table 5. A strong effect of leading edge sweep is shown in -(b). The 18° and 30° sweep wings have the highest CL,max of 0.97-0.98 with a significant sudden drop of lift at stall. As the leading edge sweep reduces, the stall behaviour becomes more gradual with decreased CLmax .

The sweep angle also has a noticeable influence on the pitching moment coefficient. The trapezoidal wings have positive Cmα at incidences between -9° to 5°, implying longitudinal instability if the wings are used alone at low incidences. The 2D spanwise lift distribution, C*l* in , indicates that the tapered wings have a more constant C*l* distribution along the wing span than the rectangular wing. They also show higher loading near the wing tips.

|  |  |
| --- | --- |
| K:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV1_2_CL.emf   1. CL | 1. CL (zoom in) |
| J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV1_2_CD.emf   1. CD | J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV1_2_CM.emf   1. CM |

Figure Aerodynamic performances for swept wing planforms

J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\without_fuselage\Fluent_No_fuselage\MAV2\2d_Cl_MAV1_2_035.emf

Figure 43 Spanwise C*l* distribution at design condition

shows the surface streamlines for the trapezoidal wings at the design condition. Fully attached flow is observed on all the upper wing surfaces, and leading edge separation bubbles form on the lower surfaces for all shapes. All the trapezoidal wings show the wing tip effects on the surface streamline curvature. The spanwise flow strengthens as the leading edge sweep increases. On the lower wing surface, variation of the leading edge separation bubble is also observed, depending on the leading edge sweep. There is a large difference in the LSBs for the tapered wing with zero sweep and the rectangular wing.

The spanwise Cp distributions at different chordwise locations are shown in . The suction peaks are most prominent in the plots for the 25%c location, indicating the wing tip vortex effect. The peak Cp value is similar to the Zimmerman wing case shown later.

|  |  |  |
| --- | --- | --- |
| 1. Upper surface | 1. Lower surface | 1. 3D view |
| K:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\without_fuselage\Fluent_No_fuselage\MAV1\fluent\new\MAV1_DC_up.jpg  Rectangular | J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\without_fuselage\Fluent_No_fuselage\MAV1_035_low.jpg | J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\without_fuselage\Fluent_No_fuselage\MAV1\fluent\new\MAV1_DC_tip.jpg |
| K:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\without_fuselage\Fluent_No_fuselage\sweep1_035_up.jpg  Tapered 0° LE | K:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\without_fuselage\Fluent_No_fuselage\sweep1_035_low.jpg | K:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\without_fuselage\Fluent_No_fuselage\MAV2\sweep1_nofuse\fluent\Sweep1_DC_2.jpg |
| K:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\without_fuselage\Fluent_No_fuselage\sweep3_035_up.jpg  Tapered 9° LE | K:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\without_fuselage\Fluent_No_fuselage\sweep3_035_low.jpg | J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\without_fuselage\Fluent_No_fuselage\MAV2\sweep3_nofuse\fluent\Sweep3_DC_tip.jpg |
| K:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\without_fuselage\Fluent_No_fuselage\sweep2_035_up.jpg  Tapered 18° LE | K:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\without_fuselage\Fluent_No_fuselage\sweep2_035_low.jpg | K:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\without_fuselage\Fluent_No_fuselage\MAV2\sweep2_nofuse\fluent\sweep2_upper_DC2.jpg |
| J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\without_fuselage\Fluent_No_fuselage\MAV2\sweep4_nofuse\Sweep4_DC_UP.jpg  Tapered 30° LE | J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\without_fuselage\Fluent_No_fuselage\MAV2\sweep4_nofuse\Sweep4_DC_down.jpg | J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\without_fuselage\Fluent_No_fuselage\MAV2\sweep4_nofuse\Sweep4_DC_tip.jpg |

Figure Stream patterns at design condition, flow from left

|  |  |
| --- | --- |
| **Design Condition: wing planforms** | |
| **Rectangular: (α = -0.72°)** | **Swept-0°: (α = 1.57°)** |
| K:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\without_fuselage\Fluent_No_fuselage\MAV2\035_Spanwise_MAV1_Cp_DC.emf   1. Cp distribution | K:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\without_fuselage\Fluent_No_fuselage\MAV2\035_Spanwise_Sweep1_Cp_DC.emf   1. Cp distribution |
|  |  |
| **Swept-9°: (α = 1.59°)** | **Swept-18°: (α = 1.45°)** |
| K:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\without_fuselage\Fluent_No_fuselage\MAV2\035_Spanwise_Sweep2_Cp_DC.emf   1. Cp distribution | K:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\without_fuselage\Fluent_No_fuselage\MAV2\035_Spanwise_Sweep3_Cp_DC.emf   1. Cp distribution |

Figure 45 Spanwise Cp distribution at different chordwise locations at design condition

#### Zimmerman versus Inversed-Zimmerman

URANS simulations are performed for both Zimmerman and inversed-Zimmerman wing planforms. The simulation was performed with various incidences from -6° to 27°, and the time step was set to be 4.428×10-4 seconds for both wing planform modeling. The time averaging for URANS was made based on the results of 300 time steps after the solution was converged, which corresponded to 6 units of flow through time (t = c/U∞). illustrates the aerodynamic forces versus the number of time steps. The forces typically begin with quite high fluctuations and then settle into a constant value as the time stepincrease further.

L:\jasonChen\PHD_year3\MAV4_zim\MAV4_propeller\MAV4_prop_x\FLUENT\unsteady_MAV3_MAV4\isolated_wing\MAV4\10ms\MAV4_residural.emf

Figure Aerodynamic forces development

(a) shows the lift coefficient CL of the Zimmerman and inverse-Zimmerman wings. The lift slops are near constant for angles of attack α ≤ 9° and CL curves become nonlinear asα ≥ 9°. In general, the Zimmerman wing has higher lift than the inversed-Zimmerman wing at all incidences before stall. Both wing planforms have the same stall angle of 12°. The upward shift of the lift curve for the Zimmerman wing is due to its higher effective positive camber from the larger local span at around 25%c. (b) shows the drag coefficient CD for both wings and the Zimmerman wing shows lower values at moderate incidences. As a result, a much better aerodynamic performance is achieved with the Zimmerman wing as shown in .

For a thin aerofoil, the aerodynamic centre is at the quarter chord location. The CMac plot around quarter mean aerodynamic chord has shown in Figure 47 (c) Here the Zimmerman and inversed Zimmerman wings show an opposite sign in the Cm slope around the design condition. For the Zimmerman wing at design, the Cm slope is positive, indicating longitudinal instability. On the other hand, the inversed Zimmerman wing has a negative Cm slope at design, showing longitudinal stability.

shows the lift distribution along the span at the design condition, the Zimmerman wing has a lower C*l* distribution at z/b < 0.6 and the lift increased dramatically near the wingtip location. The inversed-Zimmerman wing, on the other hand, has the lift C*l* dropped at the tip suddenly. The reason for the lower lift at the wingtip is because the inversed-Zimmerman wing has a maximum span at its maximum reflex camber position (i.e. maximum span at x/c = 0.85). The corresponding surface flow pattern in clearly showed the flow structure for both wing planforms.

|  |  |
| --- | --- |
| L:\jasonChen\PHD_year3\MAV4_zim\MAV4_propeller\MAV4_prop_x\FLUENT\unsteady_MAV3_MAV4\isolated_wing\MAV4\10ms\unsteady_MAV3_4_CL.emf   1. CL | L:\jasonChen\PHD_year3\MAV4_zim\MAV4_propeller\MAV4_prop_x\FLUENT\unsteady_MAV3_MAV4\isolated_wing\MAV4\10ms\unsteady_MAV3_4_CD.emf   1. CD |

L:\jasonChen\PHD_year3\MAV4_zim\MAV4_propeller\MAV4_prop_x\FLUENT\unsteady_MAV3_MAV4\isolated_wing\MAV4\10ms\unsteady_MAV3_4_CM.emf

c) CM

Figure Aerodynamic coefficients for inversed- and Zimmerman wing planforms

K:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\without_fuselage\Fluent_No_fuselage\MAV2\2d_Cl_MAV3_4_4p5deg_035_2.emf

Figure spanwise lift distribution at design condition

Surface flow visualization of the Zimmerman and inversed-Zimmerman wings at the design condition is given in . shows the spanwise pressure distribution at different chordwise locations. The stream pattern show that different flow structures form due to different wing planforms. The wing planform affected both wingtip vortex formation and separation bubble variations on both upper and lower wing surfaces.

At the design condition, both Zimmerman and inversed-Zimmerman wings in (a) show the upper wing separation occur at x/c = 0.25 (maximum positive camber location), and the reattachment is located at x/c = 0.85 (maximum reflex camber location). This contrasts with the upper surface flow structures for the trapezoidal wings discussed in the last section. The confined elliptical shaped separation bubbles on both upper wing surfaces has a similar length in its chordwise direction due to the camber variation, but shorter size has been found on the Zimmerman wing surface in its spanwise direction. In general, the area occupied by the separation bubbles on the inversed-Zimmerman wing is larger than that for the Zimmerman wing. This leads to the increased drag and therefore poorer aerodynamic performance. The leading edge vortex, on the other hand, has different structures due to the wing planform effect. On the inversed Zimmerman wing, the large leading-edge vortices induce strong lateral flow toward the edge and generate the secondary vortices, as shown on the surface stream patten in (a). Spanwise flow at the trailing edge has increased due to the reflex camber effect. Although the flow on the upper wing surfaces is separated over a significant area, the lift continues to increase with the incidence. This phenomenon can be attributed to the additional lift from wing tip vortices, which counteracts the negative effects of flow separation on lift. On the lower wing surfaces, separation bubbles form at both leading, trailing edges on the inversed-Zimmerman wing. The Zimmerman wing, however, has only the leading edge bubble on the lower wing surface at the design condition, as shown in (b).

To investigate the vortex lift on the wings, the spanwise pressure distributions at different chordwise locations are shown in . Three different streamwise locations are plotted at x/c = 0.25 (maximum positive camber location), x/c = 0.58 (zero camber location), and x/c = 0.85 (maximum reflex camber location). Note that the inversed-Zimmerman wing’s maximum span is located at x/c = 0.85, however, the Zimmerman wing’s at x/c = 0.25. (a) shows the pressure distribution from these three locations and (b) the corresponding Cp values. On the inversed Zimmerman wing, a pressure suction peak region between z/b = 0.65 and z/b = 0.78 is observed at ak location of x/c = 0.25 due to the addition lift generated from the strong leading edge vortex. A vortex suction region is also observable at x/c=0.58 but it disappears at x/c=0.85 as the leading edge vortex lifts off the surface. For the Zimmerman wing, the vortex lift (suction) is only observable at x/c=25%c to a significantly smaller scale. This can be better comprehended by combining with the wing tip vortex structures shown in . At the design condition of the same lift, the inverse Zimmerman wing benefits from the vortex lift but the Zimmerman wing more from the positive camber effect.

|  |  |
| --- | --- |
| 1. Upper surface | 1. Lower surface |
| K:\jasonChen\PHD_year3\MAV4_zim\MAV4_propeller\MAV4_prop_x\FLUENT\unsteady_MAV3_MAV4\isolated_wing\MAV3\mav3_invzim_no_fuse_unsteady_10ms-1-02200_Up.jpg  U∞  Inverse Zimmerman wing | K:\jasonChen\PHD_year3\MAV4_zim\MAV4_propeller\MAV4_prop_x\FLUENT\unsteady_MAV3_MAV4\isolated_wing\MAV3\mav3_invzim_no_fuse_unsteady_10ms-1-02200_Low.jpg |
| K:\jasonChen\PHD_year3\MAV4_zim\MAV4_propeller\MAV4_prop_x\FLUENT\unsteady_MAV3_MAV4\isolated_wing\MAV4\10ms\MAV4_zim_No_fuse_Unsteady_N0p72deg10-1-02240_up.jpg  Zimmerman wing | K:\jasonChen\PHD_year3\MAV4_zim\MAV4_propeller\MAV4_prop_x\FLUENT\unsteady_MAV3_MAV4\isolated_wing\MAV4\10ms\MAV4_zim_No_fuse_Unsteady_N0p72deg10-1-02240_low.jpg |

Figure Instantaneous stream pattern at design condition (CL = 0.35)

|  |  |
| --- | --- |
| **Design Condition: wing planforms** | |
| **Inverse-Zimmerman: (α = 0.76°)** | **Zimmerman: (α = -0.73°)** |
| K:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\Zim_camber_location.emf   1. Cp distribution locations | K:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\Zim_camber_location.emf   1. Cp distribution locations |
| K:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\without_fuselage\Fluent_No_fuselage\MAV2\035_Spanwise_MAV3_Cp_DC_unsteady.emf   1. inverse Zimmerman | K:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\without_fuselage\Fluent_No_fuselage\MAV2\035_Spanwise_MAV4_Cp_DC_unsteady.emf   1. Zimmerman |

Figure Mean spanwise Cp distribution at different chordwise location at design condition

### Wing-fuselage interaction

Figure 51 shows the geometry of the wing planforms with a fuselage, and the wing planform specifications are discussed in the previous section (). The wing planform effects were studied in the previous section, and the fuselage effects on the aerodynamics will be investigated in the following section. The fuselage has a total length hL = 0.216m, the fuselage front height hf = 0.06m, the rear height ht = 0.021m and the width is 0.048m Figure 52 (a). This fuselage is specially designed to meet the requirements, such as electronic components and payload. The dotted line in Figure (52) marked inside the fuselage represents the maximum available space for managing the electronic components; the volume is roughly 3.2×10-4m3.It is also necessary to provide enough internal spaces to allow the adjustment of centre of gravity at early stage for the flight tests. This is due to the small flying vehicles are highly sensitive to the longitudinal position of the centre of gravity. Albertani [29] pointed out that the range of flyable CG location is generally a few percent of the mean aerodynamic chord (in their case, the distance between the centre of gravity and aerodynamic centre is about 10mm). Stanford [103] also mentioned that the stability concerns are a primary target of design improvement from one generation of MAV to the next. It is true to say that as the size of MAV decreases the available of CG margin decreases which is also a strenuous weight management challenge to those small air vehicle designs. The endurance as another design point, for the battery-based propulsion system, the endurance is generally proportional to the battery size Figure 52 (b) by Mueller, et al [1]. The battery as a ‘fuel’ is that it is not consumed during the flight. Namely, the weight is fixed. In Figure 52 (b), it shows that keep the Wo constant (the aircraft weight without included the battery), and varying Wb (battery weight), it achieved the maximum endurance when the battery weight Wb = 2Wo. It is not surprised that the battery occupies most of the internal space. To achieve the maximum endurance, a battery nearly twice as wing’s weight has been selected and other components weight fractions are listed in Figure 53.

Fuselage size is associated with the overall available capacity. However, the fuselage shape is associated with both aerodynamic characteristics and the fabrication difficulties. A streamlined fuselage may have reasonable good aerodynamics but it would cost too much to build. The blunt fuselage, our design in Figure 51, is relative simple to build. At the early generation of the MAV designs, a blunt fuselage may have more advantages than a streamlined fuselage (such as low cost, fast fabricate, easy assemble with the wing, and straight forward for centre of gravity adjustment at flight test stage). Fuselage usually decreases the overall aerodynamic efficiencies. This is because the aerodynamic interaction occurred with the wing planform and separation usually takes place around the fuselage. MAV with a blunt fuselage has been studied by Brion [106] numerically. Based on their results, the fuselage reduced the lift slope, CLα, and the drag force increased dramatically. However, the fuselage showed a positive contribution on the statically longitudinal stability, where the negative pitching moments slope is shown at incidence between 0° and 21°. Authors such as Stults [107] and Null [1, 4] also used a blunt fuselage for their MAV individually. Although the streamlined fuselage has the difficulties on the manufactory, few authors such as Ramamurti [108], Stanford [109], Galinski [110], and the MAV “Sanders MicroSTAR” designed with a streamlined fuselage is also introduced by Lockheed MAV team [14].

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| --- | --- | --- | --- |
| J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV1_rect\MAV1_rect\Rec_WF.jpg   1. Rectangular | J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV2_sweep\iceberg_results\MAV2_sweep\Sweep1\sweep1.jpg   1. LE sweep-0° | | K:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV2_sweep\iceberg_results\MAV2_sweep\sweep3\Sweep3.jpg   1. LE sweep-9° |
| J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV2_sweep\iceberg_results\MAV2_sweep\Sweep2\sweep2.jpg   1. LE sweep 18° | J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV2_sweep\sweep4\MAV2_sweep4.jpg   1. LE sweep 30° | |  |
| J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV3_invzim\MAV3_invzim\invZim_WF.jpg   1. Inversed-Zimmerman | | J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV4_zim\MAV4_zim\Zim_WF.jpg   1. Zimmerman | |

Figure wing planform-fuselage geometries

|  |  |
| --- | --- |
| C:\Users\jason\Desktop\vector_plot\fuselage_specification.emf   1. fuselage specifications | C:\Users\jason\Desktop\publications\endurance.emf   1. battery weight versus endurance, [1] |

Figure fuselage specifications and battery weight

J:\Jason_chen\PHD2\PHD_year2\publications\Components2.emf

(i.e. Components from left to right: battery, receiver, ESC, motor, propeller, shaft, gear box, servo, carbon-fiber-rod, MAV, and motor amount)

Figure Components weight fraction for MAV

**Wing-fuselage Aerodynamics**

URANS with similar settings as used for wing planforms was applied to wing-fuselage cases. shows the convergence history for rectangular wing-fuselage model. Aerodynamic forces are fluctuated at the early time steps and convergence was found after t > 0.8 (averaged values were calculated from t > 0.8, such as CL = 0.35014, CD = 0.10229, and CM = 0.00328).

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Figure Convergence history for rectangular wing-fuselage model at CL = 0.35

Table 10 shows the aerodynamic performance for wing-fuselage cases at design condition CL=0.35. It is clearly shown in Table 10 that the design lift is achieved at slightly larger incidences when compare with the wing planform cases in .

Table wing-fuselage aerodynamics at design condition (CL = 0.35)

|  |  |  |  |  |  |  |
| --- | --- | --- | --- | --- | --- | --- |
|  | CD, DC | CM, DC | CL/CD, DC | α°DC | CD,fuselage | Lift slope: |
| Rectangular | 0.1011 | 0.0065 | 3.462 | 2.27 | 0.0433 | 0.0485 |
| Tapered 0° LE | 0.1078 | 0.0067 | 3.246 | 3.27 | 0.0511 | 0.0435 |
| Tapered 9° LE | 0.1028 | 0.0017 | 3.406 | 2.93 | 0.0457 | 0.0448 |
| Tapered 18° LE | 0.0950 | 0.01 | 3.683 | 2.45 | 0.0382 | 0.0489 |
| Tapered 30°LE | 0.0849 | -0.0415 | 4.121 | 2.42 | 0.0345 | 0.0497 |
| Inv-Zimmerman | 0.0991 | 0.0135 | 3.531 | 1.52 | 0.0365 | 0.0494 |
| Zimmerman | 0.0666 | 0.0368 | 5.256 | -0.14 | 0.0198 | 0.0485 |

With the fuselage, it seems to have a dramatic reduction on the overall aerodynamic performance for those wing-fuselage models (as compared with previous work in Ref. [111]). All the tapered wings, with the same taper ratio of 0.44, are shown a lower aerodynamic performance than the rectangular wing at low angles of attack (α < 3°). However, as incidence increases (α > 3°), the leading sweep 30° model shows a better lift coefficient than other models. The leading-edge-sweep 0° shows a highest drag coefficient on fuselage (about 47.4% of overall drag coefficient) while the Zimmerman wing gives the best aerodynamic performance at the design condition (the drag force from the fuselage is about 29.7%). A series of angles of attack from -9° to 27° are simulated at every three degrees

**Tapered wing-fuselage aerodynamics**

The coefficient of lift, drag, and pitching moment are given in Figure 55 (a-d) for both rectangular and trapezoid wing-fuselage models. Similar lift trends are obtained from those trapezoid wing-fuselage models. Figure 56 (a) clearly shows that a sudden drop of lift coefficient at post stall region is observed by LE-sweep 9°, 18° and 30°. However, CL with a gradually decrease at post stall region is shown by rectangular and LE-sweep 0° models (Figure 55 b). LE-sweep-30° shows a best maximum lift, CL,max , at α = 12°. In contrast, the wing-fuselage models shows a less linear CLα than the wing planform [112]. This may due to the significant aerodynamic interaction between the wing and the fuselage. To achieve the same amount of lift (CL = 0.35), higher incidences are required for the wing-fuselage models, (Table 10). Apparently, more separated area is found on the upper wing surface at the higher incidences for design condition as compared with wing planform (Figure 58 and Figure 44 for trapezoidal wing-fusealge and wing planform models, respectively). Higher the incidence, stronger aerodynamic interaction between the wing planform and the fuselage will be occurred. The 30° sweep wing has a maximum lift coefficient, CL,max, is about 0.67. While, both 9° and 18° leading edge sweep wings have shown a lowest CL,max which are 0.59 and 0.6 respectively. Harper [113] pointed out that the spanwise flow on the sweep wings acts as a natural boundary-layer control system and increases section maximum lift on the swept wings.

In addition, the increase of drag ( c), compared with the wing planforms, is in fact mainly due to the fuselage effect. At the design condition (inside the low incidence range), the total drag coefficient decreases as the leading edge sweep angle increases (), which did not show a pronounced similar trend in the previous wing planform study. As incidence increases, high drag values were found from both rectangular and LE-sweep-30° wing-fuselage model for α between 6° and 27°. However, LE-sweep-0°, and 9° show a decrease on drag coefficient as incidence increased. LE-sweep 18°, on the other hand, shows a very similar drag coefficient trend with LE-sweep 9°. Zhan [114] showed that the leading edge sweep angle has a strong effect on the aerodynamic performance. Their results have shown that the drag increases as sweep angle increases. The drag coefficient on fuselage, in Table 10, decreases as the LE-sweep angle increases.

The pitching moment, CM,ac, in Figure 55(d) indicates that the fuselage has a positive contribution on the static longitudinal stability. The wing planform was statically longitudinally unstable at α < 5°, as showed in Section 4.3. However, the wing-fuselage models are all statically longitudinally stable at α > 0° (i.e. MAV is operating at incidences between 0° and 5°). The rectangular-fuselage model shows a negative moment slope for all incidences. At CL,max region, the unstable variation of the pitching moment curve for wing-fuselage is degraded, and less contribution from the fuselage is observed as the leading edge sweep angle increases. This is because the sweepback wing move the aerodynamic centre further downwards trailing edge as compare with the unswept wing-fuselage models. The sweepback angle also improves the static lateral stability which due to the natural dihedral effect.

The local lift distribution is also shown in Figure 56 at the design condition. All wings suffer a huge impact on the lift at wing root due to the fuselage effect (i.e. C*l* is roughly around 0.1 to 0.2). For rectangular and LE-sweep 0°, 9°, and 18° models, as towards the wingtip, the C*l* increases dramatically at z/b between 0.06 and 0.3. However, the LE-sweep 30° has a mild lift distribution along the wingspan except at the wingtip, and the relative constant C*l* valueis about 0.35. Both the rectangular and trapezoidal (with LE-sweep 0°, 9°, 18°) wing-fuselage progressively decreases the C*l* as the spanwise location between 0.3 and 0.8. The possible reasons are: a) the separated area on the upper surface near negative camber area is modified, and b) a spanwise flow is introduced due to the wing swept angles effect. The sudden increase on C*l* at z/b beyond 0.8 indicates the vortical flow at the wingtip area. Viieru [27] also studied the tip-vortex effects on the low-aspect ratio MAV wings.

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| K:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAVs_with_fuselage_84ms\post_process\MAV1_4_Fuse_CL.emf   1. CL | 1. CL (zoom in) |
| K:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAVs_with_fuselage_84ms\post_process\MAV1_4_Fuse_CD.emf   1. CD | K:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAVs_with_fuselage_84ms\post_process\MAV1_4_fuse_CM.emf   1. CM |

Figure Aerodynamic performances for swept wing-fuselage

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Figure 56 Spanwise C*l* distribution at design condition

*Surface and 3D streamlines*

Before analyzing the 3D flow structures for the wing-fuselage models, streamlines (from Zimmerman wing-fuselage) at wing root are used to show the critical points (the spatial derivative of the velocity are zero at those points) and bifurcation lines in the flow. The critical points are usually grouped as nodes, foci, and saddles. The node points are stable and the foci points are unstable. “The bifurcation lines are lines drawn in the flow toward which the trajectories are asymptotic” [89]. In general, they are two different types: the negative bifurcation (NBL, means the flow trajectories approach) and the positive bifurcation (PBL, means the flow trajectories diverge), and NBL, PBL are also called the separation and reattachment line, respectively. Figure 57 shows the flow topology at the symmetric plane, and according to Perry [89], the relationship between the saddle and node points need to obey.

|  |  |
| --- | --- |
|  | Eq. |

where the number of nodes and foci, the number of half-nodes (nodes on the boundaries), the number of the saddles points and the the saddle points on the boundaries. *n* is the connectivity of the section of the flow, n = 1 for a single connected region without body, n = 2, for one body, (i.e. as like our case), and n = 3 for two bodies.

To analyze whether the flow topology showed in Figure 57 satisfies the topology constraint in Eq. 79, we have, , , ans , so that

This satisfies the constraint in Eq. 79, since *n* = 2 for our case. Therefore, the figure represents a flow that is kinematically possible.

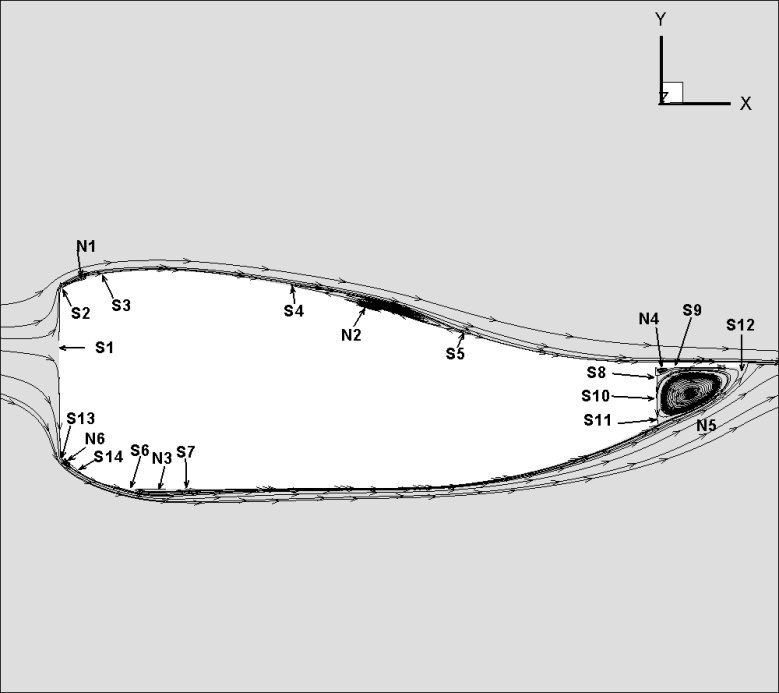


Figure flow topology at root location for Zimmerman wing-fuselage model at CL = 0.35 (3D flow is also shown in b)

Figure 58 shows the surface streamlines for both rectangular and trapezoidal wing-fuselage models at the design condition. In contrast, the spiral flow structure is formed on the upper wing surface near the negative camber region (Figure 58). However, fully attached flow was observed on upper surface of the isolated wing planform models (). This spiral vortex can cause a low pressure area on the upper wing surface. Zuo [94] pointed out that the disappearance of the spiral vortex corresponds to the abrupt drop of CL. The wing-fuselage junction area, on the other hand, has a majority effect on the flow structure, especially with the vortical flow formed on both sides of the fuselage. Similar 3D vortical flow structures are observed on both rectangular and LE-sweep-0° wing-fuselage models. However, the vortical flow structure shrinks as the leading edge sweep angle increases. This may due to the stronger spanwise flow is induced as the leading edge sweep angle is increased.

The spanwise Cp distributions at different chordwise locations are shown in . Three different chordwise locations: x/c = 0.25, 0.58 and 0.85 are representing maximum positive camber location, zero camber location and maximum negative camber location, respectively. The suction peaks are most prominent in the plots for the 25%c locations, indicating the wing tip vortex effect. Similar peak Cp values at wing tip locations are observed from both rectangular and trapezoidal wing-fuselage models. The fuselage effect also is indicated by a sudden drop of Cp at wing root regime.

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| --- | --- |
| J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAVs_with_fuselage_84ms\MAV1_4_design_condition\MAV1_design_condition_upper_surface2-1.jpg   1. MAV1 (up surface) | E:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAVs_with_fuselage_84ms\MAV1_design_condition_lower_surface2.jpg  MAV1 (lower surface) |
| J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAVs_with_fuselage_84ms\MAV1_4_design_condition\sweep1_design_condition_upper_surface2-1.jpg   1. LE-Sweep-0° (up surface) | E:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAVs_with_fuselage_84ms\sweep1_design_condition_lower_surface.jpg  LE-Sweep-0° (lower surface) |
| J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAVs_with_fuselage_84ms\MAV1_4_design_condition\sweep3_design_condition_upper_surface2-1.jpg   1. LE-Sweep-9**°** (up surface) | E:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAVs_with_fuselage_84ms\sweep3_design_condition_lower_surface.jpg  LE-Sweep-9° (lower surface) |
| J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAVs_with_fuselage_84ms\MAV1_4_design_condition\sweep2_design_condition_upper_surface2-1.jpg   1. LE-Sweep-18**°** (up surface) | E:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAVs_with_fuselage_84ms\sweep2_design_condition_lower_surface.jpg  LE\_Sweep-18° (lower surface) |
| E:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAVs_with_fuselage_84ms\sweep4_design_condition_upper_surface2.jpg   1. LE-Sweep-30**°** (up surface) | E:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAVs_with_fuselage_84ms\sweep4_design_condition_lower_surface.jpg  LE-Sweep-30**°** (lower surface) |

Figure Stream patterns at design condition, flow from left

|  |  |
| --- | --- |
| **Design Condition (CL = 0.35): wing-fuselage models** | |
| K:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAVs_with_fuselage_84ms\MAV1_4_design_condition\MAV1_Cp_spanwise.emf   1. Rectangular: (α = 2.27°) | K:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAVs_with_fuselage_84ms\MAV1_4_design_condition\Sweep1_cp_spanwise.emf   1. LE-swept-0°: (α = 3.27°) |
| K:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAVs_with_fuselage_84ms\MAV1_4_design_condition\Sweep3_Cp_spanwise.emf   1. LE-swept-9°: (α = 2.93°) | K:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAVs_with_fuselage_84ms\MAV1_4_design_condition\Sweep2_Cp_spanwise.emf   1. LE- swept-18°: (α = 2.45°) |
| K:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAVs_with_fuselage_84ms\MAV1_4_design_condition\Sweep4_Cp_spanwise.emf   1. LE-swept-30°: (α = 2.42°) |  |

Figure 59 Spanwise Cp distribution at different chordwise locations

**Zimmerman and inversed-Zimmerman wing-fuselage aerodynamics**

Fuselage effects on Zimmerman and inversed-Zimmerman wings are discussed in this section. Both models shows a better CL,max than the rectangular and trapozoid wing-fuselage models. However, a less linear CL curve is found. Sudden stall is observed from the Zimmerman wing-fuselage, however, inversed-Zimmerman wing-fuselage model shows a mild decrease on CL at post-stall region. Both models are shown a similar pitching moment trend with a negative slope, CMα < 0, for angle of attack between -3**°** and 27**°**, indicating statically longitudinally stable.

*Aerodynamic performances*

The plot depicted in (a) displays the models’ polar diagrams lift coefficient CL versus the angles of attack. The lift slopes are nearly constant for angles of attack α ≤ 0°, and less linear CL curves are found at α ≥ 0°. Zimmerman wing-fuselage shows an earlier stall angle (αstall = 12°), whereas, inversed-Zimmerman-fuselage stalls at about 15°. In general, the Zimmerman wing-fuselage has higher lift than the inversed-Zimmerman wing at all incidences before stall. The upward shift of the lift curve for the Zimmerman wing is due to its higher effective positive camber from the larger local span at around 25%c. (b) shows the drag coefficient for both models, and the Zimmerman wing shows lower CD values for α < 15**°**. Fuselage with less drag value is found from the Zimmerman wing-fuselage model, and results are listed in .

The static longitudinal stability, detailed via the pitching moment coefficient about the 25% of the mean chord (Figure 60 c). The vehicles are highly sensitive to the longitudinal position of the centre of gravity. The stability margin is defined as the distance between the centre of gravity (CG) and the aerodynamic centre (AC). Albertani [29] mentioned that there is only few percent of the mean aerodynamic chord, less than 10mm long in their case of MAV. Similar pitching moment trends are shown in (c) which are statically longitudinally stable (at α > -3°).

Figure 61 shows the lift distribution along the span at the design condition, and the fuselage shows a clearly effect on lift distribution at the wing root location for both models (z/b < 0.113). The Zimmerman wing shows a lower C*l* distribution at z/b < 0.57 and the lift increases significantly as towards wing tip. The inversed-Zimmerman wing-fuselage, on the other hand, shows higher C*l* distribution at z/b < 0.57. This may due to the vortical flow occurred on the inversed-Zimmerman upper surface which contributes the nonlinear lift. Zuo [94] who also mentioned that the spiral vortex can cause a low pressure area on the surface, and enhance lift. Another interesting point is that both models show a strong C*l* distribution at wingtip area. However, the inversed-Zimmerman-fuselage model shows a smaller Cp peak value at wingtip. This is because the inversed-Zimmerman wing-fuselage is designed to have a larger area at the maximum reflex camber position (which at x/c = 0.85) where the negative lift is produced. Nevertheless, the Cp peak value at wingtip area from inversed-Zimmerman wing-fuselage model is still quite pronounced. The corresponding surface flow pattern in Figure 62 clearly showed the flow structure for both Zimmerman and inversed-Zimmerman wing-fuselage MAVs.

|  |  |
| --- | --- |
| K:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAVs_with_fuselage_84ms\post_process\MAV3_4_Fuse_CL.emf   1. CL | K:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAVs_with_fuselage_84ms\post_process\MAV3_4_fuse_CD.emf   1. CD |

K:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAVs_with_fuselage_84ms\post_process\MAV3_4_fuse_CM.emf

c) CM

Figure Aerodynamic coefficients for Zimmerman and inversed-Zimmerman wing-fuselage MAVs

I:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAVs_with_fuselage_84ms\MAV1_4_design_condition\MAV3_4_2d_lift.emf

Figure spanwise lift distribution at design condition

*Surface and 3D streamlines*

The flow structure is given at the design condition for both inversed-Zimmerman and Zimmerman wing-fuselage models are shown in (a, c) and (b, d) for the upper and lower surfaces, respectively. The plotted streamlines reside close to the surface typically near the boundary layer. Over the upper surface of the inversed-Zimmerman wing-fuselage model ( a), a large spiral vortex structure is formed at the centre part of the wing. This would give a low pressure regime to provide the additional lift (which also discussed in the 2D spanwise lift distribution in ). Separated areas are found at the wing-fuselage junction area, especially at both the back of the fuselage and near the leading edge area.

The Zimmerman wing-fuselage (-c and –d) shows a low pressure region near the leading edge, corresponding to the flow separation. A small separation bubble close to the 3/4 root chord is formed in-board portion of the wing. The appearance of this separation bubble would re-curved the negative camber of the original aerofoil (reduce the amount of the reflex camber), and hence to reduce the overall negative lift. The lower surface (-d) shows a similar flow structure compared with the inversed-Zimmerman model. In general, separated area on the inversed-Zimmerman wing is larger than that on the Zimmerman model. This leads to increase the form drag and therefore poorer aerodynamic performance. The large leading edge vortices, on the inversed Zimmerman model, induce strong lateral flow toward the edge and generate the secondary vortices, in (a). This phenomenon can be attributed to the additional lift from wing tip vortices, which counteracts the negative effects of flow separation on lift.

The spanwise pressure distributions at different chordwise locations are shown in . Locations x/c = 0.25, 0.58, and 0.85 are ploted. Note that the inversed-Zimmerman wing’s maximum span is located at x/c = 0.85, however, the Zimmerman wing’s at x/c = 0.25. (a) shows the chordwise locations and (b) shows the corresponding Cp values. In general, the fuselage shows a major effect on the Cp distribution at wing root area for both models. On the inversed Zimmerman wing, a pressure suction peak region between z/b = 0.65 and z/b = 0.78 is observed at location of x/c = 0.25 due to the addition lift generated from the strong leading edge vortex. A vortex suction region is also observable at x/c=0.58 but it disappears at x/c=0.85 as the leading edge vortex lifts off the surface. For the Zimmerman wing, the vortex lift (suction) is only observable at x/c=25%c to a significantly smaller scale. This can be better comprehended by combining with the wing tip vortex structures shown in -c. At the design condition of the same lift, the inverse Zimmerman wing benefits from the vortex lift but the Zimmerman wing more from the positive camber effect.

|  |  |
| --- | --- |
| E:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAVs_with_fuselage_84ms\MAV3_design_condition_upper_surface.jpg   1. Inversed-Zimmerman (up surface) | E:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAVs_with_fuselage_84ms\MAV3_design_condition_lower_surface.jpg   1. Inversed-Zimmerman (lower surface) |
| E:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAVs_with_fuselage_84ms\MAV4_design_condition_upper-surface.jpg   1. Zimmerman (up surface) | E:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAVs_with_fuselage_84ms\MAV4_design_condition_lower_surface.jpg   1. Zimmerman (lower surface) |

Figure Instantaneous stream pattern at design condition (CL = 0.35)

|  |  |
| --- | --- |
| **Design Condition: Wing-fuselage models** | |
| **Inverse-Zimmerman: (α = 1.52°)** | **Zimmerman: (α = -0.14°)** |
| K:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\Zim_camber_location.emf   1. Cp distribution locations | K:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\Zim_camber_location.emf   1. Cp distribution locations |
| K:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAVs_with_fuselage_84ms\MAV1_4_design_condition\MAV3_Cp_spanwise.emf   1. Inverse Zimmerman | K:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAVs_with_fuselage_84ms\MAV1_4_design_condition\MAV4_Cp_spanwise.emf   1. Zimmerman |

Figure Mean spanwise Cp distribution at different chordwise location

### Transition region

In the solution process, the intermittency factor γ triggers the local transition on the wing surface. The intermittency function turns on the production of the turbulence kinetic energy downstream of the transition point based on the relation between the transition momentum thickness and the strain-rate Reynolds number, which is set to zero in the laminar region and increases to one for fully turbulent flow.

and show the intermittency factor γ distributions along the chord at four spanwise locations, for the inverse Zimmerman and Zimmermann wings, respectively. and show the corresponding turbulent kinetic energy contour plots. The steep increase of the intermittency factor on the upper surface is clear shown, indicating transition to turbulent flow. The inversed-Zimmerman planform ( and ) shows that γ starts to increase approximately at a location of 30%c at z/b = 0.23 span and reaches a peak value of 0.9 at about 40%c. Towards the wingtip, transition location is found at 45%c and a maximum value of 0.6 at bout 60%c was shown at the spanwise location of z/b = 0.68. At the wingtip, z/b = 0.9, vortical flow dominates and the flow becomes laminar beyond 40%c. The lower surface is dominated by laminar flows away from the inboard span region.

The transition region for the Zimmerman wing is shown in and Figure 67. The transition zone is located approximately between 30%c and 40%c at z/b = 0.23 on the upper wing surface. The transition region moves towards the trailing edge as the span location moves to the wingtip (i.e. 46%c to 62%c at z/b = 0.68). At the wing tip, the flow is dominated by laminar flows. The lower surface shows a much longer transition zone with gradual reduction of the intermittency towards the trailing edge except near the tip.

|  |  |
| --- | --- |
| L:\jasonChen\PHD_year3\MAV4_zim\MAV4_propeller\MAV4_prop_x\FLUENT\unsteady_MAV3_MAV4\isolated_wing\MAV4\10ms\MAV3_intermittency_005.emf   1. Z/b = 0.23 | L:\jasonChen\PHD_year3\MAV4_zim\MAV4_propeller\MAV4_prop_x\FLUENT\unsteady_MAV3_MAV4\isolated_wing\MAV4\10ms\MAV3_intermittency_01.emf   1. Z/b = 0.45 |
| L:\jasonChen\PHD_year3\MAV4_zim\MAV4_propeller\MAV4_prop_x\FLUENT\unsteady_MAV3_MAV4\isolated_wing\MAV4\10ms\MAV3_intermittency_015.emf   1. Z/b = 0.68 | L:\jasonChen\PHD_year3\MAV4_zim\MAV4_propeller\MAV4_prop_x\FLUENT\unsteady_MAV3_MAV4\isolated_wing\MAV4\10ms\MAV3_intermittency_02.emf   1. z/b = 0.9 |

Figure Instantaneous intermittency values at design condition for inversed Zimmerman wing

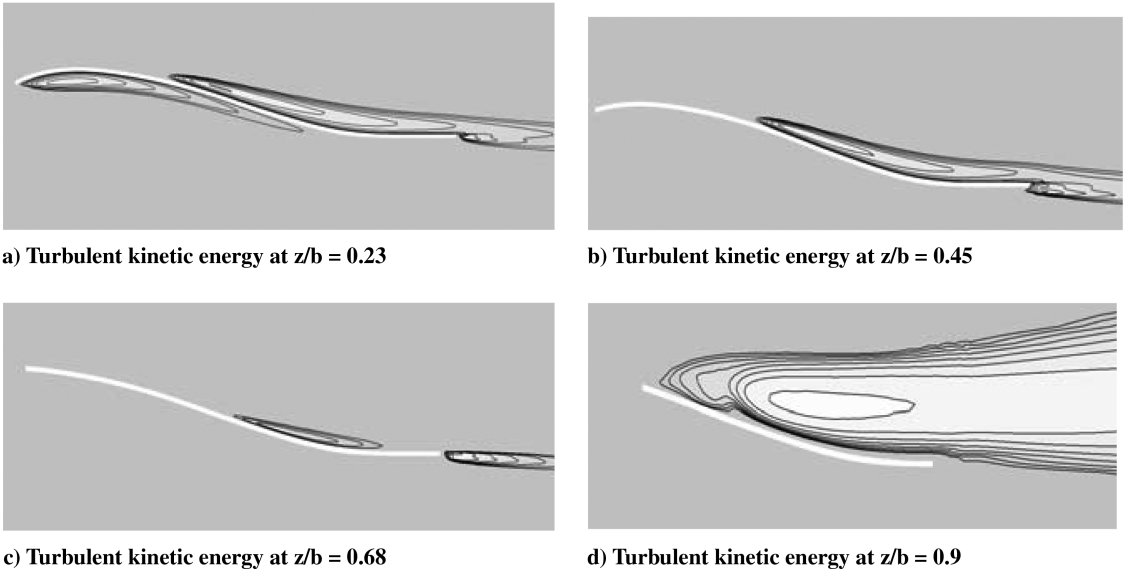


Figure Turbulent ke at different span location for inverse Zimmerman wing

|  |  |
| --- | --- |
| L:\jasonChen\PHD_year3\MAV4_zim\MAV4_propeller\MAV4_prop_x\FLUENT\unsteady_MAV3_MAV4\isolated_wing\MAV4\10ms\intermittency_005.emf   1. Z/b = 0.23 | L:\jasonChen\PHD_year3\MAV4_zim\MAV4_propeller\MAV4_prop_x\FLUENT\unsteady_MAV3_MAV4\isolated_wing\MAV4\10ms\intermittency_01.emf   1. z/b = 0.45 |
| L:\jasonChen\PHD_year3\MAV4_zim\MAV4_propeller\MAV4_prop_x\FLUENT\unsteady_MAV3_MAV4\isolated_wing\MAV4\10ms\intermittency_015.emf   1. z/b = 0.68 | L:\jasonChen\PHD_year3\MAV4_zim\MAV4_propeller\MAV4_prop_x\FLUENT\unsteady_MAV3_MAV4\isolated_wing\MAV4\10ms\MAV4_intermittency_02.emf   1. z/b = 0.9 |

Figure Instantaneous intermittency values at design condition for Zimmerman wing

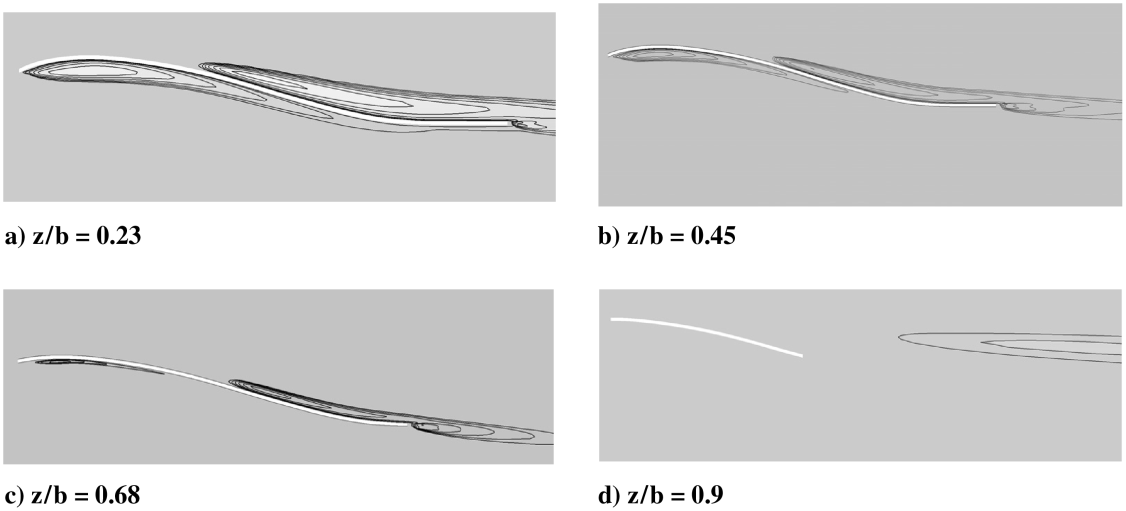


Figure Instantaneous ke values at different span location for Zimmerman wing

and Figure 69 show the intermittency factor γ distributions and turbulent kinetic energy along the chord at four spanwise locations for the Zimmerman wing-fuselage model at design condition. For spanwise location z/b = 0.23 (where close to the fuselage), the transition on upper surface starts at about 27%c (Figure 68 a) and ends at about 50%c (the corresponding high turbulent kinetic energy starts to occur, shown in Figure 69 a). On the lower surface, however, the pronounced turbulent kinetic energy is found near the leading edge, which presumably is the separation bubble occurred and the bubble length is roughly 40%c (Figure 68 a). Moving towards z/b = 0.45, the transition location moves towards trailing edge quite significantly (which roughly 55%c) compared with inboard spanwise location. The separation bubble length, on the lower surface, is shrunk a lit bit (see Figure 68 b). Further moving towards z/b = 0.68 and 0.9, fully laminar flow is occurred on both the upper and lower surfaces, except at z/b > 0.7 on the upper wing surface.

|  |  |
| --- | --- |
| L:\jasonChen\PHD_year3\MAV4_zim\MAV3_MAV4\unsteady\Wing_Fuse\MAV4_redesign\MAV4_Fuse_0P726deg_intermittency005.emf   1. z/b = 0.23 | L:\jasonChen\PHD_year3\MAV4_zim\MAV3_MAV4\unsteady\Wing_Fuse\MAV4_redesign\MAV4_Fuse_0P726deg_intermittency01.emf   1. 2/b = 0.45 |
| L:\jasonChen\PHD_year3\MAV4_zim\MAV3_MAV4\unsteady\Wing_Fuse\MAV4_redesign\MAV4_Fuse_0P726deg_intermittency015.emf   1. z/b = 0.68 | L:\jasonChen\PHD_year3\MAV4_zim\MAV3_MAV4\unsteady\Wing_Fuse\MAV4_redesign\MAV4_Fuse_0P726deg_intermittency02.emf   1. z/b = 0.9 |

Figure Instantaneous intermittency values at design condition for Zimmerman wing-fuselage

|  |  |
| --- | --- |
| L:\jasonChen\PHD_year3\MAV4_zim\MAV3_MAV4\unsteady\Wing_Fuse\MAV4_redesign\mav4_zim_fuse_0P73deg_unsteady_005.jpg   1. z/b = 0.23 | L:\jasonChen\PHD_year3\MAV4_zim\MAV3_MAV4\unsteady\Wing_Fuse\MAV4_redesign\mav4_zim_fuse_0P73deg_unsteady_01.jpg   1. z/b = 0.45 |
| L:\jasonChen\PHD_year3\MAV4_zim\MAV3_MAV4\unsteady\Wing_Fuse\MAV4_redesign\mav4_zim_fuse_0P73deg_unsteady_015.jpg   1. z/b = 0.68 | L:\jasonChen\PHD_year3\MAV4_zim\MAV3_MAV4\unsteady\Wing_Fuse\MAV4_redesign\mav4_zim_fuse_0P73deg_unsteady_02.jpg   1. z/b = 0.9 |

Figure 69 Instantaneous ke values at different span location for Zimmerman wing-fuselage model at CL = 0.35

### Logitudinal stability analysis for isolated-wing and wing-fuselage models

It is also been found that the location of the center of gravity is important in the flight stability of micro air vehicles [100, 115]. For the flying wing, the centre of mass lies in front of the neutral point (i.e. the point where the pitching moment is independent of angle of attack). The detailed study on aerodynamic centre for low aspect ratio wings were studied by Torres and Mueller [9]. Randall and Shkarayev [116] studied the AC locations variation for the fixed-wing VTOL (vertical take-off-and landing) MAV with the propeller included. However, if the centre of mass lies behind the neutral point, then the MAV is unstable. The larger the distance between cg and neutral point is, the greater is the measure of stability margin. shows that the moment is taken at the centre of gravity and the stability margins (the distance between the cg and the quarter mean chord) are calculated and listed in (the non-dimensional format is written shown in ). shows the static longitudinal stabilities for the wing-fuselage models. All models show the positive value of indicating statically longitudinally stable () and the corresponding values are listed in . The Zimmerman wing also shows a higher CL/CD = 5.37 at its maximum stability margin (CL/CD = 4.88 for rectangular-fuselage model). However, the inversed-Zimmerman, leading-edge sweep 0°, 9°, and 18° wing-fuselage models shows a lower CL/CD value with combined a smaller stability margin ().

|  |  |
| --- | --- |
|  | Eq. |
|  |
|  |

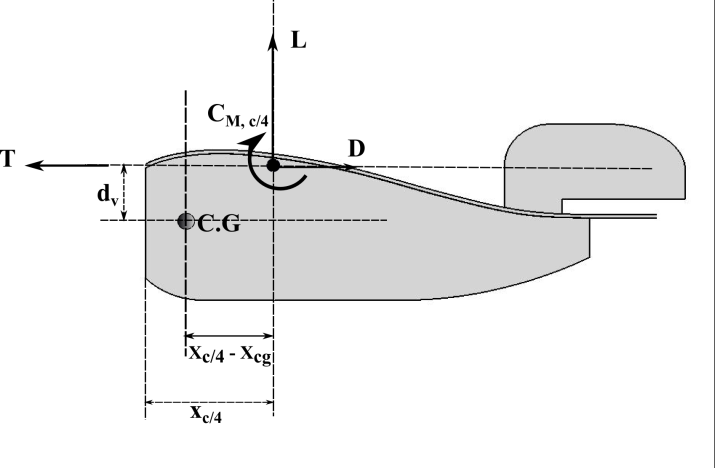


Figure moment balance for wing-fuselage model

J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV_Wing_fuselage_logitudinal_stability.emf

Figure 71 stability margin for wing-fuselage models

Table stability margin for wing-fuselage models

|  |  |  |
| --- | --- | --- |
| Case with fuselage |  |  |
| Rectangular | 0.0252 | 0.0097 |
| Tapered 0° LE | 0.0179 | 0.0069 |
| Tapered 9° LE | 0.0169 | 0.0065 |
| Tapered 18° LE | 0.0153 | 0.0059 |
| Inversed-Zim | 0.0333 | 0.0129 |
| Zimmerman | 0.0853 | 0.033 |

### Summary

Similar aerodynamic results are observed for trapezoidal wings with different leading edge sweeps in the low incidence range at a low Reynolds number. However, at higher incidences, the leading edge sweep directly affects the stall behaviour. Different lift-curve behaviours are found at near and post stall conditions for different planforms with positive and reflex cambers. A higher leading edge sweep gives a higher maximum lift, accompanied by a more sudden stall. On the other hand, a lower leading edge sweep produces less maximum lift and more gradual stall behaviour. All wing planforms except the inversed-Zimmerman wing show a positive pitching moment slope CMα at incidence α<5°, which indicates that the MAV wings themselves are not longitudinally stable at the design condition. The inversed Zimmerman wing however shows longitudinal stability at the design condition, due to the amplified reflex camber by the increased span near the rear of the wing. According to the overall aerodynamic performances, the Zimmerman wing shows the lift-to-drag ratio, CL/CD = 8, while the trapezoidal wing with a leading edge sweep above 18° produces the highest maximum lift.

The fuselage, however, shows a positive contribution on the static longitudinal stability. All wing-fuselage models are statically longitudinally stable at deign condition. On the aerodynamic side, the fuselage decreases the aerodynamic efficiency significantly from 7.4 to 5.2 for Zimmerman model at design condition (other wing-fuselage models are reduced by almost 50% of the total CL/CD at designed lift).

The flow visualization on both upper and lower surfaces is used to analyze the different flow structures at the design condition for the different models. The observations are:

1. Elliptical separation bubbles form on both upper and lower surfaces for the Zimmerman and inversed-Zimmerman wings. It is much larger for the latter. However, for trapezoidal wings, fully attached flow is observed on the upper surface, with separation bubbles forming at the leading edge region on the lower wing surface.
2. The local camber has a strong effect on the separation/reattachment location of the bubbles for the Zimmerman and inversed-Zimmerman wing. The upper surface separation starts around x/c =0.25, the maximum positive camber location, and reattached around x/c = 0.85, the maximum reflex camber location for both planforms.
3. The Zimmerman wing shows the wingtip vortex developing from the maximum span location while the inversed Zimmerman wing has a much earlier start of the wing tip vortices. The vortices generally follow the outline of the wing up to a point and then separate from the wing. The leading edge vortices affect a much larger area on the inversed-Zimmerman wing upper surface than that on the Zimmerman wing. This helps the inversed-Zimmerman wing to generate more vortex lift. The bubble on the Zimmerman and inversed-Zimmerman wing is limited to the inboard section of the wing because, near the wingtips, the tip vortices energize the flow, confining the separation bubbles. On the other hand, for the trapezoidal wings, the effects of the wingtip vortices on the flow structure are very limited.
4. For the Zimmerman and the inverse Zimmerman wings at the given design lift condition, transition starts around 30-40%c on the upper surface away from the wing tip after the laminar separation, the transition on the lower wing surface shows relaminarisation towards the trailing edge.
5. The fuselage shows a positive contribution on the static longitudinal stability. Flow structures with larger separation areas were observed on the wing-fuselage models at the design condition, except the Zimmerman-wing-fuselage model (which a smaller separation bubble is formed).

# Propeller Slipstream Effect

Most of the MAVs are equipped with an electric motored propulsion system that contributes to the simplicity of operation and significantly reduced noise signature [117]. The flow is often dramatically altered by the propeller and usually generates an unsteady pressure field downstream. Therefore, the aerodynamic performance for the wing planform only is no longer applicable after a propeller is installed. Marretta[10] found that the swirling flow significant modifies the surface pressure distribution and a considerable shifting of the centre of pressure occurs, resulting in a change in overall pitching moment. The swirling flow produced by the propeller also generates an additional yawing moment and a side slip can occur [118]. A detailed explanation of propeller swirling flow was discussed by Witkowski [119]. He showed that the propeller would effectively modify the angle of attack of the downstream wing, thereby changing the wing’s circulation. On the other hand, the local angle of attack of the propeller blades is also modified by the wing behind the propeller. The downward rotating blade increases the local angle of attack and increases the local lift and blade loading, which augments the thrust and torque on the blade.

From the flow status, the propeller modifies the flow structure of the wing as it causes the downstream flow fully turbulent. Lynch [120] pointed out that unlike the viscous wake, the propeller wake combines both normal wake (wake behind a body) and wake from the propeller’s response to inflow turbulence. Further, this cascade wake generated by the propeller has a significant higher intensity level, which would retard the upper wing surface separation and hence improve the stall characteristics. Catalano [121] found that turbulence separation points can be moved downstream by up to 56%c due to the propeller slip stream effect. He also pointed out that the propeller inflow effects are very dependent on the propeller/wing relative position, and the propeller effect will be small if the propeller is positioned more than one propeller diameter behind the wing, in a pusher configuration. The tractor configuration propeller effect on a flexible wing UAV was investigated experimentally by Gamble [122]. He found that as the wing moves closer to the propeller, the drag-to-thrust ratio of the wing is increased and the thrust is relatively unaffected by the presence of the wing. The mutual influence problem between the propeller and wing was investigated by Fratello [28]. He found a significant increase of the wing’s mean drag as compared to the case without the propeller. The induced lift was reduced at low rotational speeds, and the reciprocal wing influence on the propeller performance showed an increase of thrust and power coefficients. Most recently, the propeller effect has been investigated on the vertical take-off and landing Micro air vehicle vehicles [111] [123] [124], in which the authors found that the control surfaces becomes more effective as the slipstream strengthens. However, the overall wing aerodynamic efficiency was decreased by the slipstream.

In an effort to gain a physical understanding of the propeller-wing interaction problem and predict the performance changes for effective MAV designs, a computational study has been performed to investigate the mutual aerodynamic influences between an MAV configuration and its propeller, here.

### Geometry description and sliding mesh for propeller

The model used in this study is based on the flying MAV developed at the University of Sheffield [104, 125]. This model is composed of a Zimmerman planform wing, a fuselage, and a propeller in the tractor configuration. The model is shown in (b) with a coordinate system with x in the chordwise, y in normal, and z in spanwise direction. The wing is installed with a setting angle of 6.32°, (Figure 72 f). The vertical stabilizer, (Figure 72 c), is designed and installed on the up wing surface at rear towards the trailing edge. The propeller slipstream effect on the vertical stabilizer will be discussed in Section 5.5.

Considering the MAV structure, the fuselage is designed to carry all the electronic devices, such as battery, motor, and servos, etc. The fuselage dimensions are shown in Figure 52 and components are listed in Figure 53. This design of the fuselage was primarily dictated by the size and placement of the components. The internal volume, in Figure 52, the dashed line, is designed to investigate the centre of gravity margin and hence to evaluate the static stability margin. For testing purposes, the battery is designed to be movable to adjust the centre of gravity. To determine the static stability margin, a similar method to evaluate the centre of gravity can be found from the literature, [109]. Another interesting point is how the fuselage affects the overall aerodynamics. Brion [106] simulated the fuselage and wing separately and the relevant aerodynamic forces are showed and discussed. However, the authors did not mention anything about the interaction between the wing and the fuselage. Ramamurti’s [108] numerical results showed MAV with fuselage reduced the lift-to-drag ratio dramatically and the drag for all configurations considered is nearly the same. The effects of the fuselage are also investigated in the present study.

The GWS8040 slow-flyer propeller has been chosen for the MAV here. It operates at a Reynolds number of 9.3×104. The Reynolds number (Reprop) is based on three quarter radius of the propeller and the sea level conditions. The GWS8040 propeller is installed at a distance dt = 0.068m ahead of the wing planform, as shown in Figure 72 (b), and it has a diameter of 8in (Dia = 0.203m), the pitch is 4inchs (0.1016m), and the hub diameter is 0.014m. (a) shows the propeller geometry and the blade azimuth angle. The propeller rotates in an anti-clockwise direction viewing from the front, and Figure 73 (b) shows the propeller geometric characteristics. For this investigation, the rotational speeds are set from 500 rad/s to1131rad/s.

The vertical stabilizer, in (c), shows the full dimensions. The radius for the rounded corners is specified as r1, r2, and r3, and they are 0.022m, 0.024m and 0.037m, respectively. The relevant heights for the vertical stabilizer are: Vt1 = 0.045m, Vt2 = 0.04m and Vt3 = 0.0088m. The stabilizer length is divided into two parts as shown in the plot: Ht1 and Ht2, and they are 0.028m and 0.06m, respectively. The vertical stabilizer has been installed on the upper wing surface with a distance of 0.175m from the front of the fuselage.

MAV aerodynamics was investigated using the assembly of the wing-fuselage, vertical stabilizer and propeller, as described in sub-sections 5.3, 5.4, and 5.5. In this case, the aerodynamic balance determines a horizontal component X and vertical component Y of the total force acted on the MAV model, as shown in Figure 72 (d). To obtain the overall lift and drag force on the model, horizontal and vertical components (i.e. X and Y components) were transferred into L and D components (i.e. based on the incoming freestream coordinates), see the relevant equations from Eq. 82to Eq. 86. A slightly different propulsion system has been introduced by Randall [116, 126], of which the thrust axis is aligned with the wing’s chord line.

|  |  |
| --- | --- |
|  | Eq. |
|  | Eq. |
|  | Eq. |
|  | Eq. |
|  | Eq. |
|  | Eq. |

Table 12 Propeller location and fuselage specifications

|  |  |  |  |  |
| --- | --- | --- | --- | --- |
| MAV model | dt (dt/) | hf (hf/) | hL (hL/) | ht (ht/) |
| 0.068 (31.7%) | 0.059 (26.6%) | 0.216 (97.6%) | 0.021 (9.49%) |

|  |  |
| --- | --- |
| 1. Top view | J:\JasonChen\transfer report\propeller_effect\MAV4_Prop_0deg.wmf  ω  U∞   1. 3D view (no vertical stabilizer) |
| L:\Jason_chen\MAV4_Tail\0deg\tail_dimentions.png   1. Vertical stabilizer | L:\Jason_chen\MAV4_Tail\0deg\MAV4_prop_2012_new2_3_tail-1-03240_geo1.jpg   1. 3D view (with vertical stabilizer) |
| C:\Users\jason\Desktop\vector_plot\MAV_thrust_line.png   1. Orientation of forces | C:\Users\jason\Desktop\vector_plot\wing_setting _angle.png   1. Wing setting angel |

Figure 72 MAV geometrical descriptions

|  |  |
| --- | --- |
| K:\Jason_chen\PHD2\PHD_year2\MAV\propeller\papers\matlab\Propeller.emf  ψ   1. Blade azimuth angle, ψ | J:\Jason_chen\PHD2\PHD_year2\MAV\propeller\papers\matlab\geometry_paramatric.emf   1. GWS 8040-propeller geometrical characteristics |

Figure 73 Propeller specifications

A multi-block structured mesh was generated using the ANSYS ICEM program, and the mesh topology is shown in Figure 74. In general, the structured mesh topology contains two domains: the inner domain (i.e. the rotating domain contains the propeller) and the outer domain (i.e. the stationary domain which generated the block for the MAV). The mesh generation is done in several stages. First, the blocks associated to the propeller (i.e. the rotating domain is generated). Two O-topologies are created to cover the propeller and the centre spinner segments. The O-grid includes 10 cells normal to the propeller wall surface with a first cell distance of 2×10-4m, 30 cells in the propeller radial direction (i.e. each blade), and 56 cells in the circumferential direction. A cylindrical wake block is used to ensure a good resolution of the blade wakes and the tip vortices, of which the results would have significant influences on the MAV domain (i.e. the outer domain). The rotational domain has a total diameter of 6 (i.e. six mean aerodynamic chords), as shown in Figure 74 (d). The outer domain has a similar mesh topology as we showed in the validation case section, in Figure 33 (a). To have reasonable numerical results, the y+ value of the first grid point in the order of 1 is required, as shown in Figure 75. The aircraft mesh block has a total size of 8,000,000 nodes.

All these uRANS numerical simulations were run in Fluent-14 in parallel with 48 processors. The flow conditions set for the computation correspond to one of the flight tests. As mentioned earlier, the four equations Transition-SST model was selected for this investigation, solving for the pressure, the momentum, the turbulent kinetic energy (κ), the specific dissipation rate (ω), the intermittency (γ), and momentum thickness Reynolds Number (Reθ). The SIMPLE pressure-velocity coupling algorithm was used for the unsteady cases. For the final setting of the dual time method for the propeller rotations is 0.5 degree per time step, and thus a time step of 7.716e-6 s with 30 sub-iterations are selected for rotational speed of 1131rad/s.

|  |  |
| --- | --- |
| F:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\MAV4_prop_2012_new2_3_tail_more_mesh_n555-1-12240_mesh_tail.jpg   1. Mesh Topology in 3D view | F:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\MAV4_prop_2012_new2_3_tail_more_mesh_n555-1-12240_mesh_tail2.jpg   1. Front view |
| C:\Users\Jason\Desktop\MAV4_prop_tail_mesh\side_view.jpg   1. Mesh domain in top view | C:\Users\Jason\Desktop\MAV4_prop_tail_mesh\front_view.jpg   1. Mesh domain in front view |

Figure 74 Mesh topology for MAV-Propeller model

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Figure 75 y+ value for MAV-propeller model

### Isolated propeller aerodynamic performance

The isolated propeller (GWS8040) has been investigated with various rotational speed (ns). The rotational speed changes from 500 to 1131 rad/s. The airload measurements (averaged thrust and torque) are evaluated. The standard propulsion equations have been used to calculate the non-dimensional propeller performance parameters. These coefficients are the thrust coefficient CT, the torque coefficient Cq, the power coefficient Cpo, the advance ratio J, and the propeller efficiency η, as shown from Eq. 87 to Eq. 92.

|  |  |
| --- | --- |
|  | Eq. |
|  | Eq. |
|  | Eq. |
|  | Eq. |
|  | Eq. |
|  | Eq. |

The advance ratio, J, is a measure of the vehicle speed against the propeller rotation. Increasing the propeller speed increases the efficiency, the thrust coefficient, and the power efficiency. However, propeller performances reduced at a high advance ratio.

shows the propeller performances for different advance ratios. The peak efficiency is around 48% with an advance ratio of 0.3. The corresponding values of Cq, CT, and Cpo are 0.00346, 0.0352, and 0.0217, respectively. Table 13 shows the isolated propeller mean aerodynamic force and moment values with advance ratio J = 0.23. The positive sign of drag value, CD, Prop = 0.83, indicating the thrust force (i.e. thrust is opposite direction of the drag force). Both lift and side force coefficients (i.e. CFx,Propeller and CFz,Propeller) are showing a nearly zero magnitude, which indicates that a symmetrical force has been produced by the isolated propeller.

**G:\jasonChen\PHD_year3\MAV4_zim\MAV4_propeller\MAV4_prop_x\FLUENT\papers\matlab\Propeller_only_data.emf**

Figure 76 Cq, CT, CPo, and η as a function of J for isolated propeller at α = 0°

Table Mean aerodynamics for isolated-propeller at J = 0.23

|  |  |  |  |  |  |  |
| --- | --- | --- | --- | --- | --- | --- |
| Configuration | CL, Prop | CD, Prop | CFz, Prop | CMx, Prop | CMy, Prop | CMz, Prop |
| Propeller-only | 1.23e-5(or 0) | 0.83 | 1.3e-5(or 0) | -0.030 | -1.22e-5 | -9.80e-3 |

### Propeller slipstream effect on MAV model (ω = 1131 rad/s)

There have been some previous investigations on both pusher and tractor propeller configurations. The tractor propeller was found to have higher aerodynamic efficiency than the pusher propeller, and it also produces better control surface’s efficiency at landing and take-off. Although, the overall drag can be reduced due to the pusher propeller suction effect. Generally for UAVs, the ground clearance is also more problematic for installing the pusher propeller at rear. The pusher configuration increases the propeller thrust, and the tractor configuration manifests itself mainly on the wing-induced drag reduction.

The MAV model considered here has a tractor configuration, and it is mainly because the vector thrust system is chosen here to improve the overall manoeuvrability. The vector-thrust-system consist a servo on the top wing surface which controls the fly direction, showed in Figure 3(b). Figure 77 illustrates the tractor propeller configuration effect on left- and right-side of the Zimmerman wing. The propeller wake flow generates areas of upwash and downwash on the wing. Figure 77 (a) shows the wing-aerofoil aerodynamic loading in the upwash case where the propeller tangential velocity is greater than the downwash. The resultant force is tilted forward and a localized wing thrust is created at this section. Figure 77 (b) indicates the aerodynamics of wing section under the propeller downwash. The wing downwash and propeller tangential velocity act together and this makes a further reduction of local angle of attack. The resultant force, F, shifts further backwards and results in an increased section drag.

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| J:\Jason_chen\PHD2\PHD_year2\MAV\propeller\papers\matlab\Camber_prop_UpWash.emf   1. Wing section in propeller up-going blade | J:\Jason_chen\PHD2\PHD_year2\MAV\propeller\papers\matlab\Camber_prop_downWash.emf   1. Wing section in propeller down-going blade |

Figure 77 Propeller-wing in tractor configuration

#### MAV aerodynamic performance

The aerodynamic performances, CL, CD, and CM with various angles of attack at J = 0.23 are presented in Figure 78 (a-d). Figure 78 a) shows the lift coefficients CL of both models with/without propeller installed. The lift slope CLα is very similar and is the same at angles of attack α < 0°, and the slope reduces for the wing-fuselage model as angles of attack increase further. However, the lift slope, CLα, remains approximately constant until the incidence α reaches the stall angle, αstall = 12° and a higher slope is found at incidence between 12° and 24°. In contrast, the wing-fuselage model has a stall angle of α = 15° and the maximum CL is around 0.75 which is less than half of the CL, max obtained by the propeller model. This is because the “slipstream effect” from the propeller, which delays the separation, and hence increases the CL, max. In general, the wing lift is augmented by the increase in axial velocity in the propeller slipstream.

The drag coefficient, CD with a negative sign, means the propeller thrust is in positive direction. Figure 78 b) indicates that the propeller slipstream effect has a large impact on the drag force. The wing-fuselage model has a drag coefficient at zero angle of attack, CD0 = -0.0677. However, CD0 from the wing-fuselage-propeller is approximately three times larger than that from the wing-fuselage. A quite similar drag plot is found at incidence between -6° and 12°. The drag coefficient plot shifted up for the propeller installed model. As the incidence increases further, two different drag trends are found. For the propelled model, the drag coefficient increases dramatically as incidence increases further. For the wing-fuselage model, on the other hand, a relative gradual increasing in drag can be seen at angles of attack greater than 12°.

The pitching moment coefficient, CM,ac, as mentioned earlier, is defined from the quarter mean aerodynamic chord . The moment coefficient plot, CM,ac versus α, is shown in Figure 78 c). It clearly shows that the moment coefficient has significant difference as compared to the isolated wing. A slightly positive slope is obtained at low incidence range between -9° and 0° and it turns into negative slope as the incidence increases.

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| E:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV4_zim\MAV4_propeller\MAV4_prop_CL_2_unsteady.emf   1. CL,MAV | E:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV4_zim\MAV4_propeller\MAV4_prop_CD_2_unsteady.emf   1. -CD,MAV (means thrust is positive) |
| E:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV4_zim\MAV4_propeller\MAV4_prop_CM_2_unsteady.emf   1. CM,MAV |  |

Figure 78 Mean coefficient values as a function of α at J = 0.23 (forces and moments on wing-fuselage only)

a) shows the Cp distribution on the Zimmerman wing at r/R = 1 (or at the wing span location of 2z/b = ± 0.5). The Cp distribution locations are indicated by the dash line (i.e. up-going blade and down-going blade side). Both Cp with and without propeller slipstream effects at spanwise locations 2z/b = ± 0.25 and ± 0.5 with 0° angle of attack are presented in b-e respectively: the instantaneous Cp distribution are plotted with various azimuth angles, shown in solid lines and the Cp plot for isolated MAV-fuselage as shown by dashed line. The Cp plots show that the positive lift force is generated at the positive camber region (where roughly between leading edge and x/c = 0.53) and the negative camber contribution is to generate the negative lift force, which produces the restoring moment for such cambered thin wings. The turning point is the point where the negative lift force starts to occur, and it clearly shows that the turning point on the down-going blade side is a bit ahead than the up-going blade side at 2z/b = 0.25 with ψb = 0°, as shown in Figure 79 (b). Towards the wingtip, at 2z/b = 0.5, the turning point is less difference on up- and down-going blade sides, as shown in Figure 79 (c). The propeller slipstream effect is reduced towards the wingtip and hence less differences on turning point are observed at 2z/b = 0.5. When the propeller rotates at ψb = 90°, the difference of turning points at 2z/b = 0.25 on up- and down-going blades are more significant, shown in Figure 79 (d) and the turning point locations are almost overlapped towards the wingtip at 2z/b = 0.5, showed in Figure 79 (e). In general, turning point locations with less differences are observed towards the wingtip, and more negative lift can be produced with the turning point shifts towards the leading edge, and hence more restoring moment is produced from the negative camber region.

The Cp plot in Figure 79 also shows that different suction peaks are formed on the wing at inner and outer spanwise locations, 2z/b = 0.25 and 0.5, respectively. Higher suction peaks are formed on the up-going blade side and the location is close to the leading edge. However, on the down-going blade side, a reduced suction peak has been found and it has been shifted backwards as compared with the up-going blade side. The negative lift is also generated on the down-going blade side at the leading edge, due to a large separation bubble being formed on the lower surface at the down-going blade side. Towards the wingtip, at 2z/b = 0.5, Cp less differences are observed at negative camber region, whereas at leading edge region, the difference on the lift is still quite pronounced. The asymmetric pressure distribution also indicates that the additional torque is generated by the propeller slipstream, which would lead to a difficulty in the moment balance, both longitudinal and lateral, and hence it would affect the overall handling control. This problem can be important in the design of modern micro air vehicles, because autonomous computed flight systems do not impose the same constrains as those by human pilots. In other words, a properly designed vertical tail is required and the setting angle for the vertical tail is unavoidable in the case of no additional trim moment input from the pilot.

The skin friction coefficient in x direction is given in Figure 81 (a-d). Fully attached flow on both upper and lower surface except at the leading edge region can be observed. The length of the bubble on up-going blade side (roughly the length is about 0.09c) is much shorter than on the down-going blade side (which is about 0.3c) at 2z/b = 0.25 with ψb = 0°. Towards the wingtip, the length of the bubble has been shortened to about 0.2c on down-going blade side, as shown in Figure 81 (b). For propeller azimuth angle ψb = 90°, no significant changes can be observed.

In general, the propeller slipstream has a significant effect on both positive and negative lift. Negative lift is generated at both leading and trailing edge area on the downward-going blade side and asymmetric forces were produced due to the propeller slipstream, as shown clearly from the pressure distribution in Figure 79. The skin friction drag, in Figure 81, shows high value at leading edge and it is reduced towards the trailing edge on the down-going blade side. An opposite skin friction trend has been observed on the up-going blade side. The averaged pressure coefficient distribution for 2z/b = ± 0.5 is shown in Figure 80. The unbalance forces on each side of the wing can be seen clearly, and the difference of the turning point (i.e. from positive lift to negative lift) on both sides can be clearly observed.

|  |  |
| --- | --- |
| J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV4_Prop_Cp_location_lines.emf  Downward-going blade  Upward-going blade   1. Spanwise location (“--”: propeller tip location) | |
| F:\Jason_chen\more_mesh(no_vertical_tail)\0deg_more_mesh_no_tail\Cp_2zb005_12600.emf   1. ψ = 0°, 2z/b = 0.25 | F:\Jason_chen\more_mesh(no_vertical_tail)\0deg_more_mesh_no_tail\Cp_2zb01_12600.emf   1. ψ = 0°, 2z/b = 0.5 |
| F:\Jason_chen\more_mesh(no_vertical_tail)\0deg_more_mesh_no_tail\Cp_2zb005_11700.emf   1. ψ = 90°, 2z/b = 0.25 | F:\Jason_chen\more_mesh(no_vertical_tail)\0deg_more_mesh_no_tail\Cp_2zb01_11700.emf   1. ψ = 90°, 2z/b = 0.5 |

Figure 79 Cp distributions with J = 0.23 and α = 0° (No vertical stabilizer installed)

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Figure Mean Cp distribution at 2z/b = ± 0.5 with J = 0.23 and α = 0° (no vertical stabilizer installed)

|  |  |
| --- | --- |
| F:\Jason_chen\more_mesh(no_vertical_tail)\0deg_more_mesh_no_tail\Ctx_2zb005_12600.emf   1. ψ = 0°, 2z/b = 0.25 | F:\Jason_chen\more_mesh(no_vertical_tail)\0deg_more_mesh_no_tail\Ctx_2zb01_12600.emf   1. ψ = 0°, 2z/b = 0.5 |
| F:\Jason_chen\more_mesh(no_vertical_tail)\0deg_more_mesh_no_tail\Ctx_2zb005_11700.emf   1. ψ = 90°, 2z/b = 0.25 | F:\Jason_chen\more_mesh(no_vertical_tail)\0deg_more_mesh_no_tail\Ctx_2zb01_11700.emf   1. ψ = 90°, 2z/b = 0.5 |

Figure CTx distributions with J = 0.23 and α = 0° (No vertical stabilizer installed)

The asymmetric flow generated by the propeller blade has a pronounced impact on the forces acting on the MAV as well as on the propeller itself. The unsteady numerical results show a detailed force development presented in the following section. Figure 82 shows the development of the aerodynamic forces acting on the propeller during one cycle (i.e. one revolution). All the values are calculated based on the reference values: wing area, S = 0.0895m2, mean chord, and the wingspan b = 0.44m. It is noticeable that the evaluation of the forces is performed in a stationary coordinates system fixed to the MAV as shown in Figure 72 (d), (i.e. according to the right hand rule, x is the thrust (i.e. opposite to the incoming flow direction) direction, y is pointing up, the lift, and the z axis represents the side force or pitching moment axis).

The integration of blade aerodynamic loads provides strong periodic influenced on the propeller forces projected in the absolute reference frame, in Figure 82 (a-b). The evolution of the thrust coefficient, Figure 82 (a), is nearly sinusoidal as a function of the azimuth angle ψ°. The positive sign indicates the thrust force (i.e. drag force in a negative direction based on the right hand rule). The cycle in Figure 82 (a) is related to the passage of the two-blade propeller in front of the wing, which causes a periodic unsteady fluctuation with two peaks. The peak values are obtained when the propeller has just past the horizontal position (i.e. ψ = 100° and 280°, as shown in b) and the minimum values are at its vertical position (i.e. ψ = 0° and 180°). Such sinusoidal evolution is attributable to the unsteady variation of the local flow the blades experience during the rotation. Since the propeller is coupled with the wing and fuselage, the comparison can be made between the isolated propeller (Table 13) and the propeller-wing configuration (i.e. in Table 14). The thrust coefficient for the isolated propeller with an advance ratio of 0.23 is about 0.83. However, after propeller is coupled with the wing and fuselage, the propeller thrust coefficient is increased to 1.036 which is nearly enhanced by one fifth. The reciprocal influence of the wing on the propeller, as expected, is probably caused by the straightening effect of the wing on the propeller slipstream. The mean value of the thrust coefficient, in Figure 82 (a), is actually larger than the required drag coefficient (i.e. CD, MAV = -0.1817), indicating that the MAV is accelerating at 0° for the advance ratio of J = 0.23. Therefore, for a steady state flight condition, a higher advanced ratio may be considered (i.e. lower propeller rotational speed).

The evolution of propeller lift coefficient CFy(propeller), in Figure 82 (b), is performed in a stationary coordinate system. This asymmetric sinusoidal evolution has a minimum value when the blade position of ψ = 270° during the upward sweep and the minimum value is obtained at ψ = 45° on the downward sweep. Such asymmetric phenomenon would usually be due to the presence of the wing (i.e. the propeller-wing interaction, or mutual interaction problem). The lift force coefficient due to the presence of the wing is increased significantly as compared with the isolated propeller (CL, Propeller-only = 1.23e-5). However, it is still quite small as compare to the wing lift coefficient (CL,MAV = 0.38).

The MAV-propeller side force development in Figure 82 (c) shows an asymmetric evolution. The side force coefficient of the propeller shows a positive mean value about 0.0013. It is in the opposite direction as compare to the negative side force from the wing (CFz, MAV = -0.0087, see Table 14). The averaged side force coefficient of the isolated propeller, CFz(propeller-only) = 1.28e-5, in Table 15, is negligible as compared with the MAV-propeller’s. The overall side force coefficient for MAV-propeller is -0.0074 (Table 16).

The moment values, rolling (Mx), yawing (My), and pitching (Mz), are also listed in Table 14 and Table 15 for MAV-Propeller and isolated-propeller configurations. The average rolling moment coefficient for an isolated-propeller is roughly around -0.03, and the rolling moment with presence of the MAV is increased to -0.0352. The reciprocal influence of the wing on the propeller for the aerodynamic forces and moments changing are defined as:

*The forces/moments are computed from the model which contains both MAV-fuselage and the propeller, and take off the forces/moments are computed from the isolated propeller model (i.e. model contains the propeller only), shown in . The mutual interaction aerodynamic forces/moments can be found in* Table 15*.*

|  |  |
| --- | --- |
| C:\Users\jason\Desktop\vector_plot\mutural_interaction.png | Eq. |

The presence of the wing has a huge influence for the propeller on the drag coefficient, rolling, and pitching moment coefficient, which increased about , and , respectively. However, the aerodynamic performance of the propeller, such as lift, side force, yawing, rolling, and the pitching moment coefficients, are less significant as they are hundreds times smaller than the reference aerodynamic forces/moments, listed in Table 14 and Table 15. The thrust, on the other hand, indicates that the MAV is accelerating for the current propeller rotating speed. In other words, to achieve the steady state condition, we need to either reduce the propeller rotational speed or increase the angle of attack. It is also noticeable that a proper propeller is required for micro air vehicle as its aerodynamic performance may affect the overall stability (i.e. sensitive to the centre of gravity location), and the propeller efficiency also would affects on the overall MAV’s endurance.

The presence of the propeller has also shown a significant influence on the wing aerodynamics, which we named as the ‘propeller slipstream effects’. The mutual interaction aerodynamic forces/moments on the wing are listed in Table 14. The overall aerodynamic forces and moments for the MAV-propeller configuration are the summary of the forces/moments on both wing-fuselage and propeller, listed in Table 16.

Consequently, the overall lift coefficient of the MAV-propeller configuration at α = 0° with propeller running with an advance ratio of 0.23 is 0.382. The yawing moment coefficient is 3.26e-3, which implies a vertical tail is necessary to provide the directional stability.

Table 14 Mean aerodynamic coefficients for MAV with/without propeller installed at J = 0.23 (fixed lift coefficient, CL = 0.38)

|  |  |  |  |  |  |  |
| --- | --- | --- | --- | --- | --- | --- |
| configuration | CL, MAV | CD, MAV | CFz, MAV | CMx, MAV | CMy, MAV | CMz, MAV |
| MAV  *(without propeller)* | 0.38 | -0.0810 | 0 | 0 | 0 | 0.0256 |
| MAV  *(with propeller)* | 0.38 | -0.1817 | -0.0087 | 0.0169 | 0.0032 | 0.1623 |
| Mutual interaction  on wing due to  propeller |  | -0.1007 | -0.0087 | 0.0169 | 0.0032 | 0.1367 |

Table 15 Mean aerodynamic coefficients for the propeller with/without MAV installed at J = 0.23

|  |  |  |  |  |  |  |
| --- | --- | --- | --- | --- | --- | --- |
| configuration | CL, Prop | CD, Prop | CFz, Prop | CMx, Prop | CMy, Prop | CMz, Prop |
| Propeller  *(without MAV)* | 1.23e-5 | 0.83 | 1.28e-5 | -0.030 | -1.22e-5 | -9.8e-3 |
| Propeller  *(with MAV)* | 1.50e-3 | 1.036 | 1.3e-3 | -0.0352 | 6.083e-5 | -4.8e-3 |
| Mutual interaction  on propeller due to  wing |  | 0.206 |  | -5.2e-3 |  | -5e-3 |

Table Overall aerodynamic coefficients for MAV-fuselage-propeller at J = 0.23

|  |  |  |  |  |  |  |
| --- | --- | --- | --- | --- | --- | --- |
| configuration | CL, overall | CD, overall | CFz, overall | CMx, overall | CMy, overall | CMz, overall |
| MAV-propeller | 0.382 | 0.8543 | -7.4 e-3 | -0.0183 | 3.26e-3 | 0.1575 |

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| --- | --- |
| F:\Jason_chen\more_mesh(no_vertical_tail)\0deg_more_mesh_no_tail\C_{Fx-Prop}.emf   1. Propeller thrust coefficient | F:\Jason_chen\more_mesh(no_vertical_tail)\0deg_more_mesh_no_tail\C_{Fy-Prop}.emf   1. Propeller lift coefficient |
| F:\Jason_chen\more_mesh(no_vertical_tail)\0deg_more_mesh_no_tail\C_{Fz-Prop}.emf   1. Propeller side force coefficient | F:\Jason_chen\more_mesh(no_vertical_tail)\0deg_more_mesh_no_tail\C_{Mx-Prop}.emf   1. Propeller rolling moment coefficient |
| F:\Jason_chen\more_mesh(no_vertical_tail)\0deg_more_mesh_no_tail\C_{My-Prop}.emf   1. Propeller yawing moment coefficient | F:\Jason_chen\more_mesh(no_vertical_tail)\0deg_more_mesh_no_tail\C_{Mz-Prop}.emf   1. Propeller pitching moment coefficient |

Figure Blade force development at J = 0.23 (without vertical stabilizer)

#### Effect on flow structure

Flow visualization can give mostly qualitative information, about the flow in the vicinity of the propeller and can reveal areas of importance. Detailed flow structures can be interesting either in the wake, to reveal propeller related flow structures, or around the blades, to reveal the blade local flow information. Figure 83 and Figure 84 (a-i) show the asymmetric flow structure of the MAV-propeller model (without vertical stabilizer) for propeller at different azimuth angles, Ψb = 0° and 90° respectively with 0° angle of attack at a propeller advance ratio J = 0.23. In contrast, with the propeller slipstream effect, the wingtip vortex has been minimised as compared with the MAV without the propeller slipstream effects at 0° incidence, Ref. [112]. A clear asymmetric flow appears, which follows the aerofoil local inflow angle of attack evolution. Larger local angle of attack is formed on the upward-going blade side than on the downward-going blade side, Figure 83 and Figure 84 (e and g). Towards the wingtip, the local angle of attack on both side are about equal. Details are shown in Figure 83 and Figure 84 (f and h).

At the propeller azimuth angle Ψb = 0° (Figure 83 ), the leading edge separation on the upper wing surface is reduced due to the slipstream effect. The bubble is shortened and located at the wing root region, in Figure 83 (i). The low pressure region, as marked with dark blue colour, can be clearly seen on the upper surface, which indicates the asymmetric flow formed due to the propeller slipstream effects. A tiny area is occupied with the high pressure force on the down-going blade side near the leading edge indicating a high local velocity. On the lower wing surface, two separated and asymmetric bubbles are formed underneath the wing near the leading edge. The smaller one forms on the upward-going blade side and the larger one on the downward-going blade side. The shape of the bubble at different spanwise locations can be found in Figure 83 (e-f, g-h) and the lengths of the bubble are detailed in Figure 81. The area with relative larger pressure force is formed on the lower surface at the up-going blade side. This is due to the propeller tangential flow that impinges on the lower wing surface. However, on the down-going blade side, the tangential flow impinges on the upper surface.

At the propeller azimuth angle Ψb = 90°, Figure 84 (a-i), both pressure contour and the three-dimensional flow structures around the MAV model are shown. The adjacent helical filaments, in Figure 84 (a), pass over the centre upper wing surface and this mutual-interaction effect causes the adjacent spirals to roll-up around each other. Similar spirals can be also found in Figure 83 (a). The low-pressure region on both up- and down-going blade sides, in Figure 84 (a), is shifted a bit backwards as compared with the region on the up-going blade side with azimuth angle Ψb = 0° in Figure 83 (a). Another interesting point is that, a smaller separated region is formed on the fuselage at the downward-going side and a larger area is influenced on the fuselage at the upward-going side, (Figure 84 c and d). The lengths of the separation bubble at Ψb = 90° are more or less equal to the bubble length when Ψb = 0°, compared to Figure 81.

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| --- | --- |
| F:\Jason_chen\more_mesh(no_vertical_tail)\0deg_more_mesh_no_tail\MAV4_prop_2012_new2_3_more_mesh_0deg-1-12600_3D.jpg   1. Ψb = 0°(top view) | L:\Jason_chen\more_mesh\MAV4_prop_2012_new2_3_more_mesh_0deg-1-05400_side.jpg   1. Ψb = 0°(side view) |
| L:\Jason_chen\more_mesh\MAV4_prop_2012_new2_3_more_mesh_0deg-1-05400_9.jpg   1. Ψb = 0°(left view) | L:\Jason_chen\more_mesh\MAV4_prop_2012_new2_3_more_mesh_0deg-1-05400_8.jpg   1. Ψb = 0°(right view) |
| F:\Jason_chen\more_mesh(no_vertical_tail)\0deg_more_mesh_no_tail\MAV4_prop_2012_new2_3_more_mesh_0deg-1-12600_P005.jpg   1. 2z/b = 0.25(up-going blade ) | F:\Jason_chen\more_mesh(no_vertical_tail)\0deg_more_mesh_no_tail\MAV4_prop_2012_new2_3_more_mesh_0deg-1-12600_P01.jpg   1. 2z/b = 0.5 (up-going blade ) |
| F:\Jason_chen\more_mesh(no_vertical_tail)\0deg_more_mesh_no_tail\MAV4_prop_2012_new2_3_more_mesh_0deg-1-12600_N005.jpg   1. 2z/b = 0.25 (down-going blade ) | F:\Jason_chen\more_mesh(no_vertical_tail)\0deg_more_mesh_no_tail\MAV4_prop_2012_new2_3_more_mesh_0deg-1-12600_N01.jpg   1. 2z/b = 0.5 (down-going blade) |
| F:\Jason_chen\more_mesh(no_vertical_tail)\0deg_more_mesh_no_tail\MAV4_prop_2012_new2_3_more_mesh_0deg-1-12750_sym.jpg   1. Centre plane | |

Figure 83 flow structure around the MAV at α = 0°, Ψb = 0° (No vertical stabilizer)

|  |  |
| --- | --- |
| F:\Jason_chen\more_mesh(no_vertical_tail)\0deg_more_mesh_no_tail\MAV4_prop_2012_new2_3_more_mesh_0deg-1-11700_3D.jpg   1. Ψb = 90° (top view) | L:\Jason_chen\more_mesh\MAV4_prop_2012_new2_3_more_mesh_0deg-1-05580_side.jpg   1. Ψb = 90° (side view) |
| L:\Jason_chen\more_mesh\MAV4_prop_2012_new2_3_more_mesh_0deg-1-05580_9.jpg   1. Ψb = 90° (left view) | L:\Jason_chen\more_mesh\MAV4_prop_2012_new2_3_more_mesh_0deg-1-05580_8.jpg   1. Ψb = 90° (right view) |
| F:\Jason_chen\more_mesh(no_vertical_tail)\0deg_more_mesh_no_tail\MAV4_prop_2012_new2_3_more_mesh_0deg-1-11700_P005.jpg   1. 2z/b = 0.25(up-going blade ) | F:\Jason_chen\more_mesh(no_vertical_tail)\0deg_more_mesh_no_tail\MAV4_prop_2012_new2_3_more_mesh_0deg-1-11700_P01.jpg   1. 2z/b = 0.5(up-going blade ) |
| F:\Jason_chen\more_mesh(no_vertical_tail)\0deg_more_mesh_no_tail\MAV4_prop_2012_new2_3_more_mesh_0deg-1-11700_N005.jpg   1. 2z/b = 0.25 (down-going blade ) | F:\Jason_chen\more_mesh(no_vertical_tail)\0deg_more_mesh_no_tail\MAV4_prop_2012_new2_3_more_mesh_0deg-1-11700_N01.jpg   1. 2z/b = 0.5 (down-going blade ) |
| F:\Jason_chen\more_mesh(no_vertical_tail)\0deg_more_mesh_no_tail\MAV4_prop_2012_new2_3_more_mesh_0deg-1-11700_sym.jpg   1. Centre plane | |

Figure instantaneous flow structures at α = 0°, and Ψb = 90° (No vertical stabilizer)

### Propeller slipstream effect with vertical stabilizer (ω = 1131 rad/s)

In Section 5.3, we have discussed the aerodynamic performance and the stability due to the propeller effects. The MAV model in the previous section has no vertical stabilizer installed and unbalanced side forces have been generated from the propeller slipstream effect. Therefore, for the lateral stability purposes, a vertical stabilizer is installed and the geometry is shown in Figure 72 (c). This section is divided into two different subsections: 5.4.1 MAV aerodynamic performances and 5.4.2 flow structure around MAV at different propeller azimuth angle.

#### MAV Aerodynamic performance

The aerodynamic contribution, from each part: propeller, wing-fuselage, and the vertical stabilizer, have been plotted as a function of the angles of attack, and are shown in (a-f). The lift coefficient, one of the major parameters for the entire MAV model, (Figure 85 a), shows a maximum lift coefficient CL,max = 2.42 with an angle of attack about 36°. Such stall angle has been improved almost four times that of the isolate Zimmerman wing planform’s, (Figure 47 a). The main effect of the propeller slipstream on the wing at high incidence is on the turbulent separation of the boundary layer on the upper wing surface. The injection of momentum from the propeller slipstream delays turbulent separation. The vertical stabilizer, on the other hand, has a contribution on minimizing the spanwise flow as the propeller wake is highly turbulent and helical (without the vertical stabilizer, the CL,max = 1.7 at α = 24°, as shown in Figure 78 a). It is also noticeable that the propeller lift slope, CLα, Propeller, shows a linear behaviour with the incidences.

The drag coefficient, (Figure 85 b), indicates that the thrust coefficient (the dashed green line) has less influences from changing of the incidences and the drag coefficient formed on the vertical stabilizer is almost negligible. The intersection point between the two curves (CT,Propeller and -CD,MAV curves, Figure 85 (b)) indicates the tractor force can just overcome the drag force of the MAV. Namely, the balanced force in x-axis is achieved by setting the angle of attack at about 21.5°. However, the lift force is far more than the total weight for current propeller rotating speed of 1131 rad/s. The drag of the wing-fuselage at low and moderate incidences (with propeller on) is produced mainly by the increase of the pressure drag. The overall drag coefficient for the entire model, as marked by the black bold line, has clearly indicated that the acceleration motion on both forward and upward directions can be performed at. For, there may not have enough tractor force to overcome the drag force.

The side force coefficient, (Figure 85 c), indicates the lateral stabilities contribution from each component at various angles of attack. The propeller has little contribution on the side force coefficient and it has a linear trend with changing incidence. However, the main side force coefficient is produced by the wing-fuselage and the vertical-stabilizer. The side force from the MAV-fuselage (the red dashed line) generally increases with the incidence and a maximum side force is obtained at α = 24° (the negative sign indicates the opposite direction to the z-axis and a positive yawing moment will be produced). The vertical stabilizer, on the other hand, has a similar side force coefficient curve at α < 12° and a sudden increase occurred at α = 12°, the side force coefficient increases smoothly as the incidence further increases. The overall side force coefficient, the black solid line, shows a similar trend as compared with the vertical stabilizer curve.

The moment coefficients, shown in Figure 85 (d-f), are rolling, yawing, and pitching moments, respectively. Both propeller and the vertical stabilizer have a nearly constant rolling moment coefficients for the entire incidence range. A nearly zero rolling moment slope is shown. The rolling moment coefficient (for wing-fuselage part) shows a nearly constant value at incidence α < 18°, and it increased as the incidence increases.

The overall yawing moment coefficient decreases as the angle of attack is increased. A sudden dropping of the yawing moment coefficient at α = 18° has been observed, and a linear slope for α > 18° can be found in Figure 85 (e). The propeller shows a negative linear slope (). The yawing moment coefficient from the vertical stabilizer has shown a positive slope at and a negative linear slope occurs at.

The pitching moment, for overall performance, shows a positive slope at, and a negative slope occurred at. The propeller shows a linear positive slope, which indicates a negative contribution on the overall longitudinal static stability.

|  |  |
| --- | --- |
| M:\Jason_chen\MAV4_Tail\MAV4-Prop-Tail-n1131-CL2.emf   1. CL | M:\Jason_chen\MAV4_Tail\MAV4-Prop-Tail-n1131-CD2.emf   1. CD |
| M:\Jason_chen\MAV4_Tail\MAV4-Prop-Tail-n1131-CFz3.emf   1. CFz | M:\Jason_chen\MAV4_Tail\MAV4-Prop-Tail-n1131-CMx4.emf   1. CMx |
| F:\Jason_chen\MAV4_Tail\MAV4-Prop-Tail-n1131-CMy2.emf   1. CMy | M:\Jason_chen\MAV4_Tail\MAV4-Prop-Tail-n1131-CMz2.emf   1. CMz |

Figure Aerodynamic coefficients versus α at J = 0.23 (with vertical stabilizer)

As we have discussed in Section 5.3.1, similar Cp distributions, shown in Figure 86 and Figure 87, are obtained from the MAV-fuselage-propeller-tail model. For comparison purposes, we have set the advance ratio J = 0.23, the same as we tested in the previous sections used for the MAV-fuselage model. The Cp distributions are obtained from the spanwise locations at 2z/b = ± 0.25, ±0.5, where the positive sign means the up-going blade side and the negative sign indicates the down-going blade side. With the vertical stabilizer installed on the upper surface of the wing, the turning point locations have little changes as compared with the no vertical stabilizer model, (Figure 79). In general, the turning point locations are roughly at for inner spanwise location (2z/b = ±0.25) and shifts downstream to approximately at outer spanwise location (2z/b = ±0.5) for both models with/without vertical stabilizer. The unsynchronized turning point locations on the wing would affect the centre of pressure location, leading to the unbalanced moments and hence inducing the coupling problems (i.e. extra rolling, yawing moment coupling).

The unbalanced peak suction forces in the Cp plots in (a-d) are clearly visible. In contrast, similar suction peaks are found at inner spanwise location on the up-going blades side between models with/without vertical stabilizer, shown in Figure 79 (a) and Figure 80 (a), and similar Cp distribution on the down-going blade side also can be found. At the outer spanwise location, 2z/b = ±0.5, the negative lift generated from each side of the wing has a similar level, shown in (b and d). The fraction of positive and negative lift is clearly shown in the averaged Cp plot in Figure 87. In general, the Cp difference between model with/without the propeller is mainly due to the increase on both the positive and negative lift. Specially, attention should be paid to the negative lift as this indicates that less reflex camber can be applied for the propeller-driven MAV. Therefore, the camber designed for the isolated MAV-fuselage would no longer be suitable for the propeller-driven one.

The lengths of the separation bubble are also shown in the skin friction plot in Figure 88(a-d). The bubble with longer length (approximately ) is found underneath the wing on the down-going blade side and shorter one (approximately ) on the up-going blade side. Towards the wingtip, at 2z/b = 0.5, the bubble length on the up-going blade increases to. However, the bubble length on the down-going blade side has is reduced to.

|  |  |
| --- | --- |
| F:\Jason_chen\MAV4_Tail\0deg\Cp_2zb005_11070.emf   1. ψ = 0°, 2z/b = ± 0.25 | F:\Jason_chen\MAV4_Tail\0deg\Cp_2zb01_11070.emf   1. ψ = 0°, 2z/b = ± 0.5 |
| F:\Jason_chen\MAV4_Tail\0deg\Cp_2zb005_10890.emf   1. ψ = 90°, 2z/b = ± 0.25 | F:\Jason_chen\MAV4_Tail\0deg\Cp_2zb01_10890.emf   1. ψ = 90°, 2z/b = ± 0.5 |

Figure Cp distributions with J = 0.23 and α = 0° (with vertical stabilizer)

F:\Jason_chen\MAV4_Tail\0deg\MAV4_Prop_Tail_averaged_Cp.emf

Figure Mean Cp distribution at 2z/b = ± 0.5 with J = 0.23 and α = 0° (with vertical stabilizer)

|  |  |
| --- | --- |
| F:\Jason_chen\MAV4_Tail\0deg\Ctx_2zb005_11070.emf   1. ψ = 0°, 2z/b = ± 0.25 | F:\Jason_chen\MAV4_Tail\0deg\Ctx_2zb01_11070.emf   1. ψ = 0°, 2z/b = ± 0.5 |
| F:\Jason_chen\MAV4_Tail\0deg\Ctx_2zb005_10890.emf   1. ψ = 90°, 2z/b = ± 0.25 | F:\Jason_chen\MAV4_Tail\0deg\Ctx_2zb01_10890.emf   1. ψ = 90°, 2z/b = ± 0.5 |

Figure CTx distributions with J = 0.23 and α = 0° (with vertical stabilizer)

The period force, moment coefficient developments for the MAV model (including fuselage, propeller, and vertical stabilizer) of one revolution (360°) are shown in . Integration of the blade aerodynamic loads provides strong periodic fluctuations on the propeller forces projected in the absolute reference frame. The averaged aerodynamic forces and moments are separated and listed in and Table 18 for the propeller and MAV-fuselage-tail, respectively.

The presence of the vertical stabilizer has a impact on the propeller thrust coefficient mean value, as shown in , reduced by about 0.011 as compare with the non-vertical stabilizer model. The influence on the lift reduces about 4e-4. However, such lift contribution from the propeller is almost negligible as compare with the reference lift from the wing-fuselage-tail (CL, MAV = 0.384). The major influence is on the side force coefficient development with a reduced yawing moment coefficient (Table 18). In general, the presence of the vertical stabilizer increases the lift, side force coefficients and reduces the tractor, rolling, yawing, and pitching moments. Detailed force and moment improvements for α = 0° are shown in the following calculations:

a*WFVP* represents the wing, fuselage, vertical-stabilizer, and the propeller.

The following data show the aerodynamic changing on the propeller due to the vertical stabilizer installed:

In general, the overall aerodynamics for the whole model (which includes wing, fuselage, vertical stabilizer, and propeller) at an advance ratio J = 0.23 is analyzed (Table 18). The major changing due to the presence of the vertical stabilizer is on the side force and yawing moment coefficient. A positive yawing angle will produce a negative side force, and produces a negative yawing moment as according to the body-frame coordinate system. Therefore, for the configuration (i.e. J = 0.23), the model will have an acceleration motion with additional yawing, rolling moment coupled. In other words, this flight condition would not do a steady state motion. The steady state flight condition will be discussed in Section .

Table 17 Mean forces/moments coefficients for the propeller configurations (with/ without MAV) at J = 0.23, and α = 0°

|  |  |  |  |  |  |  |
| --- | --- | --- | --- | --- | --- | --- |
| configuration | CL, Prop | CD, Prop | CFz, Prop | CMx, Prop | CMy, Prop | CMz, Prop |
| Propeller-only | 1.23e-5 | 0.83 | 1.28e-5 | -0.03 | -1.22e-5 | -9.8e-3 |
| Propeller(WFP)a | 1.50e-3 | 1.036 | 1.30e-3 | -0.0352 | 6.083e-5 | -4.8e-3 |
| Propeller(WFVP)a | 1.90e-3 | 1.025 | 1.50e-3 | -0.0349 | -1.60 e-4 | -4.1 e-3 |

a*WFVP* represents the wing, fuselage, vertical-stabilizer, and the propeller.

Table 18 overall aerodynamic coefficients for different MAV configurations at J = 0.23, α = 0°

|  |  |  |  |  |  |  |
| --- | --- | --- | --- | --- | --- | --- |
| configuration | CL, overall | CD, overall | CFz, overall | CMx, overall | CMy, overall | CMz, overall |
| MAV-only | 0.38 | 0.0810 | 0 | 0 | 0 | 0.0256 |
| WFPa | 0.382 | 0.8543 | -0.0074 | -0.0183 | 3.26 e-3 | 0.1575 |
| WFVPa | 0.384 | 0.8436 | -0.0234 | -0.0142 | -8.4e-4 | 0.1468 |

|  |  |
| --- | --- |
| M:\Jason_chen\MAV4_Tail\0deg\MAV4_Propeller_Tail_Cfx_Prop.emf   1. Propeller thrust coefficient | L:\Jason_chen\MAV4_Tail\0deg\MAV4_Propeller_Tail_Cfy_Prop.emf   1. Propeller lift coefficient |
| M:\Jason_chen\MAV4_Tail\0deg\MAV4_Propeller_Tail_Cfz_Prop.emf   1. Propeller side force coefficient | M:\Jason_chen\MAV4_Tail\0deg\MAV4_Propeller_Tail_CMx_Prop.emf   1. Propeller rolling moment coefficient |
| F:\Jason_chen\MAV4_Tail\0deg\more_mesh\C_{My-Prop}.emf   1. Propeller yawing moment coefficient | F:\Jason_chen\MAV4_Tail\0deg\more_mesh\C_{Mz-Prop}.emf   1. Propeller pitching moment coefficient |
| F:\Jason_chen\MAV4_Tail\0deg\MAV4_Propeller_Tail_Cfx_MAV.emf   1. MAV drag coefficient | F:\Jason_chen\MAV4_Tail\0deg\MAV4_Propeller_Tail_Cfy_MAV.emf   1. MAV lift coefficient |
| F:\Jason_chen\MAV4_Tail\0deg\MAV4_Propeller_Tail_Cfz_MAV.emf   1. MAV,Tail side force coefficient | F:\Jason_chen\MAV4_Tail\0deg\MAV4_Propeller_Tail_CMx_MAV.emf   1. MAV,Tail rolling momemnt coefficient |
| F:\Jason_chen\MAV4_Tail\0deg\MAV4_Propeller_Tail_CMy_MAV.emf   1. MAV,Tail yawing moment coefficient | F:\Jason_chen\MAV4_Tail\0deg\MAV4_Propeller_Tail_CMz_MAV.emf   1. MAV,Tail pitching moment coefficient |

Figure 89 Blade force development at J = 0.23 (with vertical stabilizer)

#### Effect on flow structure

The relative flow structure around the MAV with the vertical stabilizer are shown in Figure 90 and Figure 91 with propeller azimuth angle Ψb = 0° and 90°, respectively, and the flow structure with various incidences are shown in Figure 92. The three-dimensional flow structure on the upper wing surface shows a similar flow structure as we discussed in the model without vertical stabilizer (Figure 83). However, the vertical stabilizer shows an effect of straightening the flow at the trailing edge region on the down-going blade side, (Figure 90 a). In other words, the helical vortex flow has more influences at trailing edge region on the up-going blade side of the wing.

Figure 92(a-f) shows the flow structure progressions on the upper wing surface at ω = 1131rad/s. The flow structure variations against the incidences can be clearly seen. Attached flow on the upper surface at low and moderate incidences roughly occurs at α < 12°. As the angle of attack increases the reversed flow starts to occur which can be found in the third column in Figure 92. A large vortical flow is found on the entire upper wing surface when the incidence α > 18°, and a large recirculation can be clearly identified near the vertical stabilizer region at the centre plane view in third column in Figure 92.

|  |  |
| --- | --- |
| F:\Jason_chen\MAV4_Tail\0deg\MAV4_prop_Tail_a_0deg-1-10710_3D.jpg   1. Ψb = 0°(top view) | L:\Jason_chen\MAV4_Tail\0deg\MAV4_prop_2012_new2_3_tail-1-03240_side2.jpg   1. Ψb = 0°(side view) |
| L:\Jason_chen\MAV4_Tail\0deg\MAV4_prop_2012_new2_3_tail-1-03240_8.jpg   1. Ψb = 0°(left view) | L:\Jason_chen\MAV4_Tail\0deg\MAV4_prop_2012_new2_3_tail-1-03240_9.jpg   1. Ψb = 0°(right view) |
| F:\Jason_chen\MAV4_Tail\0deg\MAV4_prop_Tail_a_0deg-1-11070_P005.jpg   1. 2z/b = 0.25(up-going blade ) | F:\Jason_chen\MAV4_Tail\0deg\MAV4_prop_Tail_a_0deg-1-11070_P01.jpg   1. 2z/b = 0.5 (up-going blade ) |
| F:\Jason_chen\MAV4_Tail\0deg\MAV4_prop_Tail_a_0deg-1-11070_N005.jpg   1. 2z/b = 0.25 (down-going blade ) | F:\Jason_chen\MAV4_Tail\0deg\MAV4_prop_Tail_a_0deg-1-11070_N01.jpg   1. 2z/b = 0.5 (down-going blade) |
| F:\Jason_chen\MAV4_Tail\0deg\MAV4_prop_Tail_a_0deg-1-3240_sym.jpg   1. Centre plane | |

Figure flow structure around the MAV at α = 0°, Ψb = 0° (with vertical stabilizer)

|  |  |
| --- | --- |
| F:\Jason_chen\MAV4_Tail\0deg\MAV4_prop_Tail_a_0deg-1-10890_3D.jpg   1. Ψb = 90°(top view) | L:\Jason_chen\MAV4_Tail\0deg\MAV4_prop_2012_new2_3_tail-1-03150_side2.jpg   1. Ψb = 90°(side view) |
| L:\Jason_chen\MAV4_Tail\0deg\MAV4_prop_2012_new2_3_tail-1-03150_8.jpg   1. Ψb = 90°(left view) | L:\Jason_chen\MAV4_Tail\0deg\MAV4_prop_2012_new2_3_tail-1-03150_9.jpg   1. Ψb = 90°(right view) |
| F:\Jason_chen\MAV4_Tail\0deg\MAV4_prop_Tail_a_0deg-1-11250_P005.jpg   1. 2z/b = 0.25(up-going blade ) | F:\Jason_chen\MAV4_Tail\0deg\MAV4_prop_Tail_a_0deg-1-11250_P01.jpg   1. 2z/b = 0.5 (up-going blade ) |
| F:\Jason_chen\MAV4_Tail\0deg\MAV4_prop_Tail_a_0deg-1-11250_N005.jpg   1. 2z/b = 0.25 (down-going blade ) | F:\Jason_chen\MAV4_Tail\0deg\MAV4_prop_Tail_a_0deg-1-11250_N01.jpg   1. 2z/b = 0.5 (down-going blade) |
| F:\Jason_chen\MAV4_Tail\0deg\MAV4_prop_Tail_a_0deg-1-3150_sym.jpg   1. Centre plane | |

Figure flow structure around the MAV at α = 0°, Ψb = 90° (with vertical stabilizer)

|  |  |  |
| --- | --- | --- |
| F:\Jason_chen\MAV4_Tail\N6deg\MAV4_prop_Tail_a_N6deg-1-04530_5.jpg   1. α = -6° | M:\Jason_chen\MAV4_Tail\N6deg\MAV4_prop_Tail_a_N6deg-1-03360_5.jpg | F:\Jason_chen\MAV4_Tail\N6deg\MAV4_prop_Tail_a_N6deg-1-04500_2d_symmetry2.jpg |
| M:\Jason_chen\MAV4_Tail\0deg\MAV4_prop_Tail_a_0deg-1-10800_4.jpg   1. α = 0° | M:\Jason_chen\MAV4_Tail\0deg\MAV4_prop_Tail_a_0deg-1-10980_4.jpg | F:\Jason_chen\MAV4_Tail\0deg\MAV4_prop_Tail_a_0deg-1-11070_2d_symmetry2.jpg |
| M:\Jason_chen\MAV4_Tail\6deg\MAV4_prop_Tail_a_6deg-1-08190_4.jpg   1. α = 6° | M:\Jason_chen\MAV4_Tail\6deg\MAV4_prop_Tail_a_6deg-1-09060_4.jpg | F:\Jason_chen\MAV4_Tail\6deg\MAV4_prop_Tail_a_6deg-1-12780_2d_symmetry2.jpg |
| M:\Jason_chen\MAV4_Tail\12deg\MAV4_prop_Tail_a_12deg-1-07080_5.jpg   1. α = 12° | M:\Jason_chen\MAV4_Tail\12deg\MAV4_prop_Tail_a_12deg-1-07140_5.jpg | F:\Jason_chen\MAV4_Tail\12deg\MAV4_prop_Tail_a_12deg-1-22770_2d_symmetry2.jpg |
| M:\Jason_chen\MAV4_Tail\18deg\MAV4_prop_Tail_a_18deg-1-011220_5.jpg   1. α = 18° | M:\Jason_chen\MAV4_Tail\18deg\MAV4_prop_Tail_a_18deg-1-011100_5.jpg | F:\Jason_chen\MAV4_Tail\18deg\MAV4_prop_Tail_a_18deg-1-19170_2d_symmetry2.jpg |
| M:\Jason_chen\MAV4_Tail\24deg\MAV4_prop_Tail_a_24deg-1-017880_5.jpg   1. α = 24° | M:\Jason_chen\MAV4_Tail\24deg\MAV4_prop_Tail_a_24deg-1-017400_5.jpg | F:\Jason_chen\MAV4_Tail\24deg\MAV4_prop_Tail_a_24deg-1-31200_2d_symmetry2.jpg |

Figure flow structure with propeller slipstream effect at ω = 1131rad/s

### Propeller slipstream effect on MAV model at design condition with vertical stabilizer (ω = 555 rad/s)

shows the numerical aerodynamic force residuals versus the blade azimuth angles. A full rotation is equivalent to 360° blade azimuth angle. Therefore, 4000° will be about 11 rotations. It shows that periodic pulses are produced. This type of signature is found to be relatively independent of the advance ratio and appeared to be mainly associated with the local loading on the propeller itself. Mean force values are calculated from the converged periodic iteration (i.e. from azimuth angle of 3000° to 4000°).

A steady state flight condition is defined as one for which all motion variables remain constant with time relative to the body-fixed axis system XYZ. Mathematically speaking, steady state flight conditions are shown as:

|  |  |
| --- | --- |
|  | Eq. |
|  | Eq. |

Eq. 94 implies that the MAV does not have any acceleration in any direction (i.e.), and Eq. 95 indicates that the roll, yaw, and pitch rates are zero

In fact, the main influence of the propeller slipstream appears to concern the tractor/drag components in x-direction, as shown in Figure 94. Both tractor and drag coefficients decrease as the advanced ratio increases. At an advance ratio J = 0.468, the tractor coefficient of the propeller can just balance the drag of the MAV. As advance ratio increases to J = 0.52, the MAV drag coefficient is larger than the tractor coefficient, which means there is not enough thrust to drive the MAV.

|  |  |
| --- | --- |
| F:\Jason_chen\MAV4_Tail\n555\forces_residual_data\C_{Fx-mav}0deg.emf   1. MAV drag coefficient | F:\Jason_chen\MAV4_Tail\n555\forces_residual_data\C_{Fy-mav}0deg.emf   1. MAV lift coefficient |
| F:\Jason_chen\MAV4_Tail\n555\forces_residual_data\C_{Fz-mav}0deg.emf   1. MAV side force coefficient | F:\Jason_chen\MAV4_Tail\n555\forces_residual_data\C_{Mx-mav}0deg.emf   1. MAV rolling moment coefficient |
| F:\Jason_chen\MAV4_Tail\n555\forces_residual_data\C_{My-mav}0deg.emf   1. MAV yawing moment coefficient | F:\Jason_chen\MAV4_Tail\n555\forces_residual_data\C_{Mz-mav}0deg.emf   1. MAV pitching moment coefficient |
| F:\Jason_chen\MAV4_Tail\n555\forces_residual_data\C_{Fx-Prop}0deg.emf   1. Propeller drag coefficient | F:\Jason_chen\MAV4_Tail\n555\forces_residual_data\C_{Fy-Prop}0deg.emf   1. Propeller lift coefficient |
| F:\Jason_chen\MAV4_Tail\n555\forces_residual_data\C_{Fz-Prop}0deg.emf   1. Propeller side force coefficient | F:\Jason_chen\MAV4_Tail\n555\forces_residual_data\C_{Mx-Prop}0deg.emf   1. Propeller rolling moment coefficient |
| F:\Jason_chen\MAV4_Tail\n555\forces_residual_data\C_{My-Prop}0deg.emf   1. Propeller yawing moment coefficient | F:\Jason_chen\MAV4_Tail\n555\forces_residual_data\C_{Mz-Prop}0deg.emf   1. Propeller pitching moment coefficient |
| F:\Jason_chen\MAV4_Tail\n555\forces_residual_data\C_{Fx-tail}0deg.emf   1. Tail drag coefficient | F:\Jason_chen\MAV4_Tail\n555\forces_residual_data\C_{Fy-tail}0deg.emf   1. Tail lift coefficient |
| F:\Jason_chen\MAV4_Tail\n555\forces_residual_data\C_{Fz-tail}0deg.emf   1. Tail side force coefficient | F:\Jason_chen\MAV4_Tail\n555\forces_residual_data\C_{Mx-tail}0deg.emf   1. Tail rolling moment coefficient |
| F:\Jason_chen\MAV4_Tail\n555\forces_residual_data\C_{My-tail}0deg.emf   1. Tail yawing moment coefficient | F:\Jason_chen\MAV4_Tail\n555\forces_residual_data\C_{Mz-tail}0deg.emf   1. Tail pitching moment coefficient |

Figure force history at α = 0° with J = 0.468

J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV4_zim\MAV4_propeller\MAV4_Prop_Drag_Tractor_Force.emf

Figure Drag and Tractor coefficient with a function of advance ratio

The equivalent rotational speed for the steady state condition shown in Figure 94 is 555rad/s, for which the steady state flight condition is around zero angle of attack, Figure 98 ( a and b). The lift force at α = 0° is 1.55N and the overall force in the flight direction is in equilibrium (in the x-direction, T = D).

The relevant pressure distribution at this particular flight condition is shown in (a-d) and the averaged Cp in Figure 95 (e). As the rotational speed reduces from 1131 to 555 rad/s, the amount of the positive lift at positive camber location shows less difference on up- and down-going blade sides except at the leading edge region. The averaged Cp at 2z/b = 0.5 in Figure 95 (e) shows that a similar amount of lift is produced by the isolated MAV model and the MAV -propeller model. However, the negative lift generated at the reflex camber region by the MAV-propeller model is quite significant, and more negative lift is produced due to the propeller slipstream effect. In contrast, the amount of negative lift almost doubles as compared with the isolated MAV’s (dashed line). Therefore, less reflex camber can be used for the MAV with the propeller installed.

The skin friction coefficient plots in the x-direction are shown in Figure 96, where the separation bubble length underneath the wing can be clearly observed. At the inner spanwise location, 2z/b = ±0.25, the difference of bubble length on the up-going blade side can be observed, which is approximately and for ψ = 0° and 90°, respectively. However, bubbles with nearly equal length are found on the down-going blade side for different propeller azimuth angles. At outer spanwise location, 2z/b = ±0.5, no differences on the bubble lengths due to the propeller azimuth angles are observed. However, the bubble length differences on wing sides can be seen (roughly and for up- and down-going blade sides).

Figure 97 shows the two dimensional lift and drag coefficient distribution at various spanwise locations. The difference of the lift coefficients distribution for both wing sides is clearly shown. The drag coefficient, on the other hand, has shown a higher value on the down-going blade and a slightly lowered value on the up-going blade. The possible reason could be: in the up-going blade region, the propeller swirl counteracts the effects of the wing downwash such that the local angles of attack are increased. This effect simultaneously augments the section lift and rotates the force vector forwards, which reduces the drag component at the section, as detailed in Figure 77. The sudden increase in lift or decrease in drag coefficient indicate the aerodynamic interaction on the wing and the fuselage from the propeller wake. In general, the major slipstream effect are working at 60% inboard wingspan, , and less propeller slipstream effects has been observed towards the wingtip.

|  |  |
| --- | --- |
| F:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\Cp_2zb005_12060.emf   1. ψ = 0°, 2z/b = 0.25 | F:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\Cp_2zb01_12060.emf   1. ψ = 0°, 2z/b = 0.5 |
| F:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\Cp_2zb005_12240.emf   1. ψ = 90°, 2z/b = 0.25 | F:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\Cp_2zb01_12240.emf   1. ψ = 90°, 2z/b = 0.5 |
| F:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\MAV4_prop_2012_tail_more_mesh_n555_averaged_Cp.emf   1. averaged Cp distribution at 2z/b = ±0.5 | |

Figure Cp distributions with J = 0.468 and α = 0° (i.e. steady state flight condition)

|  |  |
| --- | --- |
| F:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\CTx_2zb005_12240.emf   1. ψ = 0°, 2z/b = 0.25 | F:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\CTx_2zb01_12240.emf   1. ψ = 0°, 2z/b = 0.5 |
| F:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\CTx_2zb005_12060.emf   1. ψ = 90°, 2z/b = 0.25 | F:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\CTx_2zb01_12060.emf   1. ψ = 90°, 2z/b = 0.5 |

Figure CTx distributions with J = 0.468 and α = 0° (i.e. steady state flight condition)

|  |  |
| --- | --- |
| M:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\MAV4_prop_2d_cl_12150_0deg.emf   1. ψ = 0° (C*l*) | M:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\MAV4_prop_2d_cd_12150_0deg.emf  ψ = 0° (Cd) |
| M:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\MAV4_prop_2d_cl_12240_45deg.emf   1. ψ = 45° (C*l*) | M:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\MAV4_prop_2d_cd_12240_45deg.emf  ψ = 45° (Cd) |
| M:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\MAV4_prop_2d_cl_12690_90deg.emf   1. ψ = 90° (C*l*) | M:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\MAV4_prop_2d_cd_12690_90deg.emf  ψ = 90° (Cd) |
| M:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\MAV4_prop_2d_cl_11940_255deg.emf   1. ψ = 135° (C*l*) | M:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\MAV4_prop_2d_cd_11940_255deg.emf  ψ = 135° (Cd) |

Figure two dimensional lift/drag distribution with α = 0°, ω = 555rad/s

#### MAV aerodynamic performance and stabilities

The aerodynamic coefficients are shown in (a-f) with a relevant propeller rotational speed of 555rad/s. The forces/moments for different components are also clearly shown in the plots individually (i.e. Propeller, wing-fuselage, vertical-stabilizer). The lift coefficient in (a) shows a linear behaviour, CLα, at incidence α < 6° a nonlinear CLα can be observed for α >6°. In contrast, the large difference on the lift coefficient is due to the propeller rotational speed has been reduced, which can be found in . The maximum lift coefficient, CL,max, has been found about 2.4 at 36° angle of attack with propeller rotational speed of 1131rad/s (i.e. J = 0.23). However, the CL,max has been reduced to about 0.9 at incidence of 18° with a lower rotational speed of 555rad/s (i.e. J = 0.468). The main reason is because the propeller has a very small contribution on the lift force at low angles of attack (i.e. the lift slope of the propeller: CLα, propeller = 0.003 at and CLα, propeller = 0.02 at). The propeller has no contribution on the lift coefficient at α = 0° for both rotational speeds. The tail shows a negligible contribution on the lift coefficient. Therefore, at high advance ratio, J or low rotational speed, the lift coefficient is mainly from the wing at low and moderate angles of attack. The lift contributions from the propeller increases with the rotational speed increases (i.e. the propeller slope improves significantly), and the difference can be found in Figure 85 (a) and Figure 98 (a).

The drag coefficient in Figure 98 (b), on the other hand, shows similar trends as compared with the high propeller rotational case in Figure 85 (b). The drag increases continuously with the angle of attack when the rotational speed is fixed at 555 rad/s. A nearly constant thrust has been produced by the propeller with various angles of attack, CT is roughly around 0.09 and the negative sign represents the incoming wind direction. However, at the high rotational speed, as shown in Figure 85 (b), the propeller generates a thrust level of CT around 1 at α < 24° and reduced to around 0.9 at higher incidences. The thrust level difference between two rotational speeds (i.e.) is about two times. The main drag force is formed from the wing-fuselage and higher drag slope CDα is produced by the lower the propeller rotational speed.

Figure 98 (c) shows the side force generated by the MAV model. In general, the overall side force coefficient, the solid black line, increases as the angle of attack increases. At 0°, the overall side force coefficient is about -0.00223 (or -0.00863N) which is negligible. However, the maximum side force coefficient is -0.0073 at 36°, which is equal to about -0.0282N (roughly about 3 times higher than that at α = 0°). At 0°, the main side force is generated by the wing-fuselage body, approximately -0.0037 (or -0.0143N), the propeller generates a positive side force coefficient of 0.0016 (or 0.0062N), and the side force from the vertical stabilizer is negligible at incidence = 0˚. In general, the side force has a significant difference between the low and high incidences.

The rolling, yawing, and pitching moments are shown in Figure 98 (d-f) respectively. A continuous increase of the rolling moment coefficient is shown in Figure 98 (d). At 0°, the overall rolling moment coefficient is -0.0014 (or -0.0024Nm). The peak value is found at 18° mainly from the wing-fuselage part. The forces/moments history convergence for each components (i.e. propeller, wing-fuselage, and vertical stabilizer) at α = 18° in Figure 99 indicates that the full convergence has been achieved at high incidences. The yawing moment coefficient shows that this MAV with a propeller that spins clockwise as viewed from the rear, the moments cause a left yaw. However, it has a very small value at low incidence, α = 0°, approximately 0.00009 (i.e. it can be ignored). A linear increased of the propeller yawing moment can be found from the plot.

The pitching moment, as another important factor, has a natural statically longitudinal stable region at angle of attack α < 0°, and positive contribution on the longitudinal stability side as the incidence increases. The pitching moment coefficient slope is more linear than the wing-fuselage model’s found in Figure 60 (c). In contrast, the pitching moment slope, CMα, without the propeller included is roughly about -0.0037 (for the linear section at incidence between 6° and 36°), whereas the slope CMα is about -0.0033. The propeller has shown a negative contribution on the statically longitudinal stability and a positive pitching moment slope can be identified in Figure 98 (f).

|  |  |
| --- | --- |
| L:\Jason_chen\MAV4_Tail\MAV4-Prop-Tail-n555-CL2.emf   1. CL | L:\Jason_chen\MAV4_Tail\MAV4-Prop-Tail-n555-CD2.emf   1. CD |
| L:\Jason_chen\MAV4_Tail\MAV4-Prop-Tail-n555-CFz3.emf   1. CFz | L:\Jason_chen\MAV4_Tail\MAV4-Prop-Tail-n555-CMx4.emf   1. CMx |
| F:\Jason_chen\MAV4_Tail\MAV4-Prop-Tail-n555-CMy2.emf   1. CMy | L:\Jason_chen\MAV4_Tail\MAV4-Prop-Tail-n555-CMz2.emf   1. CMz |

Figure Aerodynamic coefficients versus α at J = 0.468 (vertical stabilizer installed)

|  |  |
| --- | --- |
| F:\Jason_chen\MAV4_Tail\n555\forces_residual_data\C_{Fy-mav}18deg.emf   1. CL,MAV | F:\Jason_chen\MAV4_Tail\n555\forces_residual_data\C_{Fx-mav}18deg.emf   1. CD,MAV |
| F:\Jason_chen\MAV4_Tail\n555\forces_residual_data\C_{Fz-mav}18deg.emf   1. CFz,MAV | F:\Jason_chen\MAV4_Tail\n555\forces_residual_data\C_{Mx-mav}18deg.emf   1. CMx,MAV |
| F:\Jason_chen\MAV4_Tail\n555\forces_residual_data\C_{My-mav}18deg.emf   1. CMy,MAV | F:\Jason_chen\MAV4_Tail\n555\forces_residual_data\C_{Mz-mav}18deg.emf   1. CMz,MAV |
| F:\Jason_chen\MAV4_Tail\n555\forces_residual_data\C_{Fy-Prop}18deg.emf   1. CL,Propeller | F:\Jason_chen\MAV4_Tail\n555\forces_residual_data\C_{Fx-Prop}18deg.emf   1. CD,Propeller |
| F:\Jason_chen\MAV4_Tail\n555\forces_residual_data\C_{Fz-Prop}18deg.emf   1. CFz, Propeller | F:\Jason_chen\MAV4_Tail\n555\forces_residual_data\C_{Mx-Prop}18deg.emf   1. CMx, Propeller |
| F:\Jason_chen\MAV4_Tail\n555\forces_residual_data\C_{My-Prop}18deg.emf   1. CMy, Propeller | F:\Jason_chen\MAV4_Tail\n555\forces_residual_data\C_{Mz-Prop}18deg.emf   1. CMz, Propeller |
| F:\Jason_chen\MAV4_Tail\n555\forces_residual_data\C_{Fy-tail}18deg.emf   1. CL,Tail | F:\Jason_chen\MAV4_Tail\n555\forces_residual_data\C_{Fx-tail}18deg.emf   1. CD,Tail |
| F:\Jason_chen\MAV4_Tail\n555\forces_residual_data\C_{Fz-tail}18deg.emf   1. CFz,Tail | F:\Jason_chen\MAV4_Tail\n555\forces_residual_data\C_{Mx-tail}18deg.emf   1. CMx,Tail |
| F:\Jason_chen\MAV4_Tail\n555\forces_residual_data\C_{My-tail}18deg.emf   1. CMy,Tail | F:\Jason_chen\MAV4_Tail\n555\forces_residual_data\C_{Mz-tail}18deg.emf   1. CMz,Tail |

Figure Force history for different components at α = 18°with rotational speed of 555rad/s

**Static longitudinal stability**

MAV stability is of paramount important. As they are small, the margin for error in stability is small. Figure 100 shows the moment balance for the current MAV with setting the propeller rotational speed at 555 rad/s. The thrust force is along the drag force line (i.e. same height away from the centre of gravity). The moment is taken at centre of gravity with positive in anticlockwise. The aerodynamic centre location for the Zimmerman wing MAV is about the quarter mean chord, and this are discussed in the previous chapter (in Chapter 4, subsection 4.6). The stability margin details for wing-fuselage models (i.e. for rectangular, trapezoid wings, inversed-Zimmerman, and Zimmerman models) are listed in Table 11. To have the longitudinal static stability, the MAV pitching moment curve must have a negative slope, (). In order to trim at a positive angle of attack, the MAV must have a positive intercept, (). Both MAV-fuselage model and MAV-propeller (at design condition, ) satisfy the conditions (negative pitching moment slope can be found in (c) and (f) for model with/without propeller, respectively). The balance of pitching moment is calculated from and the longitudinal static stability margins are listed in Table 19. It is clearly shown that with the propeller installed, the static longitudinal stability margin is increased to almost twice as large as the wing-fuselage model ( = 15.5% and 8.53%, respectively).

Table 19 Stability margin for both wing-fuselage model and MAV-propeller (rotational speed fixed at 555rad/s) at steady state flight condition

|  |  |  |  |
| --- | --- | --- | --- |
| Case |  |  |  |
| Wing-fuselage | 0.0853 | 0.0323 | -0.0104 |
| MAV-Propeller | 0.155 | 0.0599 | -0.015 |

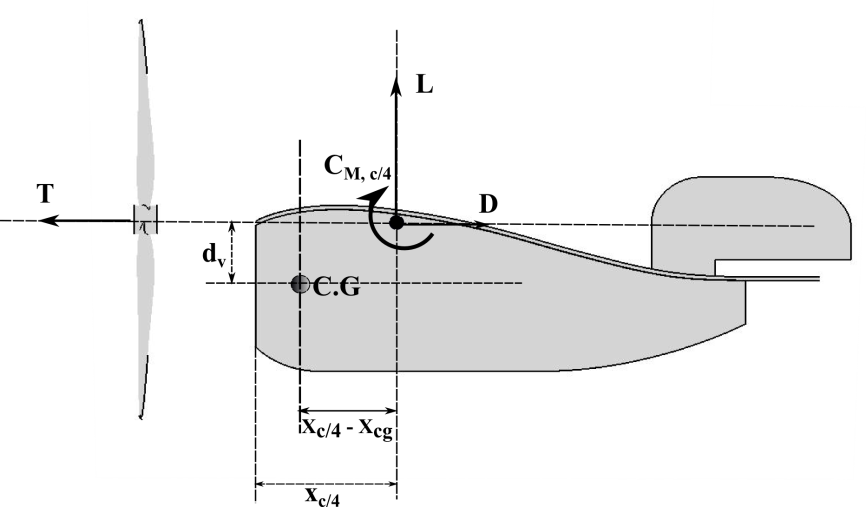


Figure 100 MAV moment balance at steady stat flight condition

**Static lateral stability**

Static lateral stability is concerned with the static stability of the MAV about the y-axis (for current model, x is the rolling axis, y is the yawing axis, and z is pitching axis). Just like the case of static longitudinal stability, it is desirable that the MAV should tend to turn to an equilibrium condition when suffered a yawing disturbance. The roll moment coefficient CMx, side force CFz, and yaw moment CMy as well as the sideslip angle β. Positive sideslip angle β is defined as the node going to the right. To have a static directional stability, the MAV must develop a yawing moment which will restore the MAV to its equilibrium state. Therefore, to have static directional stability, the MAV must have a positive yawing moment slope, (i.e. , most commonly in the text books it written as ). Note that an aircraft possessing static directional stability will always points into the relative wind, thus also called as “weathercock stability”. Another parameter to analyze the static lateral stability is the rolling moment,, and the requirement for stability is that . Namely, the net result is a rolling moment that tries to bring the MAV back to wing-level attitude. The wing planform usually has a negative contribution on the static lateral stability. The fuselage and the vertical stabilizer, on the other hand, it has a positive contribution on the static lateral stability.

Figure 101 and Figure 102 show the static lateral stability for the MAV model without/with the vertical stabilizer at zero incidence (α = 0°), respectively. In general, MAV without vertical stabilizer is statically unstable in lateral (yawing moment slope in Figure 101 b). Although, the rolling moment is satisfied the lateral stability condition (the rolling moment slope ). This can be found in Figure 101 (a). At β = 0º, the MAV without the vertical stabilizer shows a negative yawing moment coefficient (), indicating a turning towards left. This is because the drag force (due to the separation) on the fuselage side (up-going blade side) is larger than that on the fuselage at down-going blade side, and hence produces an extra yawing moment. Namely, additional yawing control is required from the pilot.

The MAV with the vertical stabilizer (Figure 102), on the other hand, is statically laterally stable. Both the rolling and yawing moments satisfy the lateral stability conditions, which are roughly about and , respectively. At β = 0º, the yawing moment is nearly zero, indicating an balanced moment on yaw.

|  |  |
| --- | --- |
| G:\Jason_chen\more_mesh(no_vertical_tail)\side_angle\C_{Mx-overall}.emf   1. Rolling moment coefficient | M:\Jason_chen\more_mesh(no_vertical_tail)\side_angle\C_{My-overall}.emf   1. Yawing moment coefficient |

Figure yaw angles (β°) effects on MAV without vertical stabilizer at α= 0°

|  |  |
| --- | --- |
| G:\Jason_chen\MAV4_Tail\n555\side_angle\more_mesh\C_{Mx-overall}2.emf   1. Rolling moment coefficient | G:\Jason_chen\MAV4_Tail\n555\side_angle\more_mesh\C_{My-overall}2.emf   1. Yawing moment coefficient |

Figure yaw angles (β°) effects on MAV with vertical stabilizer at α= 0°

#### Effect on flow structure

The detailed visualization of complex flow structures around MAV at various propeller azimuth angles are in Figure 103 and Figure 104, whereas the flow structures with changing in angles of attack are shown in Figure 105. The plotted streamlines reside close to the surface, typically within the boundary layer. At propeller azimuth angle ψ = 0°, in Figure 103, the flow is fully attached on the upper surface of the wing except the centre of the wing (i.e. wing root). The flow separated at near leading edge region and the two-dimensional bubble shape is shown in Figure 103 (i) and the three-dimensional bubble can be observed in Figure 103 (a and b).

At incidence, a high-pressure region is located close to the leading edge, corresponding to flow separation. This is followed by a pressure recovery (negative-pressure or suction force), located roughly at the maximum positive camber section. In general, pressure recovery is followed by a mild adverse pressure gradient, which is not strong enough to cause the flow to separate at. This separated area has been enlarged in spanwise as the angle of attack increased to. Symmetrical separated region has been identified by the pressure contour at leading edge area, whereas asymmetric pressure distribution is formed at negative camber location. A slightly higher pressure contour (i.e. more downward or negative lift force) can be seen on the down-going blade side. The size of the leading edge vortex increases largely with increasing α at under the propeller slipstream effect. Whereas, Okamoto [93] has mentioned that the leading edge vortex has increased a little with increasing α when the propeller is not installed.

Flow structures over the upper wing surface at incidence 6° are given in Figure 105. At this higher angle of attack, the adverse pressure gradient is too strong for the low Reynolds number flow to overcome, and a large separated flow is dominated at negative camber location. The leading edge vortex, as on the other hand, has been increased as marked with the blue pressure contour. At incidence, the separated region at negative camber location is enlarged, which however a symmetrical flow structure still can be observed. Although an asymmetric pressure contour has been shown at this moderate incidence, especially at negative camber location. The leading edge vortex structure is strengthened as compared with the lower incidences. The roll-up vortices on the upper surface of the wing creates the downwash effects from the tip to the root to change the boundary layer and force the flow field at spanwise direction to remain attached at high angle of attack. However, the propeller slipstream provides downwash effects on the down-going blade side and upwash effects on the up-going blade side.

In general, the flow structure progress at low propeller rotational speed (ω = 555rad/s) can be classified into three statuses: for, attached flow is dominated on the upper wing surface with a symmetrical leading edge separation region accompanied. For, symmetrical vortical flow is dominated at negative camber location on the upper surface, and a strengthened leading edge vortex has been observed.

|  |  |
| --- | --- |
| F:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\MAV4_prop_2012_new2_3_tail_more_mesh_n555-1-12240_3D.jpg   1. Ψb = 0°(top view) | F:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\MAV4_prop_2012_new2_3_tail_more_mesh_n555-1-12240_side_view.jpg   1. Ψb = 0°(side view) |
| F:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\MAV4_prop_2012_new2_3_tail_more_mesh_n555-1-12240_9.jpg   1. Ψb = 0°(left view) | F:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\MAV4_prop_2012_new2_3_tail_more_mesh_n555-1-12240_8.jpg   1. Ψb = 0°(right view) |
| F:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\MAV4_prop_2012_new2_3_tail_more_mesh_n555-1-12240_P005.jpg   1. 2z/b = 0.25(up-going blade ) | F:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\MAV4_prop_2012_new2_3_tail_more_mesh_n555-1-12240_P01.jpg   1. 2z/b = 0.5 (up-going blade ) |
| F:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\MAV4_prop_2012_new2_3_tail_more_mesh_n555-1-12240_N005.jpg   1. 2z/b = 0.25 (down-going blade ) | F:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\MAV4_prop_2012_new2_3_tail_more_mesh_n555-1-12240_N01.jpg   1. 2z/b = 0.5 (down-going blade) |
| F:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\MAV4_prop_2012_new2_3_tail_more_mesh_n555-1-12240_sym.jpg   1. Centre plane | |

Figure flow structure around the MAV at α = 0°, Ψb = 0° (with vertical stabilizer)

|  |  |
| --- | --- |
| F:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\MAV4_prop_2012_new2_3_tail_more_mesh_n555-1-12060_3D.jpg   1. Ψb = 90°(top view) | F:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\MAV4_prop_2012_new2_3_tail_more_mesh_n555-1-12060_side_view.jpg   1. Ψb = 90°(side view) |
| F:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\MAV4_prop_2012_new2_3_tail_more_mesh_n555-1-12060_9.jpg   1. Ψb = 90°(left view) | F:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\MAV4_prop_2012_new2_3_tail_more_mesh_n555-1-12060_8.jpg   1. Ψb = 90°(right view) |
| F:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\MAV4_prop_2012_new2_3_tail_more_mesh_n555-1-12060_P005.jpg   1. 2z/b = 0.25(up-going blade ) | F:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\MAV4_prop_2012_new2_3_tail_more_mesh_n555-1-12060_P01.jpg   1. 2z/b = 0.5 (up-going blade ) |
| F:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\MAV4_prop_2012_new2_3_tail_more_mesh_n555-1-12060_N005.jpg   1. 2z/b = 0.25 (down-going blade ) | F:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\MAV4_prop_2012_new2_3_tail_more_mesh_n555-1-12060_N01.jpg   1. 2z/b = 0.5 (down-going blade) |
| F:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\MAV4_prop_2012_new2_3_tail_more_mesh_n555-1-12060_sym.jpg   1. Centre plane | |

Figure flow structure around the MAV at α = 0°, Ψb = 90° (with vertical stabilizer)

|  |  |  |
| --- | --- | --- |
| F:\Jason_chen\MAV4_Tail\n555\N6deg\MAV4_prop_Tail_n555_N6deg-1-13320_5.jpg  α = -6° | F:\Jason_chen\MAV4_Tail\n555\N6deg\MAV4_prop_Tail_n555_N6deg-1-13080_5.jpg | F:\Jason_chen\MAV4_Tail\n555\N6deg\MAV4_prop_Tail_n555_N6deg-1-13260_2d_symmetry2.jpg |
| F:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\MAV4_prop_2012_new2_3_tail_more_mesh_n555-1-11970_5.jpg  α = 0° | F:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\MAV4_prop_2012_new2_3_tail_more_mesh_n555-1-11730_5.jpg | F:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\MAV4_prop_2012_new2_3_tail_more_mesh_n555-1-12240_2d_symmetry2.jpg |
| M:\Jason_chen\MAV4_Tail\n555\6deg\MAV4_prop_Tail_n555_6deg-1-25200_5.jpg  α = 6° | M:\Jason_chen\MAV4_Tail\n555\6deg\MAV4_prop_Tail_n555_6deg-1-24960_4.jpg | F:\Jason_chen\MAV4_Tail\n555\6deg\MAV4_prop_Tail_n555_6deg-1-25140_2d_symmetry2.jpg |
| M:\Jason_chen\MAV4_Tail\n555\12deg\MAV4_prop_Tail_n555_12deg-1-23400_4.jpg  α = 12° | M:\Jason_chen\MAV4_Tail\n555\12deg\MAV4_prop_Tail_n555_12deg-1-23160_4.jpg | F:\Jason_chen\MAV4_Tail\n555\12deg\MAV4_prop_Tail_n555_12deg-1-24360_2d_symmetry2.jpg |

Figure flow structure with propeller slipstream effect at ω = 555rad/s

#### Vortex shedding due to propeller slipstream effect

The turbulent kinetic energy, k, can be considered to determine the velocity scales for propeller slipstream flow. The turbulent kinetic energy, k, is defined as:

The turbulent kinetic energy can be regarded as a measure for ‘turbulence conservation’ and is useful in a description of time-averaged datasets. For the current study, the turbulent kinetic energy distribution at the symmetry plane for different azimuth angles (ψ°) are shown in (a-f). As mentioned in the previous subsection , Figure 68 and Figure 69 show transition regions for Zimmerman wing-fuselage on the upper wing surface. In contrast, the fully turbulent flow (the transition starts at very close to the leading edge) occurs on the upper wing surface due to the propeller slipstream effect, and the vortical regions of the propeller wake with the blade tip vortices are shown clearly in . A proper reason of the vortices formation is due to the combination of twist and bending of the specific blades. It is also noticeable that a ‘dead air’ region can be confirmed by the higher turbulent kinetic energy at near leading edge area at symmetrical plane for propeller wing-fuselage model in .

In general, the production and progression of the vertical wake is identified by the turbulent kinetic energy, as can be found in Figure 106 (a-f). The vortices with azimuth angle at 0°are marked from vortex-1 to -3 for the upper wing surface and vortex- 4 to -6 on the lower part, shown in Figure 106 (a). Vortices are moving downstream as the azimuth angle increases. It is clearly shown in Figure 106 (b and c) that both vortex -2 and -5 are moving close to vortex- 3 and -6, respectively as azimuth angle increases from ψ = 30° to 60°. As the azimuth angle keeps increasing to 90°, both vortex -2 and -5 merged with vortex- 3 and -6 to form a large vortical flow structure at downstream region. A further increasing of azimuth angle to 120°, vortices (i.e. vortex -1, -4, and new formed vortex) move further downstream as can be found in Figure 106 (e), and keep increasing of blade azimuth angle to 150°, new vortices (i.e. marked as vortex-1 and -4) are formed and further increasing of azimuth angle to 180° which would give the repeat vertical structure as shown in Figure 106 (a). Strong unsteady fluctuations were seen in the propwash during the propeller rotation with the primary effect being the periodic passage of the blade tip vortices (i.e. vortex -1 to -6). The period for the whole vortex structure progressing form Figure 106 (a) to (f) indicates that the aerodynamic forces/moments were generated with a two-cycle sinusoidal oscillation during the propeller rotation, as we detailed in subsection 5.4.1.

|  |  |
| --- | --- |
| M:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\MAV4_prop_2012_new2_3_tail_more_mesh_n555 1 12240_Vortex.png   1. Azimuth angle | M:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\MAV4_prop_2012_new2_3_tail_more_mesh_n555 1 12270.jpg   1. Azimuth angle |
| M:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\MAV4_prop_2012_new2_3_tail_more_mesh_n555 1 12000.png   1. Azimuth angle | M:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\MAV4_prop_2012_new2_3_tail_more_mesh_n555 1 12060.jpg   1. Azimuth angle |
| M:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\MAV4_prop_2012_new2_3_tail_more_mesh_n555 1 12120.jpg   1. Azimuth angle | M:\Jason_chen\MAV4_Tail\n555\0deg_more_mesh\MAV4_prop_2012_new2_3_tail_more_mesh_n555 1 12180_Vortex.png   1. Azimuth angle |

Figure 106 Turbulent kinetic energy with various azimuth angles for ω = 555 rad/s

### Summary

The effect of a tractor propeller on the flow over a Zimmerman-fuselage wing model has been investigated. Totally, three models are studied: 1) isolated propeller, 2) propeller-wing-fuselage, and 3) propeller-wing-fuselage with a vertical stabilizer. Strong unsteady fluctuations were found in the slipstream region with the primary showing the periodic aerodynamic forces/moments development.

The sinusoidal fluctuation gives a large impact on the wing-fuselage aerodynamics. The mutual influence between the propeller and the MAV become more significant as rotational speed increases. With the presence of the wing-fuselage, it induced an increase of the aerodynamic force and moment coefficients on the propeller as compared with the isolated propeller case. This is probably caused by the straightening effect of the wing on the propeller slipstream. However, the presence of the propeller gives improvements on the working range of the mean value CL of the MAV, which a delayed stall angle is achieved. The stall angle is delayed from 12° to 18°, and the CL,max is increased from 0.7 to 0.875 for isolated-MAV-fuselage and MAV-propeller model (with low rotational speed 555 rad/s), respectively. The stall angle is increased even higher when the rotational speed is 1131 rad/s (αstall = 36º with CL,max = 2.42). At CL,max, it shows that the major lift was generated by the wing-fuselage, which is 95.7% (555rad/s) and 67.8% (1131rad/s) of the total lift. Namely, the contribution on the lift from the propeller is increased as the rotational speed is increased. The overall drag curve becomes less-linear as the propeller rotational speed is increased. The propeller thrust and the drag force from the vertical stabilizer have almost a constant value for all the incidence range with both low and high rotational speeds. However, the drag force from the wing-fuselage is increased dramatically as the rotational speed is increased. This may due to the propeller wake impacted on the blunt fuselage, and high pressure drag is formed.

On the stability side, positive pitching moment slope is found at low incidence range with high rotational speed. This indicates that less reflex cambering would more suit high rotational speeds to satisfy the static longitudinal stability condition (). However, at design condition (555rad/s), the longitudinal stability margin is increased almost twice as large as a that on the isolated wing-fuselage model. The MAV without the vertical stabilizer is statically laterally unstable, although it is statically longitudinally stable. MAV with the vertical stabilizer, in general, is stable in both longitudinal and lateral.

On the flow structure side, asymmetric flow is clearly formed around the MAV. The flow (or propeller wake) is towards the up-going blade side and inclined an angle with the symmetry plane (or vertical stabilizer). The inclined angle is increased as the rotational speed is increased. This is probably the answer to explain why the additional side force and yawing moment are produced at 0º sideslip angle. The separated regions have found on both sides of the fuselage, and larger separated area is covered on the up-going blade side of the fuselage. The flow structure around the wing, however, shows that smaller separation bubble is formed on the lower wing surface near leading edge on up-going blade side. This is because the tangential flow from the propeller which impinges on the lower wing surface on the up-going side, and hence to make an attached flow. On the down-going blade side, the separation bubble can be clearly observed. Its length increases with the rotational speed is increased. The spiral flow structure at symmetric plane on the upper wing surface is clearly visualized, and the area influenced by the spiral flow is reduced with lower the propeller rotational speed.

# Fluid-Structure Interaction

The fluid-structure interaction effects on loading may lead to significant impact on overall aerodynamic performance, and hence a flexible wing may improve the MAV aerodynamics. Another interesting point is that such a flexible wing or adaptive wing MAVs is able to deal with complex flying conditions: such as gust rejection, stall delay, and also improve flying stability.

Smart material (lightweight, thin, flexible and expendable) has been identified as being well suited for micro air vehicles including composite material, usually carbon fibre [127-129], membrane skin[130-132], and depron foam [115, 133]. Composite material has a property of high rigidity and the effect on aerodynamics due to the deformation under the gust loading can be ignored. The density of the composite material is much higher than the membrane skin and depron foam. This indicates that for the same design, the composite wing has less capability on payload than the depron foam wing. The membrane wing, on the other hand, is a hyperelastic solid material (depends on how much pretension is applied on the membrane skin [134]) which may offer high deformation during the flight and this would modify the overall aerodynamic performances. The flow unsteadiness, another important issue, due to separation might cause the membrane vibration leading to aeroelastic instability [135]. An adaptive wing can be achieved using an appropriate combination of composite and membrane or depron foam (a depron foam wing is shown in Figure 3). The longitudinal static stability is improved due to the passive shape adaption from the membrane skin [103]. The fabrication of a composite-membrane wing is quite complex, where the carbon fibre skeleton is affixed to an extensible membrane skin. The depron foam, on the other hand, is a compromise as it is not as rigid as the composite material or not that flexible as the membrane skin [29, 103]. The fabrication of a depron foam wing is relativly simple. The root chord was swept in the spanwise to form a constant rectangular wing planform and then the Zimmerman wing planform is projected onto the rectangular wing surface [4].

To optimize the aerodynamic performance, it is critical to understand the aerodynamics due to the fluid structure interaction. Within current FSI research, time-dependent problems involving large structural deformations are of particular interest with potential applications ranging from hyper-elastic structural membranes to high rigidity carbon fibre structural modelling. The fluid-structure coupled problem results in a system of non-linear algebraic equations which are usually resolved by either a monolithic or a partitioned approach and a brief discussion has been introduced by Ref. [66, 76]. A monolithic approach uses all the domains simultaneously, leading to a compacted algebraic equation including both fluid and structural variables. Such approach has the advantage that the kinetic boundaries at the interface are solved simultaneously within both fluid and structural domains. The monolithic approach is a stable scheme with fast convergence and less limitations on the time scales. However, the computation time is very high, and this approach is relatively expensive. Simple academic problems using monolithic approach has been introduced by Michler et al. [136] and Hubner et al.[137]. In contrast, the partitioned approach uses separated solvers, in which each physical field is separately defined, discretised and numerically solved with a data transfer interface. Such approach costs less time and it has a relatively high flexibility due to modules are developed individually. Another general classification of the fluid-structure interaction solution is based on the treatment of the meshes: the conforming mesh methods and non-conforming mesh methods [66]. The conforming-mesh method requires re-meshing at the structural-fluid interface for the movement and/or deformation. An example of using conforming-mesh has been introduced by Miao [138]. On the other hand, the non-conforming mesh methods do not require the re-meshing procedure, it treats the boundary condition and the related interface conditions as constrain imposed on the model equations.

For the present investigation, the fluid structure interaction using a partitioned approach is employed. Both fluid and structural solver are introduced and the data transfer at the interface is also described. Also another focus point is on the low Reynolds number aerodynamic characteristics of the deformed MAV. The model employed in this investigation considers a carbon fibre rod underneath the wing to increase the wing strength. Contact algorithms are also introduced for the finite element model. On the fluids side, the Navier-Stokes equations using the conformal-structure grids are solved. The dynamic mesh technique is adopted to deal with the effect of the deformation on the flow structures on the upper and lower wing surfaces.

### Validation case study

In an attempt to validate the proposed coupled fluid-structure interaction system, results were obtained from a three-dimensional MAV wing (152mm wingspan, 124mm root chord and AR = 1.25), shown in the finite element model in Figure 107 (a). The MAV wing has a root positive camber of 6.8% locating at 0.22c and the negative/reflex camber of 1.4% at 0.86c. The model was designed to have a 7° dihedral angle from 2y/b = 0.4 to the wing tip and a positive 7° geometry twist is also built at wing tip. The wind tunnel results have been provided by the author Stanford et al. [103]. A closed-loop wing tunnel with test section dimension of 0.84m height and 2.44m deep was used. The velocity can be adjusted from 2 to 45m/s and the maximum Reynolds number is 2.7million. The free stream turbulent intensity is 0.2% and also the typical uncertainties are 5% for CL, 7% for CD and 20% for CM. The uncertainty values would double at high incidences (e.g. stall angle). The wing was built using a carbon fibre skeleton attached with a membrane skin at the centre part of the wing, as shown in Figure 107 (b). The bi-direction plain weave carbon fibre composite are used to the rigid part. The plain weave has a Young’s modulus E12 = 34.8GPa, Poisson’s ratio ν12 = 0.41 and shear modulus G12 = 2.34GPa. The wing was built from two plies of the same plain-weave carbon fibre, and the bi-directional plain weave carbon fibre composite with fibres constructing in ± 45° [29]. The average thickness of the membrane skin is roughly 0.1016 ± 0.0508mm and the density is 980kg/m3, and the reasonable membrane skin thickness assumption for finite element is approximately 0.15 ± 0.01mm [139]. The Poisson’s ratio for small strains is 0.5, a result for the material’s incompressibility. Based on the experiment data given in the reference, wind tunnel speed 15 m/s is used for the validation. This gives the corresponding Reynolds numbers 127,000.

Both fixed-wing and flexible-wing MAV are validated. The aerodynamics comparisons for the fixed-wing MAV are shown in Figure 108. It shows that the numerical method is able to predict the prestall lift and drag coefficients over a rigid-wing MAV and the trends fall consistently within the experimental error bar. The stall angle is under-predicted by about 2 deg. Both numerical and experimental results agree the linear change in lift with various angles of attack. The numerical solution is able to predict the onset of stall at about 21°. However, the lift coefficient seems under-predicted at post stall region, as the flow is known to be highly unsteady [140]. For the flexible-wing MAV, as shown in Figure 107 (b), numerical and experiment out-of-plane displacements, normalized by the root, are validated and results are quite comparable, as shown in Figure 109 and Figure 110, respectively. The aerodynamic characteristics of the flexible-wing MAV seem to have similar trends with the rigid-wing MAV. The results fall within the experimental error bar. However, under-prediction can be found at incidences larger than 15°. This may be due to the flow unsteadiness; it becomes even more complex due to the vibration of the membrane skin (i.e. vibration due to the fluid-structure interaction). This is also mentioned by Lian *et al* [141-142] who used the time-averaged unsteady computations for incidences larger than 15deg and steady computations are used at other incidences. The membrane experiences a high frequency vibration, which would modify the effective angle of attack, and hence to effect the overall aerodynamic characteristics.

The comparison is listed in , showing both lift and drag values are over-predicted, however, the uncertainty is less than 5%, (i.e. lift is less than 0.6% and drag is less than 5%). The leading edge moment seems to have a slightly larger uncertainty around 30%. Figure 111 shows the residual history of the aerodynamic characteristics and the deformation. The time step was set to sufficiently small and resolving the time that the incoming flow would need to travel one chord length with 5 time steps (i.e. 0.0016s). In this case, about 125 time steps with about 30 inner iterations per time step were required for the solution process.

|  |  |
| --- | --- |
| C:\Users\jason\Desktop\vector_plot\fixed_wing.png   1. Fixed-wing model, Ref [29] | C:\Users\jason\Desktop\vector_plot\perimeter reinforced2.png   1. Flexible-wing model, Ref [29] |
| J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV4_zim\MAV4_FSI\FSI_validation\fixed_wing.jpg   1. CFD mesh | C:\Users\jason\Desktop\0deg.jpg   1. FEM model |

Figure MAV model and mesh topology

|  |  |
| --- | --- |
| J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV4_zim\MAV4_FSI\FSI_validation\Validation_results\deformed_1_15ms(rigid)\ist_time_tried\Validation-FSI-15ms-CL.emf   1. CL | J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV4_zim\MAV4_FSI\FSI_validation\Validation_results\deformed_1_15ms(rigid)\ist_time_tried\Validation-FSI-15ms-CD.emf   1. CD |
| J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV4_zim\MAV4_FSI\FSI_validation\Validation_results\deformed_1_15ms(rigid)\ist_time_tried\Validation-FSI-15ms-CM.emf   1. CM |  |

Figure Aerodynamic coefficients for rigid-wing MAV at 15m/s, References. [140]

|  |  |
| --- | --- |
| C:\Users\jason\Desktop\validation3\CL_FSI_validation.emf   1. CL | C:\Users\jason\Desktop\validation3\CD_FSI_validation.emf   1. CD |

Figure Aerodynamic coefficients for flexible-wing PR-MAV at 15m/s, Reference. [140]

Table Comparison for flexible-wing MAV at angle of attack 15° at 15m/s

|  |  |  |  |  |
| --- | --- | --- | --- | --- |
| Flexible model | CL | CD | CM,LE | Maximum z/c (out-of-plane) |
| FSI computation | 1.0105 | 0.2392 | -0.3059 | 0.0578 |
| Experimental  [140] | 1.0237 ±0.051 | 0.2467±0.017 | -0.2606±0.052 | 0.0582±0.0013 |

J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV4_zim\MAV4_FSI\FSI_validation\out_of_plane_displacement.emf

Figure out-of-plane displacement comparison (w/c), α= 15°, Ref. [140]

|  |  |
| --- | --- |
| J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV4_zim\MAV4_FSI\FSI_validation\geo15deg\FSI_Vali_15deg_LDM.emf   1. Aerodynamic coefficient history | J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV4_zim\MAV4_FSI\FSI_validation\geo15deg\FSI_Vali_15deg_deformation.emf   1. Deformation history |

Figure Residual history

### Geometry description and moving mesh

The rigid-wing MAV geometry study on the wing planform effects earlier in Chapter 4, shows that the Zimmerman wing planform has the best aerodynamic efficiency with a reasonable good longitudinal stability margin. However, the MAV aerodynamic performance under the aerodynamic force loading with deformation still has not yet investigated. Figure 112 shows the static model for the structure modelling. The carbon fiber rods are attached underneath the wing (Figure 112 b).

For the fluid-structure interaction, the depron foam is assumed to be isotropic, with equal material properties in all directions. The density of depron foam is 46kg/m3, the Young’s modulus is 10MPa, the Poisson ratio is about 0.27. The density of carbon fibre is 1650kg/m3, the Young’s modulus is 3×1011Pa and a Poisson ratio of 0.3 is used.

|  |  |
| --- | --- |
| J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV4_zim\MAV4_FSI\mesh\carbon_fiber_rod\mesh_assembled.jpg   1. Fluid domain | J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV4_zim\MAV4_FSI\mesh\carbon_fiber_rod\mesh_assembled2.jpg   1. Solid domain |

Figure Geometry meshes for fluid-structure interaction

In Chapter 4, we have discussed the wing planform effects as well as the aerodynamic influence due to the fuselage effect. To further improving the aerodynamic performance, the flexible wing which based on the Zimmerman wing-fuselage model will be studied. The flexible wing-fuselage models with/without carbon fibre rods installed are studied in the Section 6.3.

### Flexible wing-fuselage MAV model

**Aerodynamic performance and flow structure (without carbon fibre rods installed)**

Figure 113 (a-c) shows the lift, drag, and pitching moment coefficients for the rigid and deformed MAV. The rigid-MAV, as plotted by the dashed line with circular markers, has a stall angle about 12°, and a sudden stall is observed. The possible reason would be due to an abrupt separation of the laminar boundary layer near the leading edge without subsequent reattachment. For the deformed wing, an earlier stall angle is observed (i.e. αstall = 9°). The stronger nonlinear effect is found for the deformed MAV at moderate incidences as compare with the rigid wing. Similar lift slope, CLα, between rigid and deformed MAV are shown at incidence greater than 0°, and a increased lift slope is found for the deformed MAV at α < 0°. A mild reduction on the lift at post stall region for the deformed wing is because the large vortex system formed on the upper wing surface causing a low pressure area and hence increasing the lift. However, the disappearance of this particular vortex system corresponds to the abrupt drop of lift coefficient.

The drag coefficient variation, (Figure 113 b), shows that the deformed MAV increases in drag significantly larger than the baseline model. This is mainly due to the variation of both maximum camber and minimum camber. At incidences below 0°, the camber is twisted to form a negative angle of attack locally (i.e. the leading edge deformed downwards and more positive camber is generated) as shown in the contour plot in Figure 114 (a). The drag penalty at α >0° can be clearly found in the plot. The deformed aerofoil has a very nonoptimal aerodynamic shape (most due to the geometrical and aerodynamic twist on aerofoil locally), and this may cause further flow separation.

The pitching moment coefficient, Cm,c/4 (Figure 113 c), shows a similar trend and a difference can be highlighted at low incidence regime (i.e. at 0°regime). According to the longitudinal stability rule, negative moment slope indicates a stable condition. However, only a rigid wing can be trimmed at a positive angle of attack. In order to trim at positive angle of attack, needs to be greater than zero (0). Apparently, the deformed wing shows a negative value of . The difference can be seen clearly between these two models: the deformed model is statically longitudinal unstable. The baseline model, on the other hand, is statically longitudinal stable.

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| J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV4_zim\MAV4_zim\ICEM\wing_only\MAV4_wing_only_deformed_CL2.emf   1. CL | J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV4_zim\MAV4_zim\ICEM\wing_only\MAV4_wing_only_deformed_CD2.emf   1. CD |
| J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV4_zim\MAV4_zim\ICEM\wing_only\MAV4_wing_only_deformed_CM2.emf   1. CM | |

Figure Aerodynamic characteristics for MAV

The aerodynamic twist (cambers and camber locations) and geometrical twist angle distribution for the deformed MAV models are shown in (a, b). The baseline model (or rigid model) is characterized by a gradual increase in nose-up twist towards wingtip and a maximum of 12° of geometrical twist angle is achieved at the spanwise station 2z/b = 0.82 and a slight decrease the twist angle is designed for the wingtip to make the wash-in flow, to delay the wingtip stall as much as possible. The geometrical twist details at design condition (CL = 0.35) can be found in (a). The deformed wing, on the other hand, has shown a reduced geometrical twist angle (i.e. nose down) as showed by the solid line. This would mainly due to the deformation occurred at leading edge and pushed downwards to generate more positive camber, shown in Figure 114 (c).

The comparison of the geometrical twist angle at the mean chord location (i.e. 2z/b = 0.536) between rigid and flexible wing are shown in (b). The reduced geometrical twist angle was found at α < 12° and larger angles at α > 12°. The maximum camber (hmax/c), in Figure 114 (c), is reduced as the incidence increases. The possible reason is because the increase of the twist angle towards the wingtip (i.e. due to the Zimmerman wing planform effect), and this leads to an increase on the upwards wing deformation towards the wingtip (i.e. minimize the maximum camber and wing deformations with various incidences are shown in Figure 115). The wing deforms upwards to form a dihedral angle and it increases as the incidence increases. The dihedral angle would help to increase the rolling stability (i.e.). The negative rolling moment slope indicates that this moment tries to bring the MAV back to a wing-level attitude when suffer a side slip. Figure 115 shows the variation of the deformations with increased incidences and the details of the dihedral angles are listed in Table 21.

|  |  |
| --- | --- |
| J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV4_zim\MAV4_zim\ICEM\design_condition\deformation\twist\MAV4_wing_only_twist.emf   1. Geometrical twist angle at CL = 0.35 | J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV4_zim\MAV4_zim\ICEM\wing_only\MAV4_wing_only_deformed_twist.emf   1. Geometrical twist angle with various α at 2z/b = 0.536 |
| J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV4_zim\MAV4_zim\ICEM\wing_only\MAV4_wing_only_deformed_maxC.emf   1. hmax/c (at 2z/b = 0.536) |  |

Figure Aerodynamic and geometrical twist angle variation

|  |  |
| --- | --- |
| K:\Jason_chen\wing_only\Two_way_FSI\FSI_N6deg.jpg (unit Pa) | |
| K:\Jason_chen\wing_only\Two_way_FSI\FSI_N6deg.jpg   1. α = -6° | I:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV4_zim\MAV4_zim\ICEM\wing_only\wing_only\Two_way_FSI\FSI_N1p25deg.jpg   1. α = -1.35° (CL = 0.35) |
| G:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV4_zim\MAV4_zim\ICEM\wing_only\wing_only\Two_way_FSI\6deg_up_pressure.jpg   1. α = 3° | K:\Jason_chen\wing_only\Two_way_FSI\FSI_6deg.jpg   1. α = 6° |
| K:\Jason_chen\wing_only\Two_way_FSI\FSI_9deg.jpg   1. α = 9° | K:\Jason_chen\wing_only\Two_way_FSI\FSI_12deg.jpg   1. α = 12° |

Figure MAV deformed shapes with incidences vary from α = -6° to 12°

Table wing tip deformation details at x/c = 0.25

|  |  |  |
| --- | --- | --- |
| α° | dihedral angle (Γ°) | tip deformation (y/c) |
| -6 | -4.858 | -0.07826 |
| -1.35a | 1.0633 | 0.040636 |
| 3 | 2.248 | 0.059018 |
| 6 | 9.88 | 0.177955 |
| 9 | 18.93 | 0.309773 |
| 12 | 22.64 | 0.361091 |
| 18 | 24.87 | 0.388136 |
| 24 | 26.18 | 0.403864 |

a Design condition, CL = 0.35

The three dimensional flow patterns for deformed models from -6° to 6° are presented in Figure 116 a-d. For the deformed model at -6°, fully attached flow is observed on most of the upper wing surface except at the trailing edge region where separation occurs. On the lower surface, a large leading edge vortex is found which is mainly due to the nose-down twist angle. As the incidence increased to the design condition (i.e. CL = 0.35, α -1.35°) in Figure 116 (b), a ‘dead-air’ region is formed on the upper wing surface at the wing root location, the size is much larger than that for the baseline model, Figure 62 (c). A mild leading edge separation region is found on the lower surface and the fuselage effect is reduced as compare with the previous incidence. For incidence at, the ‘dead-air’ region has been enlarged to form the vortical flow on the upper wing surface which covers the entire chord length at the centre of the wing. The leading edge vortex is also clearly shown on the wing surface, which gets stronger towards the wingtip and interacts with the tip vortex.

As the incidence increases (from 3° to 6°), the vortical flow on the upper central wing surface getting stronger. The progress of the leading edge vortex structure from low to high incidence can be clearly seen from the plots (Figure 116 c-d). The vortical flow and the tip vortex would interact as the angle of attack further increases. The large low pressure area formed on the upper surface due to the vortical flow, which can produce the additional lift. This is the main effect of the large deformation.

|  |  |
| --- | --- |
| J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV4_zim\MAV4_zim\ICEM\wing_only\wing_only\Two_way_FSI\MAV4_wing_only_N6deg_unsteady-1-02100_up_surface.jpg   1. α = -6° (upper) | J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV4_zim\MAV4_zim\ICEM\wing_only\wing_only\Two_way_FSI\MAV4_wing_only_N6deg_unsteady-1-02100_lowwer_surface.jpg  α = -6° (lower) |
| J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV4_zim\MAV4_zim\ICEM\wing_only\wing_only\Two_way_FSI\MAV4_N1p25deg_up_surface.jpg   1. α = -1.35° (upper)-DC | J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV4_zim\MAV4_zim\ICEM\wing_only\wing_only\Two_way_FSI\MAV4_N1p25deg_lowwer_surface.jpg  α = -1.35° (lower)-DC |
| J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV4_zim\MAV4_zim\ICEM\wing_only\wing_only\MAV4_wing_only_3deg1_unsteady-1-02100_up_surface.jpg   1. α = 3° (upper) | J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV4_zim\MAV4_zim\ICEM\wing_only\wing_only\MAV4_wing_only_3deg1_unsteady-1-02100_lowwer_surface.jpg  α = 3° (lower) |
| J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV4_zim\MAV4_zim\ICEM\wing_only\wing_only\MAV4_wing_only_6deg1_unsteady-1-02100_up_surface.jpg   1. α = 6° (upper) | J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV4_zim\MAV4_zim\ICEM\wing_only\wing_only\Two_way_FSI\MAV4_wing_only_6deg1_unsteady-1-02100_lowwer_surface.jpg  α = 6° (lower) |

Figure 116 Surface flow pattern at series of angle of attack

**Aerodynamic performance and flow structure (with carbon fiber rod installed)**

The depron foam has a low strength and the large deformations lead to a significant drop on the overall aerodynamic performances as discussed in the previous section The advantage of using the flexible material is that it can facilitate passive shape adaption, which would delay stall. However, flexible wing also would make some undesirable deformation, which degraded the overall vehicle performances. For the current MAV design, carbon fibre rods with diameter of 3mm are added underneath the wing to make a desirable deformation to improve the overall aerodynamic performance. A proper location might increase the structure strength without lower the overall aerodynamic performances or sometimes even improve the overall aerodynamic efficiency. The current location configuration as plotted in Figure 117 shows a reasonable good result (i.e. will be discussed in the following subsection). The rod locations are listed in Table 22.

Using a carbon fibre rod to improve the overall aerodynamic performance is a quite new idea. However, using rod to make a controllable wing shape was attempted before. A general description of using rod to change the camber by a novel rod-and-driving tube hinge was studied by Radespiel [106]. The change of camber can be made from 3% to 9%. The details of aerodynamic characteristics are not available in the paper and no further improvements were made by the author. Another design was introduced by Stanford and Abdulrahim et al. [143], who used the rod to achieve the roll control. The rod command is implemented through a torque-rod-actuation structure, which forces the wing into an asymmetric wing twist.

C:\Users\jason\Desktop\vector_plot\fuselage_rod.emf

Figure 117 Carbon fiber rod configuration

Table rod location cases

|  |  |  |
| --- | --- | --- |
| No. of rods |  |  |
| 2 | 0.15 (33mm) | 0.935 (207mm) |

#### FSI on aerodynamics

Simulations were run for a range from -3° to 24° and the freestream velocity is 8.4m/s. The convergence history was plotted with normalized displacement (y/c) versus the time step in Figure 118. The time step was set to be 2.6357×10-4 s for freestream 8.4, corresponding to 1/100 chord of flow through time (i.e.). Namely, resolving the time that the incoming flow would need to travel one chord with 100 time steps [144]. Steady state FSI solutions were found from all the simulations, (Figure 118), less than 0.4s time duration were usually adequate for simulations at low incidences, and through up to 0.65s may be needed otherwise. Convergences are given in terms of the maximum displacement at wing tip computed at each time step, and all results are normalized by the mean chord. Smoother flow field were obtained at low angles of attack, whereas full separated flow were found at high angles of attack facilitates slower aeroelastic convergence.

Figure 119 (a-d) shows the aerodynamic performance of the flexible MAV model at freestream speeds 8.4m/s. For a constant MAV weight, W = 140g, the design lift coefficient will be fixed at CL =0.35. The base-line model, rigid MAV, needs a lower angle of attack to achieve the same amount of lift as compared with the flexible MAV at same flying speed 8.4m/s (i.e. CL8.4-rigid = 0.35 at α8.4-rigid = - 0.1348°). The, , and relationships polar are detailed for both rigid and flexible wings. The CLα has shown a linear property at low to moderate incidences (i.e. -3°<α<9°), despite the MAV has a low aspect ratio of 2.16. Torres and Mueller [9] has shown that aspect ratio of 1.25 as a general cut-off between linear and non-linear behaviour. However, Stanford [145] pointed out that the MAV with aspect ratio of 1.2 still provide linear properties. For all cases considered, the slope of the lift curve for deformed MAV is more linear than the rigid model. The lift variation with incidence is gradual and continuous for the rigid MAV, which experiences trailing-edge stall. Similar stall trend can be found from the deformed MAV case, whereas the stall angle as compared with the rigid MAV which has been extended for 3° (i.e.). A mild reduction on the lift at post stall region is because the large vortex system formed on the upper wing surface which can cause a low pressure area and hence increase the lift. However, the disappearance of this particular vortex system corresponds to the abrupt drop of lift coefficient.

The drag coefficient variation, in Figure 119 (b), has shown that the deformed MAV is increased drag dramatically, which larger than the baseline model. This is mainly due to the variation of both maximum camber and minimum camber. As incidence increases, the maximum camber (h1) is increased and the maximum camber location (d1) is shifted towards the trailing edge. This leads to the penalty on the drag coefficient which is increased significantly. The camber variation can be found in Figure 120 (a-f).

The longitudinal pitching moment coefficient characteristics, CM0.25, calculated about the quarter mean aerodynamic chord and the lift-to-drag ratio as a function of incidences are given in Figure 119 (c-d). For the all the models, as expected, the pitching moment has a negative slope which indicated all models are longitudinal statically stable (i.e. CMα < 0), and the negative magnitude means the nose down moment will be achieved. Stanford et al. [145] has shown the pitching moment coefficients summed around the leading edge and he mentioned that the summing moments about the quarter chord will have two orders of magnitude less than the leading edge pitching moment. From the pitching moment plots, it is noticeable that the pitching moment plot about the quarter mean-aerodynamic chord, CMα, is more negative for the deformed MAV than the rigid MAV. This is because that both upward force and the moment arm are adaptively increased with angles of attack and dynamic pressure, from the increased camber and the camber location shifted backwards (i.e. towards leading edge, this can be evidenced in Figure 120 (b) ).

The major difference appears between the aerodynamic efficiency, CL/CD, for both cases are around the zero incidences. The rigid wing has a much higher aerodynamic efficiency; the proper reason may be the carbon fibre rods are attached under the wing, which modifies the shape of the lower surface of the wing. However, the deformed MAV has a higher aerodynamic efficiency between moderate to high angles of attack.

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Figure 118 maximum deformation convergence history

|  |  |
| --- | --- |
| I:\jasonchen\MAV4_FSI\MAV4_2Rods_2way_FSI\mav4_2rods_33_3\deformed_CL2.emf   1. CL | L:\jasonchen\MAV4_FSI\MAV4_2Rods_2way_FSI\mav4_2rods_33_3\deformed_CD2.emf   1. CD |
| L:\jasonchen\MAV4_FSI\MAV4_2Rods_2way_FSI\mav4_2rods_33_3\deformed_CM3.emf   1. CM | L:\jasonchen\MAV4_FSI\MAV4_2Rods_2way_FSI\mav4_2rods_33_3\deformed_CL_CD2.emf   1. CL/CD |

Figure Aerodynamic characteristics for rigid and flexible MAV models

To further the analysis of the wing deformation, the local chordwise (in X direction) shape (i.e. wing section deformed and rigid profiles) are plotted in Figure 120(b) at a spanwise location 2Z /b= 0.536 (i.e. mean aerodynamic chord in spanwise location). The aerofoil sections have been normalized by the mean aerodynamic chord (i.e. h/c) and for the purpose of easy comparison the aerofoils are plotted without the twist angles (i.e. zero twist angle). Results are given for both rigid and flexible models and the aerodynamic twist angles for the models are shown in (a). The baseline model (or rigid model) is characterized by a constant nose-up twist angle with various angles of attack. The flexible model, on the other hand, shows a linear function between the twist angle and the incidences at low to moderate incidences (i.e.-3° <α<6°) and non-linear slope can be found as the incidence keeps increasing. The adaptive washout or substantial rotation of the aerofoil profile about the leading edge can be found from the figure, provides the superior lift slope and extended stall angle for deformed model which has been discussed previously in Figure 119 (a).

The maximum camber in Figure 120 (c) has been normalized by the aerodynamic chord (i.e. h1/c), maximum camber for both rigid and deformed models are plotted. The rigid MAV shows a constant maximum camber, h1/c = 0.0419, as marked with the black dashed line. The maximum camber for the deformed MAV is increased as the angle of attack is increased. The maximum camber shows a linear slope at incidence between -3° and 3° for deformed model. The camber is a largely linear function of angle of attack, a direct result of the linear CLα relationship. Beyond 18° angle of attack, the wing camber starts to level off. This may be the results of large separated flow occurred on the upper surface, causing the wing to decamber. The maximum camber location (d1/c) in (d) shows the maximum camber location moves towards the trailing edge as incidence increases and it is increased up to 40% for incidence between 9° and 24°. The rigid wing, however, has a constant maximum camber location which is about 25% of the mean chord for all incidences.

The maximum reflex camber and its location, h2/c and d2/c, are also quite important for the small air vehicles (Figure 120 e-f). The reflex camber is usually for balancing the forces and moments, and usually the large negative, nose-down pitching moment coefficients were found for cambered, low-aspect-ratio wings at low Reynolds number. Therefore, the biggest factor driving the design of the small air vehicles is the longitudinal stability and control. MAV wing under the aerodynamic loading usually would make the deformation and hence to modify the reflex camber. In Figure 120 (e-f), the rigid MAV has the maximum reflex camber is about -1.6%c at 0° incidence. The deformed MAV, on the other hand, shows a less amount reflex camber. The reflex camber is about -0.128%c at 0° of angle of attack for deformed model. This is because the pressure force on the lower wing surface pushes the wing upwards and hence reduces the amount of the reflex camber. In general, the maximum reflex camber continuously decreases as the angle of attack increases for the deformed model. The maximum reflex camber location locates at about 84%c for the rigid MAV; the reflex camber location of deformed MAV varies with the angles of attack, as shown in Figure 120 (f).

Wing deformation on spanwise direction (i.e. x/c = 0.5) has shown in Figure 121. The deformation apex occurs approximately at the maximum wing span and this location is a function of the angle of attack, as the peak will move slightly downwards (towards trailing edge seen in Figure 120 d) with increased incidences. The spanwise deformation shows a nonlinear slope and the nonlinearity increases with increased angle of attack. At the wing root section, the wing has attached on the fuselage, no deformation is occurred. The flat portion in the plot in Figure 121 shows the width of the fuselage and the details of deformation are shown in the contour plots in Figure 122.

|  |  |
| --- | --- |
| I:\jasonchen\MAV4_FSI\MAV4_2Rods_2way_FSI\mav4_2rods_33_3\deformation_property\84ms\mean_camber_twist_angle2.emf   1. Mean chord geometrical twist angle (°) | L:\jasonchen\MAV4_FSI\MAV4_2Rods_2way_FSI\mav4_2rods_33_3\deformation_property\84ms\deformed_airfoil.emf   1. Mean Camber deformation |
| I:\jasonchen\MAV4_FSI\MAV4_2Rods_2way_FSI\mav4_2rods_33_3\deformation_property\84ms\maximum_camber2.emf   1. Maximum camber (h1/c) | I:\jasonchen\MAV4_FSI\MAV4_2Rods_2way_FSI\mav4_2rods_33_3\deformation_property\84ms\maximum_camber_location2.emf   1. Maximum camber location (d1/c) |
| I:\jasonchen\MAV4_FSI\MAV4_2Rods_2way_FSI\mav4_2rods_33_3\deformation_property\84ms\minimum_camber2.emf   1. Minimum camber (h2/c) | I:\jasonchen\MAV4_FSI\MAV4_2Rods_2way_FSI\mav4_2rods_33_3\deformation_property\84ms\minimum_camber_location2.emf   1. Minimum camber location (d2/c) |

Figure camber shape, locations, and twist angle variations at 2z/b = 0.536

L:\jasonchen\MAV4_FSI\MAV4_2Rods_2way_FSI\mav4_2rods_33_3\deformation_property\84ms\spanwise_deformation_84ms.emf

Figure wing spanwise deformation at x/c = 0.5

|  |  |
| --- | --- |
| L:\jasonchen\MAV4_FSI\MAV4_2Rods_2way_FSI\mav4_2rods_33_3\8_4_ms\MAV4_2rods_33_3_N3deg_84ms.eps   1. α = -3° | L:\jasonchen\MAV4_FSI\MAV4_2Rods_2way_FSI\mav4_2rods_33_3\8_4_ms\MAV4_2rods_33_3_0deg_84ms.eps   1. α = 0° |
| L:\jasonchen\MAV4_FSI\MAV4_2Rods_2way_FSI\mav4_2rods_33_3\8_4_ms\MAV4_2rods_33_3_3deg_84ms.eps   1. α = 3° | L:\jasonchen\MAV4_FSI\MAV4_2Rods_2way_FSI\mav4_2rods_33_3\8_4_ms\MAV4_2rods_33_3_6deg_84ms.eps   1. α = 6° |
| L:\jasonchen\MAV4_FSI\MAV4_2Rods_2way_FSI\mav4_2rods_33_3\8_4_ms\MAV4_2rods_33_3_9_84ms.eps   1. α = 9° | L:\jasonchen\MAV4_FSI\MAV4_2Rods_2way_FSI\mav4_2rods_33_3\8_4_ms\MAV4_2rods_33_3_12_84ms.eps   1. α = 12° |
| L:\jasonchen\MAV4_FSI\MAV4_2Rods_2way_FSI\mav4_2rods_33_3\8_4_ms\MAV4_2rods_33_3_15_84ms.eps   1. α = 15° | L:\jasonchen\MAV4_FSI\MAV4_2Rods_2way_FSI\mav4_2rods_33_3\8_4_ms\MAV4_2rods_33_3_18_84ms.eps   1. α = 18° |

Figure MAV deformed shapes with various incidences

#### FSI on flow structure

To help understand the aerodynamics of the flexible MAVs, CFD with the flow structure around the MAVs has been analyzed. Although the true nature of transition between laminar and turbulent flow is still not fully solved. The three dimensional flow structures with various angles of attack has shown in Figure 123 (a-j) flight speed V = 8.4m/s. The three dimensional flow structures for the rigid model are also plotted in Figure 123.

The pressure contours on the upper wing surface are plotted for incidences from -3° to 12° for both rigid and deformed models. The low pressure area on the upper surface near leading edge region and towards the wing tip, associated with the leading edge separation, can be visualized for both models. However, a tiny leading edge separation region can be found at low incidence α = -3° for the deformed model. The large pressure drop on the wing tip at moderate incidence indicates a strong wing-tip vortex; this can be seen clearly with blue contour on the upper wing surface for both models. A ‘dead air’ region is formed on the upper wing surface at near negative camber location for both models, and this separation area enlarges as the incidence increases (). Fully separated flow is observed on the entire upper surface for. This separation region also gives the relevant reason why such a significant increasing on the drag coefficient for the deformed wing at high incidences. In contrast, the separation region on the upper surface of the rigid model at low incidence (i.e. ) is smaller than the flexible wings. However, the separated region on the upper wing surface for the rigid wing is much larger than the deformed wing at moderate to high incidences (Figure 123 ).

|  |  |
| --- | --- |
| L:\jasonchen\MAV4_FSI\MAV4_2Rods_2way_FSI\mav4_2rods_33_3\8_4_ms\MAV4_FSI_84ms_N3deg.png  α = -3° (V = 8.4m/s) | J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV4_zim\MAV4_zim\MAV4_fuse\steady\84ms\MAV4_84ms_N3deg2.png  α = -3° (8.4m/s, rigid) |
| L:\jasonchen\MAV4_FSI\MAV4_2Rods_2way_FSI\mav4_2rods_33_3\8_4_ms\MAV4_FSI_84ms_0deg.png  α = 0° (V = 8.4m/s) | J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV4_zim\MAV4_zim\MAV4_fuse\steady\84ms\MAV4_84ms_0deg2.png  α = 0° (V = 8.4m/s, rigid) |
| L:\jasonchen\MAV4_FSI\MAV4_2Rods_2way_FSI\mav4_2rods_33_3\8_4_ms\MAV4_FSI_84ms_3deg.png  α = 3° (V = 8.4m/s) | J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV4_zim\MAV4_zim\MAV4_fuse\steady\84ms\MAV4_84ms_3deg2.png  α = 3° (V = 8.4m/s, rigid) |
| L:\jasonchen\MAV4_FSI\MAV4_2Rods_2way_FSI\mav4_2rods_33_3\8_4_ms\MAV4_FSI_84ms_6deg.png  α = 6° (V = 8.4m/s) | J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV4_zim\MAV4_zim\MAV4_fuse\steady\84ms\MAV4_84ms_6deg2.png  α = 6° (V = 8.4m/s, rigid) |
| L:\jasonchen\MAV4_FSI\MAV4_2Rods_2way_FSI\mav4_2rods_33_3\8_4_ms\MAV4_FSI_84ms_9deg.png  α = 9° (V = 8.4m/s) | J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV4_zim\MAV4_zim\MAV4_fuse\steady\84ms\MAV4_84ms_9deg2.png  α = 9° (V = 8.4m/s, rigid) |
| L:\jasonchen\MAV4_FSI\MAV4_2Rods_2way_FSI\mav4_2rods_33_3\8_4_ms\MAV4_FSI_84ms_12deg.png  α = 12° (V = 8.4m/s) | J:\Jason_chen\PHD2\PHD_year2\MAV\MAV1_MAV6\MAV4_zim\MAV4_zim\MAV4_fuse\steady\84ms\MAV4_84ms_12deg2.png  α = 12° (V = 8.4m/s, rigid) |

Figure Flow structures for both speeds for rigid and flexible MAVs

### Summary

The deformed Zimmerman wing-fuselage model (without carbon fiber rods) shows a significant wing deformation under the aerodynamic loading. The substantial increase on the drag coefficient reduces the overall aerodynamic performance. Most importantly it also shows that this deformed MAV is statically longitudinally unstable at design condition. The deformed MAV with carbon fiber rods, on the other hand, substantially increases the maximum lift (CL,max) and the lift slope as compared to the rigid wing. This indicates that the deformed MAV shows more capabilities for carrying heavy payloads as compare to the rigid wing. The lift curve of the deformed wing shows less nonlinear behaviour than the rigid MAV, and a possible reason behind this would be the variation on the maximum camber locations which can offset the growth of the tip vortices.

Separated flow is observed on the upper wing surface for both models (rigid and flexible model with carbon fiber rods). The separation area on the upper wing surface for the deformed wing is minimized as compared with the rigid wing. The stall angle has been extended to 15° for the deformed wing, whereas rigid wing has a stall angle about 12°. The deformed wing shows a reduced separated region on the upper surface. This may give the reason why the stall angle is delayed. The overall aerodynamic efficiency is increased between moderate and high incidences.

On the stability side, similar pitching moment slopes are obtained from those two models. The deformed wing shows more linear hebaviour on the pitching moment curve at low to moderate incidences (-3° < α < 9°), and a nearly constant pitching moment coefficient is shown at high incidences.

# 

# Conclusion and Future Work

### Conclusions

The laminar separation bubble (LSB) formation and character with both two-dimensional and three dimensional cases were investigated. By comparing the result for LSB at symmetrical planes with those from 2-D, a delayed LSB formation is found on the 3-D wing upper surface. With increasing angles of attack, the 3-D LSB expands chordwise downstream and spanwise outboard gradually. The bubble is tapered towards wingtip and eventually merged with the wingtip vortex. The sudden jump in the lift-curve slope is shown to be associated with the expansion of the LSB for both 2-D and 3-D cases. The strong trailing edge recirculation is clearly shown on upper surface for both 2-d and 3-D cases. LSB also forms near the leading edge on the lower surface for both 2-D and 3-D cases at zero incidence. Both cases show that the trailing edge separation moves towards leading edge as incidences increase. Due to the strong 3-D LSB effects shown for the moderated aspect ratio rectangular wing planform at low Reynolds number, direct application of 2-D profile aerodynamics to MAV design has to be carried out with caution.

The wing planform effects on the aerodynamic performance have been studied for different wing planforms (Rectangular, trapezoidal, Zimmerman and inversed-Zimmerman wings). The leading edge sweep angle (for trapezoidal wing planforms) affects directly the stall behaviour. Higher leading edge sweep angle gives higher maximum lift, accompanied by a more sudden stall. The Zimmerman wing offers the best aerodynamic efficiency (CL/CD = 8) at the design condition, although it is longitudinally unstable. The inversed-Zimmerman wing shows a reduced level of aerodynamic efficiency as compared with the Zimmerman wing. However, it is longitudinally stable. The flow structure at the design condition shows that the shape of the leading edge separation bubble is directly related to the wing planform shape. Bubble with an elliptical shape is formed on both the upper and the lower surface for the Zimmerman and inversed-Zimmerman wing, whereas rectangular separation bubbles are formed on the trapezoidal and rectangular wing planforms.

The aerodynamic interaction between fuselage and wing planform has also been studied. The major effect from the fuselage is to degrade the overall aerodynamic performance. However, fuselage provides a positive contribution on the longitudinal stability, which shows a negative pitching moment slope at design condition.

The propeller slipstream effects have been studied for the overall aerodynamic performance. The maximum lift coefficient (CL,max) increases significantly. Asymmetric flow has been observed for the up- and down-going blade sides. The mutual influence between the propeller and the wing-fuselage has been addressed. Periodic forces and moments were produced by the propeller. These induced significant changes on the wing load coefficients from propeller slipstream have been clearly revealed. However, interaction effects also imply a modification of the slipstream axisymmetry due to the presence of the wing that, in turn, induces local modifications in the propeller rotation plane and changes the propeller performance. On the stability side, MAV models with/without the vertical stabilizer are all statically longitudinally stable (except at high propeller rotational speed, ω = 1131 rad/s, at α between -6° and 6°). At the design condition (ω = 555 rad/s), the longitudinal stability margin,, is increased to almost twice as large as the MAV without the propeller slipstream effect (from 8.5% and 15.5% for without/with propeller effect, respectively). The presence of the vertical stabilizer appears to have the positive contribution on the static lateral stability. In general, the MAV without the vertical stabilizer is statically laterally unstable (). However, MAV with the vertical stabilizer shows a positive yawing moment slope ().

Fluid-structure interaction has been investigated for a flexible wing to further understand the longitudinal stability margin and aerodynamic performance. The effect on minimizing the separation on the upper wing surface is studied, which shows an improvement of the CL, max for the MAV. The fluid-structure interaction results have shown that the wing deformation increases the camber, and shifts the maximum camber location towards the trailing edge. The substantial pressure redistribution increases both lift and drag coefficients, and a separation bubble is formed on the deformed wing near negative camber for low incidences (α between -3° and 0°). However, it increases the overall aerodynamic efficiency for α between 3° and 12°. The possible reason is that the upper surface separation region is reduced due to the wing deformation. The pitching moment coefficient curve drops more smoothly as incidence increases. The pitching moment at moderate to high incidences (9° < α < 21°) is nearly constant.

### Future work

1. More investigations on the propeller slipstream effects can be conducted on: a) locations of the propeller installation (i.e. the distance between the propeller and the fuselage); b) the propeller installation angle (i.e. for current model, the propeller is perpendicular with the fuselage), c) different propeller rotational speeds.
2. An optimization on the carbon fiber rod shape and locations would be very helpful for improving overall aerodynamic efficiency for the flexible model. Investigation on the diameter of the carbon fiber rods can also help on the deformation magnitudes (i.e. the current carbon fiber has a 3mm diameter ), or even chose a carbon fiber flat beam instead of using the rod, as this would reduce the form drag
3. Combine both fluid-structure interactions with the propeller sliding mesh (propeller slipstream effects). This would make the situation much closer to the real design condition, although the computational cost will be even more expensive.

# List of Publications

* + - 1. Chen, Z. J., Qin, N., and Nowakowski, A. "Numerical Simulations of Flow Separation on Cambered Plate Wing MAVs with Various Aspect Ratios," *RAeS Aerodynamics Conference*. Bristol, UK, 2010.
      2. Chen, J. Z., and Qin, N. "Propeller Effects on MAV Aerodynamics," *RAeS Aerodynamics Conference*. Bristol, UK, 2012
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