Computational investigation on aerodynamic and stability characteristics of a vertical take-off and landing micro air vehicle

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**NANYANG
TECHNOLOGICAL
UNIVERSITY**

**COMPUTATIONAL INVESTIGATION ON AERODYNAMIC
AND STABILITY CHARACTERISTICS OF A VERTICAL
TAKE-OFF AND LANDING MICRO AIR VEHICLE**

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SCHOOL OF MECHANICAL AND AEROSPACE ENGINEERING

2009

**COMPUTATIONAL INVESTIGATION ON AERODYNAMIC
AND STABILITY CHARACTERISTICS OF A VERTICAL
TAKE-OFF AND LANDING MICRO AIR VEHICLE**

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Abstract

This research focuses on the numerical investigation of the aerodynamic and stability characteristics of a Vertical Takeoff and Landing (VTOL) Micro Air Vehicle (MAV). The numerical investigation is divided into two parts: statics and dynamics. In the static analysis, the full configuration of the VTOL MAV as well as various design and operating parameters are investigated. The dynamic analysis is conducted to obtain the dynamic stability derivatives due to angular rate variations. Time and computational costs have always been the issue in the numerical investigation. As a solution, an innovative method termed the Improved-Component Buildup Method (I-CBM) is developed. The results obtained indicate that the VTOL MAV has favorable aerodynamic performances and static stability quality in the longitudinal and directional modes, although it tends to be unstable in the lateral mode. The results also show that the I-CBM provides fast and reasonably accurate prediction of the aerodynamic and stability properties of the MAV.

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List of Variables

Notation	Description	Unit
Alpha, α	Freestream angle-of-attack	$^{\circ}$ (degree)
Beta, β	Freestream sideslip angle	$^{\circ}$ (degree)
Cm-x, C_l	Rolling moment coefficient	
Cm-y, C_n	Yawing moment coefficient	
Cm-z, C_m	Pitching moment coefficient	
Re	Reynolds number	
Re _c	Reynolds number calculated using chord as the reference length	
AR	Aspect ratio	
cg	Center of gravity	
ac	Aerodynamic center	
O	In the order of	
C _L	Lift coefficient	
C _D	Drag coefficient	
A, S	Area	m ²
V	Velocity	m/s
u/U	Velocity parallel to the x-body axis of the vehicle at hover flight	m/s
ρ	Density	kg/m ³
$\dot{m}/m$	Mass flow rate	kg/s
μ	Viscosity of air	kg/(ms)
d	Diameter	m
b	Span	m
c	Chord length	m
γ	Setting angle of stator	$^{\circ}$ (degree)
δ	Setting angle of horizontal tail	$^{\circ}$ (degree)
ϵ	Setting angle of vertical tail	$^{\circ}$ (degree)
i	Setting angle	$^{\circ}$ (degree)
Ω	Angular velocity	rad/s, deg/s
ΔT	Time step size	s
n	Number of time steps	

Notation	Description	Unit
ϕ	Roll angle	° (degree)
θ	Pitch angle	° (degree)
ψ	Yaw angle	° (degree)
$C_{L_{max}}$	Maximum lift coefficient	
$\alpha_{C_{L_{max}}}$	Angle-of-attack corresponds to $C_{L_{max}}$	° (degree)
L/D	Lift-to-drag ratio	
$C_{l_{\beta}}$	Rolling moment coefficient variations with respect to sideslip angle	/deg
$C_{m_{\alpha}}$	Pitching moment coefficient variations with respect to angle-of-attack	/deg
$C_{n_{\beta}}$	Yawing moment coefficient variations with respect to sideslip-angle	/deg
C_{l_p}	Damping-in-roll derivative	
C_{l_r}	Rolling moment coefficient due to yawing derivative	
C_{m_q}	Damping-in-pitch derivative	
C_{n_r}	Damping-in-yaw derivative	
C_{n_p}	Yawing moment coefficient due to rolling derivative	

Subscript Notation:

∞	freestream
local	local
max	maximum
L	Left part/plane
R	Right part/plane
W	Wing
HT	Horizontal tail
VT	Vertical tail

Superscript Notation:

*	correction
#	calibration

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Chapter 1 Introduction

1.1 Background

One hundred years after the Wright Brothers' airplane, the Wright Flyer I, flew over Kill Devil Hills, mankind is still in the midst of a fascinating quest to conquer the sky. Since that historical day in 1903, much advancement in aeronautical engineering has been achieved. Intercontinental and supersonic flights are just a few breakthroughs which become the hallmarks of the aviation history.

The exponential development of aeronautical engineering is mainly driven by the aim of building an airplane which is able to fly faster and higher than its ancestors. Faster and higher? How about smaller? It was not until about twenty years ago that people started to put intensive efforts in developing a small/compact unmanned airplane for multiple applications. Pioneered by the U.S., researches are conducted to design, build and operate small airplane which is later termed as Micro-Air-Vehicles (MAV).

MAVs were first defined by DARPA (Defense Advanced Research Projects Agency) as six-degree-of-freedom machines capable of performing multiple tasks, including reconnaissance, surveillance, targeting, tagging and biochemical sensing [1]. Based on DARPA definition, MAVs should have a maximum dimension of 15 cm (6 in) and be able to stay aloft for 20–60 minutes while carrying a payload of 20 grams or less to a distance of approximately 10 km. Despite DARPA definition above, an aerial vehicle with a mass of less than 500 grams and maximum length less than 50 cm is still categorized as MAVs [2].

Researches on MAVs are currently focused on military applications as an effort to provide lower echelon soldier with a system to collect real-time and accurate information on the battlefield. The Honeywell MAV [3], for example, was deployed in Iraq and proven to be reliable for operations in severe weather conditions. Outside their current military applications, the possibilities for MAVs to be used for civil applications are wide open. Several possible applications are meteorology/environmental research, urban area monitoring, traffic observations, search and rescue services, and forest monitoring. The size of MAVs makes it suitable to perform mission in limited airspace such as downtown area, forests, caves and tunnels. The autonomous feature of MAVs is critical for mission in hazardous environment. In

dangerous conditions such as nuclear contamination or natural disasters, quick and accurate monitoring can be done without endangering human life.

One parameter that is usually used to identify airplane operating conditions is the Reynolds number (Re), which is defined as $Re = \frac{\rho V_c c}{\mu}$, where ρ is air density, V_c is cruising speed, c is wing chord and μ is air viscosity. Commercial airplanes such as Boeing 747 and Airbus A320 fly at $Re_c = O(10^7 - 10^8)$. MAVs with a maximum dimension of 15 cm and V_c of 10 – 20 m/s fly at $Re_c = O(10^5)$ which is termed as low Re regime.

The development of MAVs is hindered by the lack of accurate models that can predict the aerodynamic properties in low Re regime. Classical aerodynamic theories are accurate for high Re but generally not applicable for low Re because of the difference in the flow physics. In the low Re regime, there can be complex flow phenomena such as laminar separation bubbles and tip vortex, which are not usually found in the high Re regime.

The lack of analytical/empirical models for this particular regime has put numerical analysis as an important method to investigate the aerodynamic properties of MAVs. Computation resources have become powerful yet affordable to simulate various flow field cases. Previous researches have shown that the numerical results are sufficiently accurate such that they can reliably be used in the design and optimization process of MAVs [4, 5].

1.2 Objectives

Providing numerical analysis of aerodynamic and stability characteristics of a Vertical Takeoff and Landing (VTOL) MAV is the main objective of this research. A number of simulations are conducted to achieve this objective. The results obtained are analyzed to generate a good understanding of the flow field around the MAV as well as to examine its stability properties.

The numerical data obtained from this study is the main tool to study the VTOL MAV design due to the unavailability of the experiment result. This numerical analysis is intended as a guide the iterative design process of the VTOL MAV by providing accurate data to validate the design with respect to the vehicle's requirements.

Additional objective of this research is to develop a method for fast predictions of the aerodynamic and stability properties of the VTOL MAV. The new method is expected to reduce the complexity of the numerical investigation, thus minimizes the computational costs. This method is also intended to be used throughout the design project to accommodate any design modification. So that, the next phase of the project will excel in terms of effectiveness and efficiency, particularly in numerical analysis.

1.3 Scope

This research focuses on the numerical investigation of the flow over a specific configuration of VTOL MAV. The MAV's ability to fly in horizontal and vertical directions makes it necessary to simulate the flow for a wide-range of angles-of-attack (α) and sideslip-angles (β). Simulations are conducted to check the MAV aerodynamic performances as well as its stability properties. The flight envelope of the vehicle is also investigated by testing various operating parameters. The computations are performed by considering steady flow conditions around the vehicle, except for the dynamic stability analysis where an unsteady simulation is necessary. The dynamic stability analysis of the vehicle is performed using the sliding mesh technique.

Throughout the computational process, time and computational costs are the main issues. For that reason, an innovative method is developed for fast prediction of aerodynamic forces and moments of the VTOL MAV. This method is built based on the Component Buildup Method (CBM), which is commonly used in the conceptual design phase of conventional aircraft [6]. The proposed method, the Improved-Component Buildup Method (I-CBM), is thoroughly examined in all flight-modes to formulate a complete approach that can be used in the conceptual design phase of winged-ducted propeller VTOL MAVs.

1.4 Methodology

In this work, the aerodynamic computations are performed using a commercial CFD code-package which has been used widely for design analysis of aircraft. The geometry of the model and grid domain are built using Gambit 2.3.16 and saved as a mesh file. This file is exported to Fluent 6.2.23 and then solved according to the defined case setting.

First, the CFD code is tested to examine its suitability in solving a particular case. In this research, Fluent is tested in low Reynolds number regime, which is well known for its complexity. The

NASA annular airfoil is chosen as the test/validation case because of its similarity to the VTOL MAV ducted propeller design [7].

The next step is to simulate the propeller system used in the VTOL MAV. The estimation of the propeller performance is critical because it determines the flight capabilities of the VTOL MAV. The momentum theory is used to model the thrust produced by the propeller, which has been tested in static conditions. The propeller is modeled as an actuator disc where a sudden increase of pressure occurred and spread uniformly over the disc's surface.

The MAV is ready to be simulated, once the code has been validated and the propeller system has been calibrated. The numerical investigation of the VTOL MAV is divided into two parts: statics and dynamics. In the static analysis, the full-vehicle computational model is simulated to predict the aerodynamic and stability properties of the VTOL MAV. These computational models are later modified to analyze the lifting/control surface effects and the operating conditions of the MAV. In the dynamic analysis, the full-vehicle model is tested under two different angular velocities for each moment coefficient evaluated. The moment coefficients are plotted with respect to the angular velocities, and the dynamic stability derivatives are determined from the curve-slope.

The VTOL MAV aerodynamics can be computed fully or per component. Full configuration simulations are too complex and too costly for the design parametric study. The component/individual simulations (CBM) are less complex and less costly than the full configuration. However, the individual simulations neglect the interference effects among the MAV's components which can introduce inaccuracies in the results. Another computational model of coupled duct and wing is also created to study the interference effects between the two components. For the I-CBM development, these computational models are tested for low angles-of-attack ($0^\circ - 20^\circ$) and low sideslip-angles ($-10^\circ - 10^\circ$) to analyze both the longitudinal and the lateral-directional modes. The simulation results of the individual, coupled duct-wing and full configuration models are compared and analyzed to find a general relationship among them. The general relationship of different components is then synthesized to build the complete I-CBM.

1.5 Outline

This report consists of six chapters and six appendices. The contents of each chapter and appendix are described briefly as following:

- **Chapter I Introduction**

In this chapter, the background, scope, objectives and methodology used in the research are described. A systematic outline of the report is also given to guide the readers.

- **Chapter II Literature review**

Literature study is an important part of the research. Learning from literature helps to understand the research topic, to solve problems and to analyze findings/results. A comprehensive summary of the literature study is given in this chapter.

- **Chapter III Computational Approach**

The computational approach used in the static and dynamic analysis of the VTOL MAV is described in this chapter. The geometry of various computational models, grid domain, and problem setting are explained in detail here.

- **Chapter IV Numerical Analysis of Aerodynamics and Stability of the VTOL MAV**

The numerical results of the static and dynamic analysis are presented and analyzed in this chapter. The static and dynamic stability derivatives are deducted from the aerodynamic forces and moments predictions obtained from both analyses.

- **Chapter V Improved-Component Buildup Method (I-CBM) for Fast Prediction of the Aerodynamic Properties and Stability Derivatives of the VTOL MAV**

The background, proposed method, computational modeling, validation, results and analysis of the Improved-Component Buildup Method (I-CBM) are explained in this chapter.

- **Chapter VI Conclusions and Future Works**

Conclusions from the current research project and future works are written in this chapter.

- **Appendix**

- A. Computational results of the aerodynamic and static stability analysis of the VTOL MAV-baseline
- B. Computational results of the aerodynamic and static stability analysis of the VTOL MAV
- C. Computational results of the aerodynamic and static stability analysis of the lifting/control surfaces
- D. Computational results of the effect of the VTOL MAV operating conditions
- E. Computational results of the dynamic stability analysis of the VTOL MAV
- F. Computational results of the Improved-Component Buildup Method (I-CBM)

Chapter 2 Literature Review

2.1 Full configuration of the VTOL MAV

This project is part of the development of a MAV sponsored by Defense Science Organization (DSO). The conceptual design phase of the MAV is guided by the Design Requirement and Objective (DRO). In general, the DRO states that the MAV shall capable of Vertical Takeoff and Landing (VTOL) and hover, flying for medium duration and range (target endurance of 20 minutes, range of 1 km and maximum altitude of 500 m), pack-able in a small briefcase/backpack, and easy to operate. The MAV is designed by considering all possible operating conditions. The final configuration is chosen after careful examination on four different types of MAVs; fixed wing, rotary wing, ducted fan, and flapping wing (Figure 2-1)[8].

Figure 2-1 Black Widow, Vertigo, Microbat, and iSTAR 29i [9-11]

The ducted fan configuration is finally chosen because it has unique abilities of vertical takeoff and landing (VTOL) and hovering flight. These features support a quick and easy operation of the MAV since it does not need a special takeoff space or launching requirement. The duct which enclosed the propeller offers protection from foreign-object-damage (FOD) and radar detection. A pair of wings is added to the duct, forming a winged-ducted propeller configuration which is uncommon for MAVs. Most published-designs of MAVs such as Colorado [4] and Carolo [12] are fixed wing, whereas others such as iSTAR [13] and Fantail [14] use the ducted propeller configuration with no wings.

The complete configuration of the VTOL MAV also includes stators, payload box, horizontal and vertical tails (Figure 2-2). The stators are used to counter the rolling moment caused by the engine torque. The horizontal and vertical tails, act as the longitudinal and directional control surfaces. The tails also act as stabilizer when their geometry and position are carefully designed. At the front part, a bullet-shaped payload box is installed as storage for camera, battery,

transmitter, and/or navigational equipment. The detail size and geometrical properties of the VTOL MAV are given in Section 3.2.1.

Figure 2-2 The configuration of the VTOL MAV [8]

2.2 Aerodynamics of MAVs

2.2.1 Flow of low Reynolds numbers regime

The VTOL MAV considered here has a winged-ducted propeller configuration with a wing's chord root of 25 cm and a design cruising speed (V_c) of 15 m/s. The size and operational airspeed of the VTOL MAV correspond to low Re regime. Thus, the flow around the MAV is viscous and involves laminar, transitional and turbulent boundary layers. This regime is known for its complexity where flow phenomena such as laminar separation bubbles and wing tip vortices may occur.

Laminar separation bubbles (Figure 2-3) are formed by the flow that proceeds in the tangential direction of the surface after separation occurs. At low Re, the flow is separated prior to the transition to turbulent. The circulation region of the detached flow is enclosed by the laminar shear layer, which is unstable. The layer instability increases during transitions to turbulent flow, which has higher energy to reattach to the surface. The precise condition for the occurrence of separation, transition, and reattachment is influenced by many factors such as Re, pressure distribution, surface roughness, and freestream turbulence.

At small angles-of-attack, a closed separation bubble is formed on certain parts of the lifting surfaces. As the angle-of-attack increases, these closed bubbles open up and create a large separated region over the lifting surfaces. This phenomenon results in significant loss of lift and eventually leads the lifting surfaces to stall. At stall and at angles-of-attack beyond stall, three dimensional separations and various types of rolled-up vortices may be generated [15].

Carmichael [16] explains that an airfoil with chord-Re (Re_c) less than 5×10^4 will not experience laminar separation bubbles because the flow does not have enough energy to reattach to the surface after separation occurs. Parallel to the freestream flow, laminar separation bubbles become shorter as the Re increases. In our case, where Re_c of the wing is from 1×10^5 to 2×10^5 , short bubbles are expected.

Figure 2-3 Schematic of laminar separation bubbles [17]

In the 3D flow over a wing, tip vortices are formed by the flow movement from the higher pressure region at the lower surface of the wing to the lower pressure region at the upper surface. The flow at the upper surface is bent inward (Figure 2-4a), whereas the flow at the lower surface is bent outward (Figure 2-4b). This flow is pushed downstream by the freestream flow, rolls-up, and forms the tip vortex (Figure 2-5). Tip vortex creates downwash flow which reduces the effective angle-of-attack of the lifting surfaces. For low aspect ratio wings, tip vortices can cover large part of the lifting surfaces and thus significantly deteriorate the performance of the airplane.

Figure 2-4 Flow over a wing: upper (a) and lower (b) surface [18]

The influence of tip vortex on the aerodynamics of MAVs is studied by many researchers. Viieru et al. [19, 20] study the influence of tip vortex on the membrane wing of the University of

Florida MAV and also the effects of adding end-plate at the wing tip. Ramasamy [21] investigates the flow field of a small rotor which is commonly used in a MAV configuration. One of the parameters analyzed in this work is Figure of Merit (FM) of rotors operating at various Re_c flows. Figure of Merit (FM) is defined as the ratio between the ideal power and the non-ideal power used to generate a specific amount of thrust and used to measure the rotor's efficiency in hover. The highly-viscous low Re_c flow result in lower FM of approximately 50% compared to 80% for rotors operating at higher Re_c flow.

The current research which incorporates the actuator disc model cannot be compared to Ramasamy's work because it is experimental and the analysis is based on the actual flow field over a small rotor. Nevertheless, the flow visualizations give information about the size, thickness and turbulence level of the tip vortex and vortex sheets behind the rotor-blade. Other works which focus more on low aspect ratio wings are explained in the next section.

Figure 2-5 Wing tip vortices [4]

VTOL MAVs operate in a wide range of angles-of-attack, from 0° to 90° . The flight at high angles-of-attack results in complex three-dimensional phenomena such as secondary vortices and separated flows. Rom [15] claims that such complicated three-dimensional flows with strong viscous-inviscid interactions are unsolvable by any available theoretical analysis or even a complete numerical solution using Navier-Stokes equation. The flow over VTOL MAVs combines the features of low Re and high angles-of-attack regime. Consequently, accurate analysis of the flow over VTOL MAVs should be based on wind-tunnel and flight tests. However, numerical analysis is still capable of providing results which are within the acceptable accuracy.

2.2.2 Flow around low-aspect-ratio wings

The flow around MAVs, which usually have low aspect ratio wings, is dominated by tip vortex. This vortex covers a large part of the wing and the other lifting surfaces, creating a low pressure region at the upper surface of the lifting surfaces, which consequently increases the lift-curve slope proportionally to angles-of-attack, and contributes as the source of nonlinear lift. Early works, both experimental and computational, show these characteristics and also provide the flow visualization.

The tip vortices modify the local flow by creating an induced vertical component of the velocity (Figure 2-6). This velocity reduces the local angle-of-attack and results in a component of the total force on the wing in the drag direction known as induced drag [20]. As the aspect ratio decreases, the drag becomes higher and the lift becomes lower. This leads to the poor aerodynamic efficiency of the low aspect ratio wings.

Figure 2-6 Force diagram of induced drag [20]

Ramamurti [22] tests four different platform-shapes in the wind-tunnel. The experiments are conducted on each platform for three different aspect ratios and Reynolds numbers. The experimental results show that the lift curve becomes increasingly nonlinear as the aspect ratio decreases. This conclusion is supported by Cosyn [23] who numerically investigates low aspect ratio wings at low Re. The author also develops a number of methods to predict the drag and lift of profiled (S5010) and flat/thin low aspect ratio wings in low Re regime.

More comprehensive results are found in Torres and Muller [24] who test four platform-shapes for seven different aspect ratios. It is found that aspect ratio is the most important parameter affecting the aerodynamic characteristics of low aspect ratio wings at low Re. Torres and Mueller suggest aspect ratio of 1.25 as the separator value between the lower and upper class of the low aspect ratio wings category in low Re regime. The next important parameters are the wing

platform and Re respectively. This research also highlights the high value of $C_{L_{max}}$ and stall angle-of-attack $\alpha_{C_{L_{max}}}$ as important characteristics of low aspect ratio wings operating at low Re (Figure 2-7).

Figure 2-7 Nonlinear lift-curve and high value of $C_{L_{max}}$ and $\alpha_{C_{L_{max}}}$ [24]

The influence of Re to the aerodynamics of low aspect ratio wings are explained in Pelletier [25]. Three different shapes of wing are tested in the wind-tunnel for a wide range of Re . It is found that $C_{L_{max}}$ increases with Re and aspect ratio in the range of Re tested. It is also found that $C_{D_{min}}$ increases with the decrease of Re . This paper provides suitable comparison data for the current research since one of the Re and aspect ratio combination tested is very similar to the VTOL MAV.

Computational results from Gyllhem [4], Schroeder [26], Guerrero [10] and Brion [5] support the previously mentioned experiments by showing nonlinear lift-curve and high stall angle. Numerical analysis is superior to experiments for its flow visualization capability. Using two different codes, the first and second works are able to capture the wing tip vortices. However, laminar separation bubbles are only observed in the first work.

Most of the works mentioned above provide the lift and drag data, but give less attention to the moment. Only Pelletier [25] and Brion [5] give the pitching moment variations to angle-of-attack. Another limitation of these works is that the researches are limited to small range of angles-of-attack, from 0° to 40° or 50° . Aerodynamic characteristics of wider range of angles-of-attack are important for VTOL MAVs. These data are usually proprietary and not widely available due to the military aspect of MAV applications. Some data are given in Brion [5] who developed a code to investigate the aerodynamics of iSTAR 29i.

2.2.3 Ducted Propeller

A propeller blade consists of a series of airfoil cross-section which may vary in chord-length and angle from root to tip. The propeller converts the engine power into thrust by imparting velocity to a mass of air and pushes it backward. The thrust produced by the propeller is proportional to the air mass and the induced velocity. A configuration is called ducted/shrouded propeller when the propeller is enclosed by a duct/shroud. There might be confusions in differentiating ducted-fan and ducted-propeller. The common parameter which is used in defining the two configurations is blade density. Blade density is the ratio between the total blade area and the frontal projection of the shroud/duct area. An open propeller may have a blade density of up to 0.4, while a ducted fan has a blade density close to 1.0 [27].

In a ducted propeller configuration, air is drawn into the duct and divided at a stagnation point. The airfoil cross-section of the duct generates aerodynamic forces with the resultant force works perpendicular to the airflow provides additional lift. In transitional flight, the upper part of the duct operates at higher angle-of-attack, whereas the lower part operates at lower angle-of-attack. This condition provides combination of higher and lower magnitude of lift which results in a net increase in the duct lift.

Adding a duct or shroud to a propeller increases the flow rate entering the propeller. This high flow rate results in an increase in thrust particularly during static conditions. If it is designed carefully, the combination produces higher lift due to the interference effects between the duct and the propeller. The duct/shroud protects the propeller from any damage caused by foreign fragments. It also suppresses the noise of the propeller and motor; which is very desirable.

The flow field over a propeller in cruise conditions can be divided into three regions; inflow, outflow, and freestream (Figure 2-8a). In static air mass, the flow field can be divided into two regions only, inflow and outflow (Figure 2-8b). Flow from the outflow must go to the propeller disc before making a sudden turn to enter the slipstream. This flow physics results in reversed flow near the propeller tip which reduces the effective disc area and the amount of flow that can enters the propeller [28].

Figure 2-8 Flow field around open propeller at cruise (a) and at static condition (b) [28]

The flow physics around a propeller change dramatically when a duct is added around it. The flow can still be divided into two regions, inflow and outflow, but there is the duct as a solid separator (Figure 2-9). The flow from the outflow to the inflow is guided by the duct to avoid instant turning and acceleration. The duct minimizes the reversed flow near the propeller tip and optimizes the disc area of the propeller.

Figure 2-9 Flow field around a ducted fan/propeller [29]

Early experimental work is done by Parlett [30] to investigate the effects of airspeed and angle-of-attack on the lift, drag, and pitching moment of a shrouded-propeller model. The model is tested over an angle-of-attack range of 0° to 90° with and without the propeller operating, and also with the propeller removed. These data are important to study the effects of the propeller on the whole configuration.

Abrego [31] conducts experiments to investigate the effects of the duct angle-of-attack, the exit vane flap length, the flap deflection angle, and the duct chord length. One of the study conclusions is that extending the duct chord length does not significantly change the duct thrust coefficient.

A computational analysis on ducted fan is done by Chang [32], while Lind [33] investigates a ducted rotor model. Chang's research supports the theory that the combination of duct and fan produces higher thrust than the one produced by each individual component. The simulation results of the five-bladed Trek Aerospace's ducted-fan show that ducted fan produces more than twice the thrust produced by the duct or fan alone.

Lind validates a set of experimental data using the panel method VSAERO which is combined with ROTOR propeller module. The simulation results support the research findings of Parlett [30] that from the angle-of-attack of 0° to 90° , the drag and the lift increase with the increase of airspeed although for pitching moment this relationship is not observed. The pitching moment data have an irregular pattern which is possibly caused by the difficulty in accurately measuring the small values of moments.

Some analytical theories of the ducted propeller are given by Kriebel [34] and Phillips [35]. In the first work, the propeller is assumed as a uniformly loaded actuator disc, while the duct is modeled as a thin cylinder. Other assumptions taken are that the flow is potential and steady, the actuator disc vorticity is constant, and that the slipstream velocity is large compared to the cross-flow component of the freestream. This paper gives an analytical theory to calculate the forces, moments, and stability derivatives as a function of the angle-of-attack. Clear explanations about the components of thrust and the effects of camber/taper/thickness of the duct are also given.

Phillips studies the effects of the slipstream rotation on the induced velocity and propulsive efficiency. It is found that the effects is only in the order of 5% or less for the range of thrust coefficient and advance ratio normally encountered by airplane's propellers. This finding is used as the basis for neglecting the slipstream rotation of the VTOL MAV propeller model. The author also gives an equation which relates the induced velocity and the propeller rotational speed.

The propeller in this research is modeled as an actuator disc where the pressure jump spreads uniformly over the disc's surface. The propeller is validated using the momentum theory taken from Seddon [36]. The validation process will ensure that the actuator disc is able to represent the same flow effects as the actual propeller. The equations used in the momentum theory are explained next.

The propeller model is also calibrated by comparing the static thrust produced by a ducted and a free propeller. An equation, analytically derived from the momentum theory, gives the ratio of the static thrust. Equation 2-1, which is taken from Reference [37] gives the relationship among the static thrust of the ducted propeller ($T_{ducted_propeller}$), the static thrust of the un-ducted/free propeller ($T_{propeller}$), the propeller disc area (S_2) and the duct outlet area (S_3);

$$\frac{T_{ducted_propeller}}{T_{propeller}} = 1.26 \left(\frac{S_3}{S_2} \right)^{\frac{1}{3}} \quad (2-1)$$

The detail of the computational model and the procedure of the validation are explained in Section 3.1.5.

2.2.3.1 Momentum theory in hover mode

The propeller is modeled as an actuator disc with a uniformly spread pressure jump. In hover mode, the flow over the propeller can be divided into one stream tube above and below the disc (Figure 2-10). The air outside this stream tube is undisturbed and no rotation is included in the flow.

Figure 2-10 Actuator disc concept for rotor in hover [36]

In incompressible flow, Bernoulli equations, applied before (Equation 2-2) and after the disc (Equation 2-3), gives two equations that lead to the pressure jump equation (Equation 2-4).

$$\text{Inflow: } p_{\infty} = p_i + \frac{1}{2} \rho v_i^2 \quad (2-2)$$

$$\text{Outflow: } p_i + \Delta p + \frac{1}{2} \rho v_i^2 = p_{\infty} + \frac{1}{2} \rho v_{\infty}^2 \quad (2-3)$$

$$\Delta p = \frac{1}{2} \rho v_{\infty}^2 \quad (2-4)$$

The air is accelerated by the propeller from zero velocity at “upstream infinity” to a value of induced velocity (v_i) at the disc before it rises further to a value of freestream velocity (v_∞) at the “downstream infinity”. As the velocity increases upstream of the disc, the pressure drops from p_∞ to a value of induced pressure (p_i). An increase of pressure (Δp) occurs at the disc, before the pressure drops again to a value of freestream pressure (p_∞) as the propeller effect diminishes downstream.

The **thrust** (T) on the disc can be expressed as the overall rate of increase of the axial momentum of the air,

$$T = \rho A v_i v_\infty = \dot{m} v_\infty \quad (2-5)$$

In Equation 2-5, ρ is the air density, A is the disc area, while $\dot{m}$ is the mass flow rate of the air passes the disc surface.

The **pressure jump** (Δp) is thrust over an area of the disc,

$$\Delta p = \frac{T}{A} = \rho v_i v_\infty \quad (2-6)$$

From Equations 2-4 and 2-6, the relationship between the induced (v_i) and downstream velocity (v_∞) can be obtained:

$$v_\infty = 2v_i \quad (2-7)$$

Then the **thrust** (T) equation (Equation 2-5) becomes:

$$T = 2\rho A v_i^2 = 2\dot{m} v_i \quad (2-8)$$

The **induced velocity** (v_i) is obtained from Equation 2-8,

$$v_i = \sqrt{\frac{T}{2\rho A}} \quad (2-9)$$

2.3 Stability

Stability is of high importance for a MAV due to its size and operating conditions. Its small size makes it susceptible to disturbances, and simultaneously only leaves a small margin of error for stability. In this research, the CFD code is used to obtain the aerodynamic performances and the static and dynamic stability derivatives of the VTOL MAV. The numerical results are then analyzed to determine the stability quality of the VTOL MAV and to supply the input parameters

for flight simulations. This approach has been used in previous researches on fixed and flapping wing MAVs, however limited information is found on the application for rotary wing and ducted-fan MAVs.

2.3.1 Definition and criteria

Static stability is a property of an equilibrium state [38]. An airplane is said to be in a state of equilibrium when the resultant force and moment about the center of gravity are zero. An airplane is statically stable if it has the tendency to return to its equilibrium state after a disturbance.

The motion of an airplane can be divided into lateral, longitudinal and directional modes. Accordingly, the static stability properties of an airplane is divided into three; longitudinal, lateral and directional static stability. An airplane is said to have longitudinal static stability when it develops a restoring pitching moment after a pitching disturbance. Similar to longitudinal, lateral and directional static stability are possessed by an airplane if it develops a restoring rolling and yawing moment due to rolling and yawing disturbances, respectively. These definitions are translated into static stability criteria of each mode; $\frac{dC_m}{d\alpha} < 0$ for longitudinal, $\frac{dC_l}{d\beta} < 0$ for lateral, and for directional is $\frac{dC_n}{d\beta} > 0$. In these criteria, C_l is rolling moment, C_m is pitching moment, C_n is yawing moment, α is angle-of attack, and β is sideslip angle.

There are three important derivatives for determining the dynamic stability of an airplane. The first derivative $\frac{dC_l}{dp}$ is termed as damping-in-roll. The other two are $\frac{dC_m}{dq}$ and $\frac{dC_n}{dr}$, which are known as damping-in-pitch and damping-in-yaw derivatives. These derivatives are called damping because they describe the vehicle resistance to roll rate (p), pitch rate (q) or yaw rate (r) disturbances. An airplane is dynamically stable if these three derivatives are negative. A dynamically stable airplane tends to generate rolling, pitching or yawing moments to recover from the corresponding roll, pitch or yaw rate disturbances.

2.3.2 Stability of MAVs

In general for longitudinal static stability, two important parameters in the stability calculation are aerodynamic center and center of gravity [39]. For stability of a conventional airplane, the

center of gravity must lie between the aerodynamic center of the wing and the tail. Tail fraction is also an important parameter in stability. Iterative process is usually needed to find the most optimum tail fraction to achieve good stability quality.

Small airplanes like MAVs tend to have higher dynamic frequencies [40]. Some examples of MAVs which show these characteristics are Black Widow [9] from Aerovironment, Inc. and Zagi and Dragonfly from the University of Arizona [41]. MAVs are also more susceptible to wind gust due to the low wing loading and low speed flight. Stability augmentation systems such as yaw/roll damper and autopilot such as altitude-holding are usually installed to improve the stability of MAVs.

Published works in this area are mostly about fixed-wing UAVs/MAVs. In these researches, the wing of the MAVs is modeled as thin lifting surface which operates at low angles-of-attack. These assumptions allow for the utilization of the vortex-panel-method based software such as AVL (Athena Vortex Lattice) [42] and AAA (Advanced Aircraft Analysis) [41] to predict the stability derivatives. The configuration and operating conditions of the VTOL MAV are not within the capabilities of the above mentioned software. To overcome these issues, Fluent is used to predict the static stability derivatives by varying the angle-of-attack and the sideslip angle of the vehicle.

2.4 Computational Fluid Dynamics (CFD) analysis methodology

The computational analysis is conducted using a commercial-code package, Fluent and Gambit. This code package has been used widely in educational and industrial applications. Several examples on application of Fluent in MAV research are found in Cosyn [23], Brion [5] and Boughton [40]. For low angles-of-attack, numerical results from these researches are in good agreement with experiments. Brion also uses the code to obtain information about the pitching moment, the interference effects between the wing and the fuselage, and the propeller effects on the flow over a fixed-wing MAV. These earlier works provide some insight about the accuracy, ability or limitation of the CFD code in analyzing the aerodynamics of MAVs.

2.4.1 Simulation Procedure

The simulation procedure of this research is divided into three stages; pre-processing, calculation, and post processing. In the pre-processing stage, all the parameters and boundary

conditions are set. This process, termed problem-setting, has a significant influence on the validity and quality of the numerical results.

The first step in the pre-processing stage is to draw the geometry. The drawing can be imported from CAD software such as SolidWorks® or CATIA®. The CAD software provides more flexibility in drawing compared to Gambit, although “healing” is usually required to create a mesh-able domain from the imported drawing. Grid generation is the most difficult and time-consuming process. This process needs to be done carefully to ensure an accurate and cost-effective computation. The grid/mesh file is then exported to Fluent where the case is set. The type of solver, turbulence model, pressure-velocity coupling, discretization scheme, operational conditions and reference values are examples of the variables to be set before running a simulation in Fluent.

The turbulence model which is an important parameter in the calculation stage is chosen based on prior research results. The computational works on low aspect ratio or low Re which have been mentioned in the previous chapters are the sources of information about the turbulence model. The work of Godin [43] provides the comparisons between one- and two-equation turbulence model. His work is important for this research because it explains some of the characteristics of the Spalart-Allmaras turbulence model which is used in the VTOL MAV numerical analysis. This work is the starting point to choose the optimum turbulence model for the current research. Specific study on high angles-of-attack is conducted by Shelton [44] who tests four turbulence models, including the Spalart-Allmaras, for airfoil analysis. Some results in this paper are used as the references in analyzing the simulation results of VTOL MAV as they give information about the behavior of each turbulence model in the high angles-of-attack regime.

In the calculation process, Fluent solves the flow governing equations and the turbulence equations. The aerodynamic forces and moments or parameters such as mass flow rate can be monitored through the display window. Monitoring these parameters is important to ensure that a steady solution is obtained. There is a chance that the residual has already met the convergence limit but the solution is not yet steady. A steady solution is usually achieved when a lower residual criterion is used. The number of iterations increases with the decrease of the residual criterion; therefore the solution can reach steady condition during these additional iterations.

The most challenging step in the post-processing stage is analyzing the results. Analysis is required to determine the conformability between the physical flow field and the numerical results. In this step, the data are processed and then displayed as curves, graphs, pictures or animations. Manual processing to get more specific data is usually needed, unless Fluent provides that particular data display-feature. A post-processing program, Fieldview®, is also used for simple and quick making of various displays.

2.4.2 Grid domain

Both structured and unstructured grids are utilized in the computations. Structured grid is generated by partitioning the computational domain into smaller cubes to enable meshing using hexahedral grid. This type of grid requires more “manual” works, but it gives the user more control in creating and distributing the grid near the surface of the object. Moreover, less amount of grid is needed to give results of the same accuracy as the unstructured grid. The distribution of unstructured grids is more difficult to control, but the shape of the grid makes them more suitable to mesh a complex object.

The unstructured grid can be automatically generated using a feature called size function. There are four types of size function: fixed, curvature, proximity and meshed [45, 46]. The fixed size function results in a constant value of mesh size on the source entity. The curvature size function determines the mesh sizes on the source entities with respect to the degree of the surface curvature. The proximity size function specifies the mesh sizes relative to the closeness of entities. The meshed size function has the ability to generate grid using the pre-meshed source entities. This size function radiates the mesh sizes from the pre-meshed source entities to the attached entities [46]. The grid generation of a complex geometry becomes easier using this feature because users can create a volume (3D) mesh using surface, edge or vertex grid. Users only need to determine the source and the attached entities, the growth rate, the size limit of the grid and several other unique parameters for each size function. Although this meshing tool is almost automatic, users still play a big role in the grid generation process. Users must carefully determine the combinations of the four different size functions and their parameters to be able to create a good quality grid-domain. The “limitless” possible combinations are the source of problems in using these size functions.

2.4.3 Case setting

As mentioned before, there are several variables which must be set in the pre-processing step. The type of solver, turbulence model, pressure-velocity coupling, discretization scheme and operational conditions are some of those variables. The operating conditions for the validation case are explained in Section 3.1.2, while for the VTOL MAV case, the operating conditions refer to standard sea-level.

There are two types of solver available in Fluent: segregated and coupled. The coupled solver is recommended for a case with strong interactions between density, energy, momentum and/or species such as high speed compressible flow [47]. The segregated solver is preferred in all other cases because it consumes less memory and more robust compared to the other solver. Implicit and explicit approaches available for coupled solver, while for segregated only implicit approach can be used. The implicit approach usually leads to faster convergence; however it consumes more memory than the explicit one.

Figure 2-11 Simulation procedures in the CFD code [47]

The turbulence models for steady simulations in Fluent are laminar, Spalart-Allmaras (one-equation turbulence model), $k-\epsilon$ and $k-\Omega$ (two-equation turbulence model), and RANS (Reynolds Average Navier Stokes). Based on the literature review, Spalart-Allmaras is used in both the

validation and VTOL MAV cases. Detail explanation about these turbulent models can be found in many Computational Fluid Dynamics (CFD) references [48, 49].

The pressure-velocity coupling used in the simulations is SIMPLE (Semi-Implicit Method for Pressure-Linked Equations). The detail about this algorithm can be found in Reference [50]. In brief, the solution procedure of the SIMPLE algorithm is as following:

- 1) Make an initial approximation for pressure $(p^*)^n$ and velocity $(V^*)^n$ at time level (n)
- 2) Solve the momentum equations to obtain the velocity at the next time level (n+1)
- 3) Use the results from step two as the inputs for the pressure correction equation to calculate the p' (pressure correction coefficient)
- 4) Calculate the pressure at n+1, p^{n+1} , using this equation $p^{n+1} = (p^*)^n + p'$ (2-10)
- 5) The value of p^{n+1} (step four) is used as an input for the step two in the next approximation. Furthermore, steps two to five are re-iterated until convergence is reached.

Figure 2-12 flow chart of the SIMPLE algorithm procedure

The second order upwind is used in all simulations. The detail of this scheme can be found in Reference [50, 51], therefore only a brief explanation is given here. Using the upwind scheme, up-winding in brief, properties of a cell is derived from the properties of cells upstream, relative to the normal velocity. The first order of the wave equation is taken as an example to explain the up-winding concept (Equation 2-11). A value of wave velocity in the x-axis direction is u , c is the speed of sound, t is time, while x is the direction of wave propagation.

$$\frac{\partial u}{\partial t} + c \frac{\partial u}{\partial x} = 0 \quad (2-11)$$

If c is positive, this equation represents wave propagation in the x^+ direction. The shock wave causes a discontinuity of velocity (u). As a result, flow properties of the grid point i depend only on the properties of the grid point $i-1$. In the opposite, properties of the grid point $i+1$ influence i when the central difference scheme is used. In the case of shock wave, this can creates solutions instability or blow up.

Figure 2-13 wave propagation in the x^+ direction

Up-winding results in the equation which only includes grid points $i-1$ and i (Equation 2-12). The wave velocity at grid points $i-1$ and i is u_{i-1} and u_i respectively, whereas Δx is the grid spacing.

$$\frac{\partial u}{\partial x} = \frac{u_i - u_{i-1}}{\Delta x} \quad (2-12)$$

The final differential equation of the wave equation is given by Equation 2-13, where the superscript n denotes the time level n , while the superscript $n+1$ denotes the next/subsequent time level.

$$\frac{u_i^{n+1} - u_i^n}{\Delta t} = -c \frac{u_i^n - u_{i-1}^n}{\Delta x} \quad (2-13)$$

Chapter 3 Computational Approach

3.1 Modeling of duct and propeller systems

Validation is a critical process of any numerical investigation. The complex flow around MAVs emphasizes the need of this process. Prior to the analysis of the full MAV configuration, the code and the propulsion system are investigated numerically. The code is tested to examine its accuracy in predicting the complex flow around a low aspect ratio lifting surface at low Re, which is typical for MAVs. In addition, the propeller computational model is validated to ensure that it produces the same amount of thrust as the actual propeller. There are two validation cases: one is for the code and the other one is for the propeller system.

- The code validation case is a low AR annular airfoil/duct with Clark-Y airfoil section, the experimental data of which are available. The duct was one of the five models that was investigated experimentally by Fletcher [7].
- The propeller model, the static thrust data of which are obtained from the thrust test. For simplicity, the propeller is modeled as an actuator disc, since such an option is available in Fluent.

Based on the confidence in the data accuracy gained after the validation, the entire ducted propeller is studied and presented in the last part of the section. The model developed for the ducted propeller to estimate the thrust and aero-performance will be further used in simulating the flow around the full MAV configuration, which will be shown in Section 3.2.

3.1.1 Duct Model Description

National Aeronautics and Space Administration (NASA) and U.S Department of Defense (DOD) intensively conducted research on the aerodynamics of annular airfoil/duct in the period 1950-1960. Nevertheless, the aerodynamic data of ducts are rarely available for public due to the classified nature of such work. Only a few of these reports, e.g. Fletcher's NASA technical report [7], are now publicly accessible. This technical note is used as a reference here because it provides comprehensive data on duct with various shapes, sizes and operational conditions.

The duct studied by Fletcher has a Clark Y airfoil-section, an aspect ratio of 1 and Re of 1.22×10^6 . The computation here is conducted using segregated solver, Spalart-Allmaras viscous model and 716,000 hexahedral cells. The grid independence study is performed using two other grids, each with 350,000 and 1,180,000 hexahedral cells.

The aspect ratio of the duct is 1, which means it has an equal length of diameter (d) and chord (c) of 16 in or 0.4064 m. The airfoil section is Clark Y with a maximum thickness ratio of 0.117 (Figure 3-1). The coordinate of the airfoil section is taken from the database of the University of Illinois at Urbana-Champaign (UIUC) [52]. The area of the duct is 0.165 m^2 , calculated using the following equation $A = d \times c$.

Figure 3-1 Clark Y airfoil section

One of the key steps in the pre-processing stage is to generate the grid for and around the duct (Figure 3-2). The duct has an airfoil section which requires fine grids in the boundary layer. Therefore, grid refinement is a trade-off task with respect to the grid resolution and the available computer resources. Skewness is often introduced to the model when the grid is too refined or coarsens in the area of small gap. In the case of the duct, skewness often occurs at the lip or inlet, which has a curvy profile.

Figure 3-2 Computational model of the duct

3.1.2 Operation Conditions

The operation conditions that are set for the computation must conform to the experiment in order to ensure data comparability. The experiments are conducted at a dynamic pressure of 24.9 lb/ft^2 or 1192.2 Pa , which corresponds to a Mach number of 0.13. The flow is modeled as ideal gas with pressure far-field as the boundary conditions. The velocity inlet-outflow boundary is also tested and the results are compared with the one using pressure far field boundary; the results are in a very good agreement.

The sound velocity, a is calculated from the prescribed Mach number (M) using the definition of $a = \frac{V}{M}$. In the calculation, γ is the heat capacity ratio and R is the specific gas constant. For air in standard conditions, γ is 1.4 and R is $287 \frac{J}{kgK}$. Table 3-1 summarizes the operation conditions used in the numerical simulation.

Parameter	Value	Description
Density (ρ)	1.225 kg/m ³	Standard sea level
Dynamic pressure (q)	1192.2 Pa	Prescribed in NASA reference [7]
Mach number (M)	0.13	Prescribed in NASA reference [7]
Velocity (V)	44.1 m/s	$V = \sqrt{\frac{2q}{\rho}}$
Temperature (T)	286.6 K	$T = \frac{a^2}{\gamma R}$
Operational pressure	100779.2 Pa	$P = \rho RT$

Table 3-1 Operation conditions of the validation case

It is important to understand the coordinate system used in Fluent before starting the computation. The coordinate system in Fluent is the Cartesian coordinate system. This coordinate system notation is different from the body-axis system that is commonly used in the performance, control and stability analysis of a conventional airplane [18, 38, 53]. For clarification, the notation system used in Fluent is presented here.

In the airplane body-axis system, the x^+ axis points forward to the advancing direction of the airplane. The z^+ axis points downward on the vertical symmetry plane of the airplane while the y^+ axis points out to the right if it is seen from behind the airplane (Figure 3-3a). In Fluent, the x^+ axis points to the opposite direction of the same axis in the airplane body-axis system. If seen from the x^+ axis, the y^+ axis points upward and the z^+ axis points out to the left (Figure 3-3b).

Figure 3-3 Comparison between body-axis system (a) and Fluent coordinate system (b) notation

In the computation, two methods are used to impose the boundary condition to simulate angles-of-attack and sideslip-angles in the computation. The first method uses the vector components of the velocity to represent the angle. The velocity is set as its x, y or z components with respects to the angle simulated. To simulate angle-of-attack (α), the velocity (V) is defined using its x and y components as:

$$V_x = V \cos \alpha \quad (3-1)$$

$$V_y = V \sin \alpha \quad (3-2)$$

For sideslip, the velocity is set into x and z components as:

$$V_x = V \cos \alpha \quad (3-3)$$

$$V_z = V \sin \alpha \quad (3-4)$$

The second method to impose the angle of attack or sideslip angle is by rotating the object such that it creates an angle with respect to a certain axis. As an example, the duct is rotated 10° in the clockwise direction with respect to the y-body axis to simulate angle-of-attack of 10° (Figure 3-4).

Figure 3-4 Two methods to simulate angles-of-attack and sideslip angles

3.1.3 Numerical investigation and validation

Numerical simulations are conducted using both structured and unstructured grid to support the methods mentioned in the previous section. Initially, a multi-block structured grid is used. The multi-grid is generated by dividing the domain into smaller cubes that are discretized using hexahedral cells. This type of grid is only advantageous for the velocity vector method because the structured grid must be regenerated for every angle-of-attack if the second method is used, which is impractical. Consequently, the unstructured grid is used to enable the simulation of various angles-of-attack using the second method. Unstructured grid is more suitable for complex geometry that cannot be partitioned into smaller domain.

The structured grid model is termed as the fixed duct because the duct is held fixed in a position while the velocity vectors are used to simulate the angle-of-attack. On the other hand, the unstructured grid model is termed as the rotated duct because the duct is being rotated in the simulation to represent various angles-of-attack.

Fluent has two solvers: segregated (pressure-based) and coupled (density-based). The coupled solver is initially created and more appropriate for high-speed compressible flows, and thus the duct computation is performed using the segregated solver. The Spalart-Allmaras turbulence model is used because of its ability to provide good results with low computation cost [5, 23]. Other turbulence models such as $k-\epsilon$ and RANS have been tested and found to usually over predict C_D values for a wide range of angles-of-attack [5]. The second-order-upwind scheme is used to provide higher accuracy, while the pressure-velocity coupling used is SIMPLE (Semi Implicit Pressure Linked Equation). Table 3-2 summarizes the case settings for the duct numerical investigation.

Parameters	Description
Solver	Segregated
Turbulence model	Spalart-Allmaras
Pressure-velocity coupling	SIMPLE
Discretization	Second order upwind

Table 3-2 Case setting of the duct study

3.1.3.1 Structured-grid model

A number of models are created and tested to obtain the most optimum domain configuration. The final model is achieved through a series of iterations. Different domain configurations are

tested (Figure 3-5), such as square, cubical and cylindrical ones. Different boundary distances of the domain are also simulated to study its influence to the numerical results.

Figure 3-5 Domain configurations of the duct computational model

The final model (Figure 3-6) is built by considering accuracy and computational cost. It turns out that a simple cylindrical model with minimum partitions, which is chosen as the final one, is able to give a reasonably well prediction of the aerodynamic forces and moments. This model can be meshed easily and quickly because it has the least number of partitions. The cylindrical shape also allows fine distribution of grids on the curvature surface of the duct.

Figure 3-6 Isometric view of the mesh

The final model has 12 partitions, which are meshed using hexahedral cells. The boundary condition used for the outer domain is pressure far-field. Velocity inlet and outflow are also used in this case. The duct's surface is modeled as a non-slip wall, which implies that the velocity is zero at the surface of the duct. The boundary distance of the model is stated in Table 3-3.

Domain	Distance
Front	3c
Rear	7c
Upper	3c
Lower	3c

Table 3-3 Boundary characteristic distances of the computational domain

3.1.3.2 Baseline tests for structured-grid duct model

1. Effect of convergence limit

For the steady flow situation, a simulation is converged when the residual value drops below a limit set by the user. In Fluent, 10^{-3} is the default setting of the residual criterion. In order to check the solution accuracy, a test was run with the residual limit criterion decreased to 10^{-6} . It is found that the additional computational cost is not worth since the results show that the steady convergence solution is achieved in both cases (Figure 3-7). Therefore, 10^{-3} is used as the residual criterion for the rest of the computation. Detailed definition of the residual criterion is described in the Fluent User Guide, Section 25.18.1 [51].

Figure 3-7 Convergence limit test results for structured grid

2. Grid independence study

Three different grids with 350,000, 716,000 and 1,180,000 cells are tested. The C_L and C_D results only change slightly with the increase of the grid size (Figure 3-8). This trend showed that grid independence is already achieved even with 350,000 cells. In average,

the largest grid size brings 4% improvements to the C_D and C_L predictions with more than 2 hours increase in run time. However, this grid size is the only one that gives good predictions of C_m . As a result, all computations are conducted using the grid with 1,180,000 hexahedral cells.

Figure 3-8 Grid independence study for structured grid

3. Effect of faraway boundary

The Fluent user-guide suggests 20 chord distance from the body for pressure-far-field boundary conditions [54]. Due to the limited computer memory available, the grid size is restricted to 1.4 million cells. For this size, a far field boundary defined at 20 chord distance leads to a very coarse grid, unable of capturing the near body effects. Therefore, much lower distance will be used for the outer boundary. In order to check the effect of the boundary distance, two values are tested: 3 and 6 chords (Figure 3-9). High angles-of-attack, 60° and 90° , are chosen because the wake which is induced behind the duct usually propagated and hit the upper boundary. Increasing the distance from 3c to 6c did not improve the results, indicating that the 3 chord distance is enough for recovering to an unperturbed flow and imposing the boundary conditions.

Figure 3-9 C_D Faraway boundary test results for structured grid

4. Near wall grid distance (y^+)

As mentioned in Chapter 2, structured grid gives users better control in creating and distributing the grid. This particular advantage is clearly displayed in the effort to meet the wall y^+ requirements. The turbulence model, Spalart-Allmaras, is recommended to be used in a model with y^+ smaller than 1 (<1) or larger than 30 (>30) [51]. However,

problems can occur in the case of too coarse grids with $y^+ > 400$ [55]. Based on this guideline, the wall y^+ of the current model is well within the suggested range. Grid refinement is performed as an effort to reach wall $y^+ < 1$. However, the distance from the wall (y^+) is only reduced to half of the original value, from the maximum of around 80 to 40 (Figure 3-10 and 3-11).

At the angle-of-attack of 60° , experiment shows that the duct C_L and C_D is 0.55 and 1.28 respectively. The non-adapted or the original model predicts $C_L = 0.809$ and $C_D = 1.246$, while the adapted model predicts $C_L = 0.792$ and $C_D = 1.225$. In other words, the results are improved in the order of 3% for C_L , while worsen by 1.6% for C_D . Considering the simulation time and the results improvement, the non-adapted grid will be used in the computations.

Figure 3-10 Wall y^+ of the duct model for non-adapted structured grid

Figure 3-11 Wall y^+ of the duct model for adapted structured grid

3.1.3.3 Unstructured-grid model

As mentioned earlier, the unstructured grid model is created to simulate the angle-of-attack using the second method (Figure 3-12). The duct is rotated with respect to the 0.25 chord as stated in the experiment. The model is only divided into three blocks and then meshed using unstructured grid: the tetrahedral cells. Only half of the duct is modeled using the unstructured-grid because of the limited computer resources available.

Figure 3-12 Grid front view at the angle-of-attack of 20°

For complex geometry, meshing using unstructured-grid is easier and faster. The problem with this type of grid lies in the lack of control in the grid generation process. The users can only set the initial size, the growth rate and the maximum size of the grid without having control over the distribution of the grid generated.

3.1.3.4 Baseline tests for unstructured-grid model

Similar to the structured grid model, the unstructured grid model is tested for four different parameters: convergence limit, grid independence, faraway boundary sensitivity, and near wall grid distance (wall y^+). The duct geometry, case setting, and operating parameters are remained unchanged. The results of each test are explained in the following sections.

1. Effect of convergence limit

The unstructured grid is also tested for two convergence criteria. Unlike the structured grid, the results from the unstructured grid model are improved appreciably when the residual criterion is decreased. It is clearly shown in Figure 3-13, particularly at the angle-of-attack of 60°, the computational results from the lower criterion is much closer to the experiment data. These results show that the unstructured grid needs more iteration/time to reach a steady solution. Accordingly, the next computations will be performed using the residual criterion of 10^{-6} .

Figure 3-13 Convergence limit test results for unstructured grid

2. Grid independence study

Three different grids are tested for the grid independence study. The first model has 500,000 cells, while the other two have 1 and 1.2 million cells. The results shown in Figure 3-14 suggest that grid independence is already achieved at 500,000 cells because the C_D , C_L and C_m predictions only change slightly with the increase of the grid size. There is an exception for C_m predictions, where there is a large gap between the experimental and computational results at the angle-of-attack of 20°. This results difference will be discussed later in Section 3.1.4.

The grid with 1.2 million cells gives only 2% and 6% more accurate results for C_D at 40° and C_L at 20°, respectively, and less accurate results for the other simulations. Overall, the grid with 1 million cells predicts the drag and the lift reasonably well and thus it is used for the rest of the computations.

Figure 3-14 Grid independence study for unstructured grid

3. Effect of faraway boundary

The effect of the boundary distance is tested using two models with 3 and 6 chord distances, which are referred as 3c and 6c respectively (Figure 3-15). The experiment data for the angle-of-attack of 60° is $C_L = 0.55$ and $C_D = 1.28$. The 3 chord model predicts $C_L = 0.58$ and $C_D = 1.3$, whereas the 6 chord model predicts the $C_L = 0.54$ and $C_D = 1.24$. These results show that increasing the boundary distance has small effects on the accuracy. Consequently, the computation will be performed using the model with 3 chord boundary distance.

Figure 3-15 Simulation model with 3c (a) and 6c (b) rear boundary

4. Near wall grid distance (y^+)

A very fine grid near the duct surface is very difficult to achieve. As a result, the wall y^+ of the model is in the interval of 175–275 (Figure 3-16). This value, although higher than normal, is still within the acceptable range of $y^+ < 400$. Efforts to refine the grid had failed due to the memory capacity. The non-adapted model has approximately one million cells, while adaptation adds more than 200,000 cells. The total grid size is approximately 1.2 million, which is very close to the grid maximum limit. Referring to the structured-grid results, it is expected that grid refinement will only give slight accuracy improvement. Considering the computer capacity and the minimum possible results improvement, the non-adapted grid is used for the computation.

Figure 3-16 Wall y^+ of the duct for non-adapted unstructured grid

3.1.4 Numerical result discussion of the duct model

In the fixed duct computation, the lift predictions are in a good agreement with the experimental data for low angles-of-attack (0° – 30°). For high angles-of-attack, the difference

between the numerical and experimental results is higher than for low angles-of-attack (Figure 3-17).

Overall, the rotated duct model gives more accurate results than the fixed duct. This conclusion is taken after comparing the numerical results for the fixed and the rotated duct models with the experimental data. The results using the rotated duct model are close to the experiment, except at the stall angle-of-attack. Stall region is very difficult to predict due to its complex flow phenomenon. Near the stall angle, the separation point moves rapidly forward and a large wake region is created. This region of turbulent and separated flow is unstable and thus making it difficult to predict. As a result, the computational results show higher stall-angle of 40° as compared to 25° from the experiment. Such result is commonly found in low Re, low aspect ratio wings or MAV-related research [4, 5, 10, 23].

In both models, Fluent under-predicts lift at low angles-of-attack and over-predicts lift at higher angles-of-attack. On the contrary, the drag is over-predicted at low angles-of-attack (0° - 20°) and under-predicted at higher angles-of-attack (Figure 3-18). The possible source of this error is the inability of the turbulence model in predicting the transition behavior [26]. The Spalart-Allmaras turbulent model which assumes a fully turbulent condition may overestimate the turbulent level at low angles-of-attack. In contrast, the Spalart-Allmaras model may under-predict the turbulent level at higher angles-of-attack. The former increases the separation and creates larger wake region, whereas the latter gives more attached flow. Larger separation region results in a high drag and a low lift, while narrower separation region results in a low drag and a high lift.

Figure 3-17 Lift vs. Angle of attack for various computational models

Figure 3-18 Drag vs. Angle of attack for various computational models

Figure 3-19 Pitching moment vs. Angle of attack for various computational models

The moment results showed a similar pattern as the experiment (Figure 3-19). There is shift in the angle-of-attack where the change from negative to positive moment coefficient is occurred. The experiments show that the shift is occurred between the angles of attack of 20° to 30°, but the numerical results changed sign between the angles-of-attack of 10° and 20°. There are two possible reasons behind this trend:

- In the computation, the code might capture different pressure distributions but still manage to obtain the correct force resultant.
- In the experiment, there is a high possibility of error in the moment measurement. Fletcher [7] obtained a random data of moment coefficients and mentioned the difficulties in measuring the small values of moment.

3.1.5 Isolated propeller system model

A two-blade propeller is an important part of the VTOL MAV ducted/shrouded propeller configuration. Therefore, the computational model of the propeller must be validated before the full-configuration is analyzed. The purpose of the validation process is to check whether the actuator disc model in the computation represents the actual flow field. This process is also needed to ensure that the thrust generated by the disc is equal to the actual propeller. The computational model is validated using the momentum equation, which is mentioned in Section 2.2.3.1. The numerical results obtained using the actuator disc model, are expected to be higher than the experimental ones. This is due to the assumption used in the numerical simulation that the flow is inviscid while in reality the flow is viscous. Viscous flow is identified with frictions and vortex, which can significantly reduce the efficiency of the propeller (Refer to Section 2.2.1).

3.1.5.1 Modeling of Propeller System

Analysis and tests are performed to choose the correct boundary conditions for propeller modeling. The starting point of the analysis is the known and unknown parameters in the case. Some of the known parameters are the static thrust (T), the air density (ρ) and the propeller area (A). These data can be processed using the momentum equation to obtain unknown parameters such as induced velocity (v_i) and mass flow rate ($\dot{m}$). The derived parameters can be used as inputs data in the disc actuator analysis. The actual flow physics in static conditions and the boundary conditions available in Fluent are also considered. There are only few inlet and outlet types for outer domain in Fluent. There are velocity, pressure and mass flow inlet for inlet; and outflow, pressure outlet and mass flow outlet for the outlet boundary conditions.

There are four combinations of inlet-outlet boundary conditions that are tested: mass flow inlet-pressure outlet, pressure inlet-mass flow outlet, pressure inlet-pressure outlet, and velocity inlet-pressure outlet. The most suitable combination is chosen based on the analysis of the velocity and the pressure distributions. An important factor taken into consideration is the stability of the model.

The computational model consists of a cylinder with the actuator disc modeled as 2D circular plate in the middle of the domain (Figure 3-20). The actuator disc is modeled using fan boundary condition in Fluent, while the inlet and outlet boundary conditions are modified according to the boundary condition combinations mentioned previously. The outer boundary of the cylinder, which is referred as the isolated surface in Figure 3-20, is defined as wall to maintain a smooth

flow from the inlet to the outlet of the domain. These boundary conditions are used to simulate the static conditions when the advancing/forward speed of the propeller/disc is equal to zero.

According to the momentum theory, the flow velocity downstream of the actuator disc is twice the velocity at the disc itself. Another characteristic of the flow field is that the static pressure downstream, where the influence of the disc diminishes, is dictated by the atmospheric pressure.

Figure 3-20 Baseline computational model of the propeller system

The velocity and pressure characteristics of the disc model with the pressure inlet-pressure outlet boundary condition are in a good agreement with the momentum theory. The flow enters the inlet, accelerated by the disc and leaves through the outlet. The flow velocity right behind the disc is approximately twice the velocity in front of it (Figure 3-21). The flow velocity in front of the disc is around 4 m/s, while right behind it is around 8 m/s. Furthermore, Figure 3-22 shows that the influence of the pressure jump diminishes downstream (towards the outlet) of the disc. This curve shows flat lines before and after the jump. This trend indicates that the static pressure downstream returns to the undisturbed static pressure value which is the atmospheric pressure. This combination of boundary conditions is also stable with smooth iteration process and quick convergence.

Figure 3-21 Velocity distribution of the pressure inlet-pressure outlet boundary condition

Figure 3-22 Pressure distribution of the pressure inlet-pressure outlet boundary condition

Further investigation is conducted to check the ability of the actuator disc model to represent the actual flow field. Simulations are conducted to observe the velocity distribution across the actuator disc and the velocity vector around it. The velocity distribution across the disc's surface is plotted along its diameter, which is indicated by a red line (Figure 3-23a). The velocity is lowest at the tip/perimeter of the disc and reaches its maximum value at approximately 0.75-radius (Figure 3-23b). Around the center of the disc, the flow velocity is nearly uniform. The velocity vector over the disc gives an insight on the behavior of the flow. Figure 3-24 shows the flow making a turn at the disc's tip/perimeter to enter the slipstream. This reversed flow at the disc's tip/perimeter can be the reason behind the low-velocity around this area. Moving towards the center, the turning-flow creates an acceleration effect on the inflow. The result of this behavior is the maximum velocity near the disc's edge. Around the center of the disc, the effects of this turning-flow become less obvious. The inflow is almost undisturbed, which may explain the

unvarying value of velocity near the center of the disc. Figure 3-24 shows similarities to Figure 2-8b, which is the theoretical model of the flow field around a propeller at static conditions. These results suggest that the actuator disc with pressure inlet-pressure outlet boundary condition is able to realistically represent the physics of the flow around a propeller at static conditions.

Figure 3-23 Velocity distribution on the actuator disc's surface

Figure 3-24 Velocity vector over the actuator disc's surface

The combination of boundary conditions used in the final computational model is pressure inlet-pressure outlet with wall boundary to isolate the propeller system (Figure 3-25). In the cruise mode, the pressure inlet and the wall boundary can be changed to the velocity inlet, such that the boundary conditions used in this chapter is similar to the duct computational model. The connection between the imposed boundary conditions for the duct and for the propeller is important since they will be coupled to form a ducted propeller system for the VTOL MAV full configuration analysis.

The input parameter of the fan is the pressure jump which is obtained from the iterative computations. The detail about this process is explained in the next section. The input for the pressure inlet is the gauge total pressure, whereas for the outlet is the gauge static pressure. The term “gauge” explains that the pressure is relative to the operating/atmosphere conditions. In static conditions, the gauge total pressure is zero because the velocity far upstream of the disc is assumed to be zero. The gauge static pressure at the outlet is also zero due to the assumption that far downstream the flow is undisturbed and the static pressure is approximately equal to the atmospheric pressure. The operating conditions are set to be equal to the standard sea-level conditions with an operating pressure of 101,325 Pa and a temperature of 288.15 K. The same operating conditions will be used for the VTOL MAV full-configuration analysis in Chapter 4.

Figure 3-25 Final computational model of the propeller system

3.1.5.2 Calibration Algorithm

Derived from the experimental data, the pressure jump is used as the input data for the numerical modeling of the propeller system. The output of the numerical investigation is the

thrust that will be further compared with the thrust value obtained in the experiment. The agreement between the numerical and experimental results will validate the propeller system model.

There are two sets of the experimental data used in the calibration: one is for the half-charged battery (Table 3-4), and the other is for the full-charged battery (Table 3-5). The examples given in this part are only for the full-charged battery. Nevertheless, the procedure for the half-charged battery follows the same algorithm as the fully-charged one.

Throttle	Current (A)	Voltage (V)	Thrust (N)	Rot Speed (RPM)	Battery Power (Watt)	Torque Approximation
1	0	0	0	0	0	0
2	0.20	11.15	0.12	1619	2.23	0.0021
3	0.65	11.03	0.45	2975	7.17	0.0067
4	1.17	10.80	0.72	3958	12.64	0.0121
5	1.64	10.59	1.11	4542	17.31	0.0169
6	2.13	10.34	1.43	5136	22.01	0.0220
7	2.60	10.10	1.67	5542	26.26	0.0269
8	3.30	9.90	1.93	5893	32.67	0.0341
9	3.75	9.75	2.07	6123	36.56	0.0388

Table 3-4 Experiment results for hovering mode (half-charged battery)

Throttle	Current (A)	Voltage (V)	Thrust (N)	Rot Speed (RPM)	Battery Power (Watt)	Torque Approximation
1	0	0	0	0	0	0
2	0.27	12.20	0.13	1773	3.27	0.0028
3	0.69	12.17	0.45	3132	8.40	0.0071
4	1.20	11.90	0.86	4242	14.28	0.0124
5	1.80	11.80	1.19	4921	21.24	0.0186
6	2.30	11.60	1.49	5464	26.68	0.0238
7	3.20	11.40	1.84	5968	36.48	0.0331
8	3.70	11.10	2.07	6334	41.07	0.0382
9	4.29	11.03	2.27	6587	47.32	0.0443
10	4.76	10.80	2.45	6827	51.41	0.0492
11	6.40	10.53	3.25	7645	67.39	0.0661

Table 3-5 Experiment results for hovering mode (full-charged battery)

The iterative procedure to evaluate the pressure jump is performed in the following sequence:

1. Initial guess

The initial guess of the pressure jump (Δp_l) is derived using Equation 3-5.

$$\Delta p = \frac{T}{A} \quad (3-5)$$

T is the experimental thrust and A is the actuator disc area. The disc diameter is 0.2032 m, thus the disc area (A) is 0.0324 m².

2. Iterative Procedure

Using the input value of Δp_1 from the initial guess, a full numerical simulation is performed using Fluent. The output thrust from the numerical analysis is then compared with the experimental thrust. If an agreement between the two results is not found, a new value of pressure jump (Δp_{i+1}) will be used to replace the previous one (Δp_i). The new pressure jump is predicted using Equation 3-6.

$$\Delta p_{i+1} = \Delta p_i \times \frac{T}{T_i} \quad (3-6)$$

T_i is the numerical thrust at iteration i and T is the experimental thrust. The procedure is repeated until the convergence is reached.

In the case illustrated here, the convergence is reached after three iterations (Figure 3-26). The initial guess (Δp_1) of 56.7 Pa is used as the input value for the first iteration. The output thrust of 2.99 N shows that Δp_1 over-predicts the experimental data. The second value of pressure jump (Δp_2) is 34.9, obtained using Equation 3-6. This value of pressure jump generates 1.85 N thrust, which is still above the experimental data. The third iteration with $\Delta p_3 = 34.8$ Pa gives output thrust of 1.84 N. This value of numerical thrust matches the experimental data. Therefore, the propeller system is calibrated and the iteration is stopped. Table 3-6 summarizes the numerical results of the iteration process to predict the pressure jump, which produces the static thrust of 1.84 N.

Figure 3-26 Iteration process to predict the pressure jump corresponds to $T = 1.84$ N

Iteration	Pressure Jump (Pa)	Thrust (N)
1	56.7	3.00
2	34.9	1.85
3	34.8	1.84

Table 3-6 Numerical results of the iteration process in predicting static thrust of 1.84 N

- The iterative algorithm is repeated for each throttle such that the entire throttle range of the propeller is investigated (Table 3-7).

Throttle {RPM variation}	Thrust (N)	Numerical prediction of the pressure jump (Pa)	
		Initial	Final
3	0.45	13.9	8.5
7	1.84	56.7	34.8
11	3.25	100.2	61.6

Table 3-7 Examples of the iteration results from simulation

3.1.5.3 Ducted propeller study

As mentioned in the literature review, one of the advantages of ducted/shrouded propeller is its higher static thrust compared to free propeller. The propeller is simulated using the computational model used in the propeller validation. The duct geometry is from the VTOL MAV configuration with the actuator disc located at the duct inlet. The duct has an Eppler180 airfoil section, diameter of 21 cm and length of 15 cm. The boundary conditions used in the ducted propeller model are similar to the free propeller model. The ducted propeller is shown in Figure 3-27 with the meshed surface at the duct inlet as the actuator disc.

Figure 3-27 Ducted propeller computational model

The ducted propeller is tested in static conditions, which means the propeller creates its own inflow and that there is no air passing through the propeller due to the flight speed. The

pressure jump at the actuator disc is set at 61.6 Pa which corresponds to 3.25 N of static thrust for the propeller alone.

Platt [37] mentioned that the ratio of static thrust between ducted ($T_{ducted_propeller}$) and free propeller ($T_{propeller}$) can be given by the following Equation 3-7.

$$\frac{T_{ducted_propeller}}{T_{propeller}} = 1.26 \left(\frac{S_3}{S_2} \right)^{\frac{1}{3}} \quad (3-7)$$

S_2 is the area of the propeller and S_3 is the area of duct outlet. The duct in our configuration has an equal area of the propeller and the duct outlet, thus Equation 3-7 becomes Equation 3-8.

$$\frac{T_{ducted_propeller}}{T_{propeller}} = 1.26 \quad (3-8)$$

The static thrust (T) is calculated using Equation 3-9, with m is the mass flow rate measured at the propeller surface and V_4 is the velocity downstream of the duct outlet.

$$T = mV_4 \quad (3-9)$$

Figure 3-28 Translation of the disc surface at 0.3 m from the duct inlet

A meshed-surface is translated to a certain location, where the average V_4 needs to be measured. The distance of 0.3 m (from duct inlet) is chosen because the translation surface beyond this location is of low quality (Figure 3-28). Nevertheless, two sets of data from 0.15 m (Table 3-8) and 0.3 m (Table 3-9) behind the duct inlet clearly show the increase of static thrust when the propeller is enclosed by a duct.

Parameter	Propeller	Ducted Propeller
Velocity (m/s)	8.39	8.75
Mass flow rate (kg/s)	0.24	0.27
Thrust (N)	2.01	2.37

Table 3-8 Static thrust calculation with reference to velocity at 0.15 m

Parameter	Propeller	Ducted Propeller
Velocity (m/s)	8.17	8.12
Mass flow rate (kg/s)	0.24	0.27
Thrust (N)	1.95	2.20

Table 3-9 Static thrust calculation with reference to velocity at 0.3 m

The mass flow rate ($\dot{m}$) is measured at the propeller surface. As it is seen from the data, the duct increases the mass flow rate passing through the fan and that the static thrust of the ducted propeller is higher than the free propeller. Based on the velocity at 0.15 m behind the duct inlet, the ratio of the static thrust between the ducted and the free propeller is

$$\frac{T_{\text{ducted_propeller}}}{T_{\text{propeller}}} = \frac{2.37N}{2.01N} = 1.18, \text{ while measurement at 0.3 m behind the duct inlet gives}$$

$$\frac{T_{\text{ducted_propeller}}}{T_{\text{propeller}}} = \frac{2.2N}{1.95N} = 1.13. \text{ These results indicate that the actuator disc model can give}$$

reasonably accurate results and can be used to represent the actual propeller.

3.2 Aerodynamic and static stability analysis

3.2.1 Detail configuration of the VTOL MAV

The final configuration of the VTOL MAV is obtained after an integrated and structured conceptual design phase. The technical drawing of the VTOL MAV created using SolidWorks® is shown in Figure 3-29, while the computational model built using Gambit is shown in Figure 3-30. The main configuration consists of ducted-propeller, wing, payload-box, stator, as well as horizontal and vertical tails. The winged-ducted propeller is the most suitable configuration to satisfy the Design Requirement and Objective (DRO). The VTOL MAV is capable of VTOL and hover, flying for medium duration and range, pack-able and easy to operate [8]. The ducted propeller configuration offers advantages such as high static thrust and high propulsive efficiency, able to generate additional lift from the duct and the wing, low noise and high safety level.

Figure 3-29 Technical drawing of VTOL MAV [8]

Figure 3-30 Computational model of VTOL MAV from different views

The duct has an Eppler-180 airfoil section, a chord length (c) of 0.15 m and an inner diameter (d) of 0.2 m which correspond to an aspect ratio of 1.33. The wing has the same airfoil section as the duct, a span (b) of 0.42 m and an area (S) of 0.076 m². The wing's span and area are the reference length and area used in the non-dimensional calculations of the aerodynamic forces and moments. The tapered-wing has a low aspect ratio of 2.32, a twist angle -2° and a sweep angle of 5°. The vertical tail, horizontal tail and stators are built from NACA 0018 and have a rectangular platform. Details about the size and geometrical properties of the final design are shown in Table 3-10.

Airplane	Length x Span x Height	0.4 x 0.42 x 0.2 (m)	Horizontal Tail	Span	0.17 m
	Inner Diameter	0.2 m		Chord at root	0.06 m
Duct	Chord	0.15 m		Taper Ratio	1
	Aspect Ratio	1.33		Area	0.0102 m ²
Wing	Airfoil	Eppler-180	Vertical Tail	Sweep	0
	Span	0.42 m		Setting angle (δ)	-2°
	Chord at root	0.25 m		Span	0.17 m
	Taper ratio	0.4		Chord at root	0.06 m
	Wing area	0.076 m ²		Taper Ratio	1
	Aspect Ratio	2.32		Area	0.0102 m ²
	Wing Sweep	5°		Sweep	0°
	Wing Twist	-2°		Setting angle (ϵ)	-1°
	Dihedral	0°	Airfoil for Stators, Horizontal and Vertical Tails		NACA 0018
	Airfoil	Eppler-180	Propeller	Diameter x Pitch	0.20 m x 0.0965 m
Stator	Span 1 stator	0.08 m	Left Stator	Left plane setting angle (γ_L)	-10°
	Chord	0.06 m		Number of blade	3
	Area	0.0048 m ²	Right Stator	Right plane setting angle (γ_R)	10°
	Aspect Ratio	1.33		Number of blade	3

Table 3-10 Size and geometrical properties of VTOL MAV

The location of the vehicle's center-of-gravity in the body-axis is shown in Figure 3-31. This point is used as the reference for rolling, pitching and yawing moment calculations in the aerodynamic and stability analysis.

Figure 3-31 Center of gravity of VTOL MAV in body-axis (*payload-box excluded in the picture*)

3.2.2 Computational model of the VTOL MAV

The geometry of the VTOL MAV is re-drawn in Gambit to avoid extensive healing for the imported drawing from SolidWorks®. Two different computational models are created, called the VTOL MAV-baseline and the VTOL MAV. The VTOL MAV model fully incorporates the geometrical properties of the final design. The wing, stator, horizontal and vertical tails are set according to the details in Table 3-10. Therefore, the simulations of the VTOL MAV model provide numerical analysis of the aerodynamic and stability characteristics of the vehicle.

Figure 3-32 Geometry and mesh of VTOL MAV

The differences between the VTOL MAV and VTOL MAV-baseline are the payload-box and the setting angle of the other components (Figure 3-32 and 3-33). The VTOL MAV-baseline computational model does not include the payload-box. The VTOL MAV-baseline is a reference case with all the setting angles of the lifting and control surfaces set to 0°. This reference case plays an important role in the future research about quick prediction of the aerodynamic characteristics of the VTOL MAV using the Component-Build-Up Method (Chapter 5).

Figure 3-33 Geometry and mesh of VTOL MAV-baseline

Both models are simulated with and without the propeller (Figure 3-34). In real operation, the amount of thrust needed to perform a mission is different for every angle-of-attack and sideslip-angle. However, in this research the pressure jump of the actuator disc is set fixed at 61.6 Pa which corresponds to 3.25 N in static thrust. This thrust amount is produced when the throttle is set to maximum (Table 3-5).

Figure 3-34 VTOL MAV: with propeller (a) and without propeller (b)

The computational model is obtained after several changes are made to the final configuration. The purpose of this process is to simplify the grid generation process. The difference can be seen by comparing the CAD drawing (Figure 3-29) with the VTOL MAV computational model (Figure 3-30). The first simplification is done by removing the four struts which connect the empennage and the duct, and the support structure between the right and left stators. Next, the engine and support structure of the propeller are also removed because the propeller is modeled as an actuator disc. Support struts and engine installations typically contribute up to 3% of the zero-lift drag of an aircraft [56]. Therefore, removing these components might create a maximum of 3% under predictions of the MAV's drag.

The last simplification is made to the wing part. In the CAD drawing, the movable and fixed parts of the wing are attached to each other. In the computational model, there is 1 mm gap between them to avoid skewness (Figure 3-35). This gap effect on the results is investigated through the

simulation of the duct-wing coupled model with and without the gap. The actual model (without gap) predicts $C_D = 0.0354$ and $C_L = 0.0146$, while the gap-model predicts $C_D = 0.0344$ and $C_L = 0.0143$. The difference between the two models is less than 3 %, thus it can be neglected.

Figure 3-35 Gap between the movable and fixed part of the wing

The flow domain for both the VTOL MAV-baseline and the VTOL MAV are discretized using unstructured grids. The unstructured grid is used due to the geometrical complexity of the final configuration. The two models, VTOL MAV-baseline and VTOL MAV, have approximately 1.4 million tetrahedral cells. The grid is refined in the vicinity of the wing, duct, and horizontal and vertical tails because these components are expected to dominantly contribute to the total aerodynamic forces and moments.

Referring to the validation case, there are two methods to simulate the variations of angle-of-attack and sideslip-angle; velocity vector and object rotation (Section 3.1.2). The numerical results of the baseline model show that the object rotation method needs to be applied at high angles-of-attack (above 60°). Convergence is difficult to achieve at high angles if the velocity vector method is used. To accommodate both methods, the VTOL MAV and VTOL MAV-baseline models are divided into two parts: inner and outer. The inner part is spherical in shape, such that it can be rotated in any direction without a need to change the outer part (Figure 3-36 and 3-37).

Figure 3-36 Inner part of VTOL MAV: front (a) and isometric (b) views

Figure 3-37 Inner part of VTOL MAV-baseline: front (a) and isometric (b) views

The first step in the grid generation of the inner part is to mesh all the surfaces of the VTOL MAV. The surfaces are meshed using quad cells with an exception for the airfoil-shaped surfaces where triangular cells are used. The grid is refined in the small gap area such as between the stators and the duct outlet and between the fixed and movable parts of the wing. The iterative process of meshing shows that the grid interval count on the stator surface must not exceed 0.27 and that the grid on the duct surface must be more than 30 cells to avoid skewness. After all surfaces are meshed, a mesh size function is created to generate the volume mesh (Figure 3-38 and 3-39). The growth ratio of the mesh is 1.2. This ratio gives smooth propagation of the mesh and the grid amount is still below the maximum limit.

Figure 3-38 Mesh of VTOL MAV for the angle-of-attack of 0°: z-axis (a) and y-axis (b) planes

Figure 3-39 Mesh of VTOL MAV-baseline for the angle-of-attack of 90°: z-axis (a) and y-axis (b) planes

The outer boundary/frontier is a cylinder with a hollow section in the center for the inner part (Figure 3-40). The cylindrical shape is used because it has a circular surface which is suitable for grid propagation from the spherical inner part. The outer part consists of coarse grids as it is located far from the object. A mesh size function is created with the initial grid size of the outer part is the maximum grid size of the inner part. This is done to create a smooth transition of grids between the two parts.

Figure 3-40 Geometry and mesh (z-axis) of the outer part

The mesh files corresponding to the inner and outer parts are exported separately using Gambit. Later the two files will be combined in Fluent using Append Case File option. The two parts are combined by assigning that two faces to match between them and to generate an interface in Fluent. The interface is shown in yellow, VTOL MAV is in black, while the blue color is velocity inlet boundary and the red surface at the rear side is the outflow boundary (Figure 3-41). The interface boundary condition connects the two parts and makes the code read it as one case. The flow enters from the velocity inlet, leaves through the outflow, and the code solve the flow throughout the whole domain.

Figure 3-41 Full (inner and outer) model of VTOL MAV

The full model (Figure 3-41) extends to the front, upper and lower side for as far as three times of the vehicle's length, whereas the rear boundary is twice longer than the others (six times). The operating conditions are the standard sea level with operating pressure of 101,325 Pa, operating temperature of 288.15 K and density (ρ) equals to 1.225 kg/m³. The operating conditions and the case setting are similar to the validation case; therefore, the details can be found in Section 3.1.2 and 3.1.3. The velocity magnitude of 15 m/s, which is the designed cruise velocity of the VTOL MAV, is used in all computations (Table 3-11).

Parameter	Value
Density	1.225 kg/m ³
Operational pressure	101325 Pa
Velocity	15 m/s
Temperature	288.15 K

Table 3-11 Operation conditions of VTOL MAV computation

3.2.3 Baseline tests for VTOL MAV

Similar to the validation case, the numerical analysis of the VTOL MAV and VTOL MAV-baseline is also begun with a set of baseline tests. Three tests are performed to investigate the effect of convergence limit, the grid independence study and the near wall grid distance (wall y^+). The boundary sensitivity is not tested again because the duct study proves that the current faraway boundary distance is sufficient. In this section, the baseline test results shown are only for VTOL MAV computations. The baseline test results for the VTOL MAV-baseline are given in Appendix A.

3.2.3.1 Effect of convergence limit

In order to check the solution accuracy, the effect of convergence limit is tested using two residual criteria: 10^{-3} and 10^{-6} . For both criteria, the aerodynamic coefficients and mass flow rate are monitored to check whether a steady solution is already achieved. The mass flow rate is measured at the duct inlet, and the aerodynamic coefficients are calculated using the wing area and the span as the reference area and length.

(a)

(b)

Figure 3-42 Mass-flow-rate history for the angle-of-attack of 0°: 10^{-3} (a) and 10^{-6} (b)

Figure 3-40 is the mass flow rate history of the VTOL MAV simulation at the angle-of-attack of 0°. The computation converges after around 150 iterations when 10^{-3} is used as the residual criterion. The display monitor (Figure 3-42a) shows a curve with negative slope which is still dropping. The mass flow rate has not reached a steady solution because the results are still changing with time. On the contrary, the curve for 10^{-6} (Figure 3-42b) has reached a steady result which is shown by the flat line from iteration 600 to 700.

(a)

(b)

Figure 3-43 Drag coefficient history for the angle-of-attack of 30°: 10⁻³ (a) and 10⁻⁶ (b)

The aerodynamic coefficients are recorded with respect to the number of iterations. The C_D shown in Figure 3-41 is for the VTOL MAV at the angle-of-attack of 30°. The left figure (Figure 3-43a) is the C_D history when the residual criterion of 10^{-3} is used. The computation converges after about 750 iterations, but the curve still shows a tendency to drop. When the lower criterion of 10^{-6} is used, the curve becomes flat as the iteration number increases (Figure 3-43b). These results clearly show that the lower residual criterion is needed to reach steady results.

Limit	C_D	C_L	C_l	C_n	C_m
10^{-3}	0.0985	0.0719	0.0043	-0.0021	-4.04E-05
10^{-6}	0.0967	0.0728	0.0044	-0.0019	0.0003

Table 3-12 Computational results for the angle-of-attack of 0°

Limit	C_D	C_L	C_l	C_n	C_m
10^{-3}	0.9222	1.0728	0.0010	-0.0005	0.1094
10^{-6}	0.9215	1.0717	6.25E-05	-0.0002	0.1098

Table 3-13 Computational results for the angle-of-attack of 40°

In the Table 3-12 and 3-13 the computational results for the VTOL MAV at 0° and 40° are given. The results show that decreasing the residual criterion only slightly influence the aerodynamic force predictions; however, it can significantly influence the moment predictions. The moment coefficients are very important in this research because it gives the information about the VTOL MAV stability behavior. Based on this observation, to obtain accurate results of the aerodynamic moments, lower residual criterion (10^{-5} to 10^{-6}) is used in all simulations of the VTOL MAV and VTOL MAV-baseline.

3.2.3.2 Grid independence study

In general, the duct study (Section 3.1) and the VTOL MAV-baseline (Appendix A) numerical results show that 1 million grids are sufficient to accurately predict the aerodynamic forces and moments. Thus, three models with grid amount above 1 million are created. The three models have 1.3, 1.4, and 1.55 million tetrahedral cells. These grid sizes are expected to be sufficient; however they are still tested for the angle-of-attack of 0° to confirm the grid independency.

The results which are summarized in Table 3-14 show that the grid independency has been achieved. The C_D , C_L , and C_m vary slightly with the change of grid size. The model with 1.4 million cells is chosen because it has a medium grid size. There is no experimental data available at the time, thus taking a medium grid size is a reasonable compromise between accuracy and computational cost.

Grid amount (in million of cells)	C_D	C_L	C_m
1.3	0.0958	0.0721	0.0003
1.4	0.0951	0.0724	0.0003
1.55	0.0929	0.0719	6.41E-05

Table 3-14 Grid independence study for the angle-of-attack of 0°

3.2.3.3 Near wall grid distance (y^+)

The wall y^+ of the VTOL MAV model is well within the recommended range [51]. For example, the wall y^+ of the duct and the wing's fixed-part is between 40 and 68 (Figure 3-44). Grid refinement is not performed due to two reasons. First, errors might occur if the grid is adapted because the wall y^+ possibly reaches below the suggested range. The second reason is the computer memory capacity that is limited to approximately 1.4 million grids. The grid size of the current (non-adapted) model is very close to the maximum limit, thus refinement is difficult to do.

Figure 3-44 Wall y^+ of VTOL MAV duct and wing's fixed-part for the angle-of-attack of 0°

3.3 Dynamic stability analysis

A complete set of stability derivative data is necessary for the flight-controller design of the VTOL MAV. Therefore, this part of study is dedicated to predict stability derivatives due to angular rates. The rolling (C_l), pitching (C_m) and yawing (C_n) moment coefficients due to rate of roll (p), rate of pitch (q) and rate of yaw (r) are predicted using full-configuration computational models of the MAV. In the computation, unsteady approach is necessary to impose the boundary condition to simulate the angular rates. For simplicity, the sliding-mesh technique is used to model the constant rotation of the VTOL MAV at the corresponding axis. This method is available in Fluent and simpler than other approaches such as the dynamic meshing [51].

There are five stability derivatives analyzed for the dynamic analysis. In the lateral mode, the rolling moment coefficients with respect to the rate of roll (C_{l_p}) and the rate of yaw (C_{l_r}) are analyzed. In the directional mode, the important stability derivatives are yawing moment variations to the rate of roll and to the rate of yaw; C_{n_p} and C_{n_r} respectively. The last stability derivative is the pitching moment variations to the rate of pitch (C_{m_q}).

Each derivative is calculated by testing two angular rates. The rates of roll tested to calculate C_{l_p} and C_{n_p} are 0.2 deg/s and 1 deg/s. The rates of pitch tested to calculate C_{m_q} are 0.0002 deg/s and 0.0005 deg/s. For C_{l_r} and C_{n_r} , the rates of yaw tested are 0.0005 deg/s and 0.001 deg/s. The freestream velocity (V_∞) is 15 m/s and the propeller is off for all cases.

3.3.1 Numerical parameters of dynamic stability analysis

The reference case used in this analysis is the steady numerical solution of the VTOL MAV at the angle-of-attack and the sideslip angle of 0°. This reference case is commonly selected as a trim point for the stability analysis because aerial vehicles are typically designed to cruise at 0°. This trim point is an equilibrium state on which roll, pitch and yaw disturbances are imposed to check the vehicle stability. The sliding-mesh method is used to impose these disturbances by rotating the sphere around the vehicle with a constant angular rate at the corresponding axis (Figure 3-45). Detailed information on this method can be found in the Fluent user manual [51].

Figure 3-45 Schematic diagram on sliding mesh technique to impose angular rate disturbance

Each moment is tested under different angles, angular velocities and time-step-sizes (Table 3-15). The pitching and yawing simulations require much smaller time-step-size compared to the rolling moment cases to reach a converged solution. The computational models are tested for a set of positive and negative angles of the same magnitude. For example, the vehicle is tested for the roll angles of -5° and 5° . For all moment coefficients, if the orientation angle of the vehicle is symbolized by ϕ for roll, θ for pitch, ψ for yaw, the angular rate by Ω , and the time-step-size is represented by ΔT ; The number of time step (n), pitch rate simulation for example, is calculated using the following equation (Equation 3-10).

$$n = \frac{\theta}{\Omega} \times \frac{1}{\Delta T} \quad (3-10)$$

The number of iterations per time-step is set at 5; thus the total number of iterations is the number of time-step (n) multiplied by 5 (Equation 3-11).

$$iteration = n \times 5 \quad (3-11)$$

There are twelve cases tested, each requires different running time ranging from 4 to 48 hours depending on the number of time steps. The moment data are later plotted against the corresponding angular velocity to deduct the stability derivatives around the pre-determined trim point.

Moment	Angle (deg)	Angular velocity (deg/s)	Time step size (s)	Number of time steps	Number of iterations
Rolling	5 and -5	0.2	0.1	250	1250
		1	0.1	50	250
Pitching	1 and -1	0.0002	1	5000	25000
		0.0005	1	2000	10000
Yawing	1 and -1	0.0005	1	2000	10000
		0.001	1	1000	5000

Table 3-15 Case settings for dynamic stability analysis

3.3.2 Baseline tests for dynamic stability analysis

The baseline tests are conducted to study time-step-size, discretization scheme and pressure-velocity coupling. These tests are important to ensure that the approach used in the analysis is correct and the computational results obtained are reasonably accurate. The baseline-test results are used as a guideline in setting the simulation cases for the dynamic analysis (Appendix E). The computational models for the dynamic stability analysis are set according to Table 3-16.

Parameters	Description
Solver	Segregated
Time	Unsteady
Turbulence model	Spalart-Allmaras

Table 3-16 Case setting for dynamic analysis

3.3.2.1 Effect of time-step-size

To check the effect of time-step-size to the numerical results, the rolling moment simulation is tested for ΔT of 0.01, 0.001 and 0.0001 s. The angular velocity (Ω) is set at 1 deg/s and the simulation is stopped when it reach the roll angle (ϕ) of 0.1°. The results suggest that the change in the time-step-size has a very minute influence ($\leq 1\%$) on the prediction of the aerodynamic forces and moments (Table 3-17).

Time step size (s)	C_x	C_y	C_z	C_l	C_m	C_n
0.01	0.0951	0.0724	0.0035	-0.0041	-0.0002	0.0017
0.001	0.0952	0.0725	0.0035	-0.0041	-0.0002	0.0017
0.0001	0.0960	0.0731	0.0033	-0.0042	-0.0002	0.0017

Table 3-17 Aerodynamic properties of the VTOL MAV for various time-step-sizes

3.3.2.2 Effect of discretization scheme

The first and second order upwind schemes are tested to study the effect of discretization scheme. The case tested is roll rate simulation with $\Phi = 5^\circ$, $\Omega = 0.2$ deg/s and $\Delta T = 0.1$ s. The steady solution of the MAV for the same roll angle is used as a reference. The discretization scheme that predicts the aerodynamic forces and moments closest to the steady solution is preferred due to its higher accuracy. The results reveal that the second order upwind is more accurate and thus will be used in the next simulations (Table 3-18).

Coefficient	Steady	Upwind	
		1st order	2nd order
C_x	0.0950	0.1182	0.0950
C_y	0.0718	0.0740	0.0717
C_z	0.0096	0.0103	0.0096
C_l	-0.0042	-0.0043	-0.0041
C_n	0.0017	0.0018	0.0017
C_m	-0.00001	0.0002	0.00002

Table 3-18 Aerodynamic forces and moments for various discretization schemes

3.3.2.3 Effect of pressure-velocity coupling

The Semi-Implicit Method for Pressure-Linked Equations (SIMPLE), SIMPLE-Consistent (SIMPLEC) and Pressure Implicit Splitting of Operators (PISO) are tested to analyze the effect of pressure-velocity coupling. Each case is set at $\Phi = 5^\circ$, $\Omega = 1$ deg/s and $\Delta T = 0.1$ s. Simulations which use PISO as the pressure-velocity coupling are unable to converge, and thus cannot be used in this analysis. The results for the other two schemes are compared to the steady solution of the same roll angle. The cases using SIMPLEC over-predicts C_z by 2% and C_l by 5%, whereas SIMPLE only over-predicts C_l by 2% (Table 3-19). Therefore, the dynamic simulations are run using SIMPLE as the pressure-velocity coupling.

Coefficient	Steady	SIMPLE	SIMPLEC
C_x	0.0950	0.0950	0.0950
C_y	0.0718	0.0718	0.0718
C_z	0.0096	0.0096	0.0098
C_l	-0.0042	-0.0040	-0.0040
C_n	0.0017	0.0017	0.0017
C_m	-0.00001	-0.000002	0.000001

Table 3-19 Aerodynamic forces and moments for various pressure-velocity couplings

Chapter 4 Numerical Analysis on Aerodynamics and Stability of the VTOL MAV

4.1 Aerodynamic performances and static stability analysis

4.1.1 Computational result discussion of the VTOL MAV-baseline

The computations of the VTOL MAV-baseline are performed before the VTOL MAV and provide important information. Trouble shootings during the VTOL MAV-baseline analysis become the guidelines of the VTOL MAV simulations. The results of the drag, the lift and the pitching moment predictions are analyzed below to give some examples of the important points taken from the VTOL MAV-baseline simulations. The complete simulation results of the VTOL MAV-baseline are given in Appendix A.

Similar to the duct computational model, the VTOL MAV-baseline is modeled as a fixed and a rotated object to simulate the angle-of-attack and sideslip-angle (Section 3.1.2). However, unlike the validation case, both models are built using unstructured grid. The fixed and rotated models give almost identical results up to the angle-of-attack of 20° (Figure 4-1). The largest deviation is shown between 20° to 40° , where the stall is expected to occur. Both models predict the same stall angle-of-attack, around 40° . Nonetheless, the fixed model predicts higher $C_{L_{\max}}$ (around 10 %) compared to the rotated model. The result deviation is possibly caused by the inaccuracy of the code (Fluent) in predicting the separated flow/wake region developed in the domain.

The curve for the fixed model is cut at 60° because the iteration for higher angles-of-attack is difficult to converge. At 70° and 80° , the fixed model still converges and gives results. However, the iteration requires much longer time and the residual shows an oscillatory pattern, which indicates a numerical instability. At 90° , the fixed model cannot reach convergence. This is probably due to the lack of match between the boundary conditions used in the model and the flow direction. The flow enters the velocity inlet boundary, shown with blue color in Figure 3-39, and should leave through the outflow (red color). This configuration cannot accommodate 90° simulation because of the large wake region in the domain. The flow enters in y^+ direction hits the velocity inlet surface at the top and takes dramatic turns to move to the outflow. As a result, reversed flow exists throughout the domain and steady solutions are not achieved.

Figure 4-1 Lift vs. Angle of attack for various computational models of VTOL MAV-baseline

Figure 4-2 Drag vs. Angle of attack for various computational model of VTOL MAV-baseline

The C_D predictions of the fixed and rotated models are in a very good agreement up to 40° (Figure 4-2). Similar to the C_L predictions, the deviation becomes clearer as the angle-of-attack increases. The fixed model can predict the drag well up to 60°, but it encounters difficulty in converging for the higher angles-of-attack. On the other hand, the rotated model can predict the C_L and C_D for the entire range of angles-of-attack tested.

Figure 4-3 Pitching moment vs. Angle of attack for various computational models of VTOL MAV-baseline

The fixed and rotated models give very close predictions of the pitching moment (C_m) variations with respect to angle-of-attack (Figure 4-3). As expected, the largest difference exists at the stall region, which is between 20° to 40°. Nonetheless, the difference is of a very small order of 10^{-2} . These results of the VTOL MAV-baseline simulation lead to the formulation of an optimized methodology to analyze the VTOL MAV. The fixed model is used from 0° to 60°, while the rotated model is used from the angle-of-attack of 70° to 90°. The sideslip angle is simulated for a smaller range which is from 0° to 40°, thus only the fixed model is used.

4.1.2 Computational result discussion of the VTOL MAV

The model is simulated for a wide range of angles-of-attack and sideslip angles. The angle-of-attack varies from 0° to 90°, while the sideslip-angle varies from 0° to 40° with 10° increments. The model is simulated with and without the propeller. In a real flight, the pressure jump, which corresponds to specific amount of thrust, is changing with the airplane attitude. However, the pressure jump of the actuator disc is set fixed at 61.6 Pa for each angle tested. The propeller effects on the aerodynamic and static stability characteristics of the VTOL MAV can be investigated by comparing the results from the MAV simulation when the propeller is included (with propeller) and when it is not (without propeller). Once the effects are known, it can be applied to any amount of thrust. The detail results of VTOL MAV simulations are given in Appendix B.

4.1.2.1 Aerodynamic performance of the VTOL MAV

The C_L predictions of VTOL MAV are shown in Figure 4-4. The $C_{L_{\max}}$ of the VTOL MAV without the propeller simulated is 1.08. The $C_{L_{\max}}$ is reached at the angle-of-attack of 30° , higher than the typical stall angle-of-attack of 10° - 15° for conventional airplane flying in high Re regime. The high stall angle-of-attack is one of the characteristics of low aspect ratio lifting surfaces [5, 24]. The propeller increases the lift by approximately 10%, which results in higher $C_{L_{\max}}$ of 1.22. The mass flow rate, which corresponds to the freestream velocity, increases when the propeller is operating. The higher magnitude of the freestream velocity generates more lift.

Figure 4-4 Lift vs. Angle of attack of VTOL MAV

Moreover, a higher stall angle-of-attack is expected with the propeller installed inside the duct [4, 5, 10, 26]. However, the numerical results of the VTOL MAV show that the stall angle-of-attack is 30° , which is the same for both models. This may be caused by the actuator disc model which is used as a simplification of the propeller system. The actuator disc theory is based on several assumptions, namely the flow is incompressible, inviscid, and axis-symmetrical. Consequently, this theory ignores the swirl of the flow [34, 36].

The aspect ratio of the duct is 1.33, while the aspect ratio of the wing is 2.32. Ducts and wings with aspect ratio of these values are categorized as low aspect ratio wings and ducts. Nonetheless, referring to Torres [24] and Cosyn [23] these aspect ratios are closer to the upper limit of the category. As a lift-curve becomes increasingly linear with the increasing of aspect ratio, the lift curve of the VTOL MAV is expected to be more linear [22, 24]. The expected

characteristic is shown by the MAV's lift curve which tends to be linear particularly from 0° to 20° (Figure 4-4).

Figure 4-5 Lift vs. Angle of attack for VTOL MAV and iSTAR 29i

The lift curve of the VTOL MAV follows the similar pattern as the iSTAR 29i vehicle which is numerically analyzed using AVID OAV software (Figure 4-5). In small angle-of-attack region, the lift curve of the iSTAR 29i is more linear than the VTOL MAV. This is due to the fact that the duct aspect ratio of the iSTAR 29i is 2, higher than the VTOL MAV. For both vehicles, nonlinearity becomes clearer as the angle-of-attack increases. The source of the nonlinear lift is the tip vortex which creates low pressure region on the upper surface of lifting platforms, such as the wing and the duct (Figure 4-6) [4, 19].

Both the VTOL MAV and iSTAR 29i have flat curves near and at the stall angle (30°-40°). This phenomenon is common for low aspect ratio wings and MAVs [4, 5, 10, 24, 25]. At high angles-of-attack (above 33°) a large separated vortex stays above wings and forms a stationary vortex [57]. The vortex is called stationary because it stays above the wing instead of shedding downstream into the wake. This stationary vortex creates low pressure region on the upper surface of lifting surfaces, results in higher lift, postponed stall and gradual lift loss at angles-of-attack beyond stall angle-of-attack.

Figure 4-6 Vortex at the duct-wing junction and the wing tip of VTOL MAV

As expected, the C_D increases proportional to angles-of-attack (Figure 4-7). In general, the drag increases when the propeller is included in the simulation. The C_{D0} of the VTOL MAV without the propeller is 0.097 and it increases to 0.109 under the propeller influences. In average, the propeller increases the drag coefficient by about 0.066 with highest increment occurs at high angles-of-attack, 30°-80°. The total drag is dominated by skin friction, form/pressure and induced drag. The skin friction drag is of great concern in a viscous dominant flow like the VTOL MAV case. The form/pressure drag has a great influence in high angle-of-attack region where separated flow dominates the flow field. The induced drag is also important because of the tip vortex which covers the VTOL MAV's surface.

Figure 4-7 Drag vs. Angle of attack of VTOL MAV

A ducted/shrouded propeller generates higher lift but also higher drag, and a lower lift-to-drag ratio (L/D) at low angles-of-attack [58]. The L/D at the high angles-of-attack is higher when the propeller is operating. The simulation results of the VTOL MAV show the predicted performances (Table 4-1). The maximum L/D is achieved at the angle-of-attack of 10°. Fletcher [7] mentioned 5° as the stall angle-of-attack for duct configuration [7]. To confirm his observation, additional test for the VTOL MAV is conducted at the angle-of-attack of 5° without simulating the propeller. However, the results show that the L/D at this angle is lower than the value at the angle-of-attack of 10°, which is not confirming the previously mentioned Fletcher's observation.

Angle-of-attack (deg)	Lift-to-drag ratio (L/D)	
	without propeller	with propeller
0	0.75	0.64
10	3.62	3.33
20	2.39	2.40
30	1.57	1.59
40	1.16	1.17
50	0.85	0.85
60	0.60	0.59
70	0.36	0.38
80	0.21	0.24
90	0.06	0.09

Table 4-1 Lift-to-drag ratio (L/D) of VTOL MAV with and without propeller

4.1.2.2 Static stability characteristics of the VTOL MAV

The pitching moment (C_m) variation to angle-of-attack is shown in Figure 4-8. The pitching moment coefficient has a negative slope of -0.0034/deg or -0.19/rad and C_{m0} of 0.000434. The

negative slope of the C_m vs. α curve implies the VTOL MAV tendency to develop a restoring pitching moment after a pitch disturbance. The value of C_{m0} is positive which means the VTOL MAV can be trimmed at positive angle-of-attack. Based on the criterion, the VTOL MAV possesses longitudinal static stability during cruise.

Figure 4-8 Pitching moment vs. Angle of attack of VTOL MAV

The rolling moment coefficient generally has a negative slope with respect to the sideslip angle, although at small angle it has the tendency to be unstable (positive slope) (Figure 4-9). The negative slope indicates that after a roll disturbance the VTOL MAV generates restoring rolling moment to reduce the sideslip angle. The C_l vs. β curve has a positive slope, which indicates instability in lateral mode, only between the sideslip angle -10° and 10° when the propeller is off (Figure 4-10). The C_l vs. β curve with the propeller operating shows a positive slope from sideslip angle -20° to 20° . These findings leads to the conclusion that at low sideslip angles (-20° to 20°) the propeller destabilizes the VTOL MAV in the lateral mode.

Figure 4-9 Rolling moment vs. Sideslip angle of VTOL MAV

Figure 4-10 Rolling moment vs. Sideslip angle of VTOL MAV for small sideslip angles

At low sideslip angles (-20° to 20°) and with the propeller off, the rolling moment coefficients of the VTOL MAV show an oscillatory pattern. This pattern is common for MAVs, as found in Black Widow [9], Zagi and Dragonfly [41]. In the same region, the propeller shifts the C_l vs. β curve downwards and makes it less scattered. However, the slope of the curve is positive, which means the VTOL MAV is statically unstable in the lateral mode. This characteristic is possibly caused by a phenomenon known as tip vortices' destabilization. Numerical and experimental investigation show that after a certain angle (critical angle) the separated regions above the wings become bilaterally asymmetric [57]. Following this phenomenon, the tip vortices also become bilaterally asymmetric and start to swing on the wing surfaces in an irregular manner. The instability of these vortices affects the stability of the VTOL MAV. It creates unequal/asymmetric drag and lift which produce yawing and rolling moment respectively.

Figure 4-11 Yawing moment vs. Sideslip angle of VTOL MAV

The VTOL MAV possesses directional static stability because it tends to generate yawing moment to counteract a sideslip disturbance. The C_n vs. β curve show a positive slope which meets the directional static stability criterion of $\frac{dC_n}{d\beta} > 0$ (Figure 4-11). The C_{n_β} value of the VTOL MAV is 0.016/deg, significantly lower than the value for many small-conventional airplanes which ranging from 3/deg to 9/deg [59]. The flow physics behind the very low value of C_{n_β} will be explained in Chapter 5. The propeller slightly increases C_{n_β} to 0.018/deg. However, the increase of the slope is too small to conclude that the propeller stabilizes the VTOL MAV in the directional mode.

4.1.3 Computational result discussion of the lifting/control surfaces

The VTOL MAV consists of multiple lifting/control surfaces that each uniquely contributes to the aerodynamic and stability characteristics of the vehicle. In Section 4.1.1 and 4.1.2, the full MAV configuration with the wing set at 6° , the horizontal tail at -2° , the vertical tail at -1° and the stator set at $-10^\circ - 10^\circ$ is simulated. The aerodynamic coefficients of this trim configuration are used as the reference data for the numerical study of the lifting and/or control surfaces. To further investigate each component, computational models are built and simulated to account for various design parameters such as setting angle and aspect ratio. The trend observed in the numerical results show the effect of the lifting/control surfaces configuration. The following results are for steady case with Spalart-Allmaras as the turbulence model, the freestream

velocity set at 15 m/s and propeller-off conditions, unless otherwise mentioned. The complete simulation results are presented in Appendix C.

4.1.3.1 Effect of the wing incidence angle

The wing is designed to have a variable incidence (pitch movable) to reduce altitude variations when the vehicle is transitioning from hover to cruise mode or vice-versa. To study this feature, two full computational models with the wing set at 6° and 10° are evaluated. The analysis focuses on the aerodynamic performances and the longitudinal static stability of the vehicle. For both simulations, the freestream velocity (V_∞) is set at 15 m/s and the propeller is off.

The results show that increasing the incidence angle from 6° to 10° results in a higher lift and also a higher drag for the VTOL MAV (Figure 4-12). Relative to the original configuration (6°), at the angle-of-attack of 0°, the lift coefficient increases by 60% while the drag coefficients increases by only 3.24% when the wing is set at 10°. At the angle-of-attack of 10°, the lift coefficient increases by 7.2% while the drag coefficient increases by 9.8%. These results indicate that the variable-incidence feature can improve the aerodynamic performances of the VTOL MAV at low angles-of-attack (lower than 10° in this case). It is also found that the incidence angle of the wing has an insignificant influence on the longitudinal static stability of the vehicle. For both incidence angles tested, the VTOL MAV is statically stable in the longitudinal mode, as shown by the two curves overlapping in Figure 4-12.

Figure 4-12 Effect of wing incidence angle

4.1.3.2 Effect of the horizontal tail deflection

The horizontal tail of the MAV is an all-movable lifting surface that is not only generates lift, but is also used to control and to stabilize the vehicle in the longitudinal mode. The horizontal tail can be set to produce nose-up or nose-down pitching moments to stabilize the vehicle. In the VTOL MAV trim configuration, the horizontal tail has a negative setting angle (δ) of -2° . In this case, the horizontal tail generates a nose-up pitching moment to stabilize the nose-down pitching moment generated by the wing. A total of 15 simulations have been performed to study the influences of the horizontal tail to the aerodynamic performances and the longitudinal static stability of the full vehicle.

The results show that the horizontal tail contributes only to a small percentage of the total lift of the vehicle. The lift-curve shows that the vehicle's lift only differs slightly with the change in the setting angle of the horizontal tail. A positive increase of the setting angle results in a higher lift of the vehicle and vice versa. The C_L vs. α curve for $\delta = 0^\circ$ shifts upward when δ is increased to

2°, whereas the same curve shifts downward for $\delta = -2^\circ$. The drag, which behaves in a similar fashion to the lift, increases with the increase of the horizontal tail setting angle (Figure 4-13).

The shift of the C_m vs. α curve follows the behavior of the lift. The lift of the horizontal tail becomes higher with the increase of δ . The pitching moment is the lift multiplied by the distance from the aerodynamic center (ac) of the horizontal tail to the center of gravity (cg) of the VTOL MAV. Therefore, a higher lift also creates a higher nose-down (negative) pitching moment.

Figure 4-13 Effect of horizontal tail deflection

4.1.3.3 Effect of the vertical tail deflection

The vertical tail is utilized mainly to control and to stabilize the VTOL MAV in the directional mode. The vertical tail can generate left or right yawing moments in accordance with the deflection/setting angle. The trim setting angle of the vertical tail (ϵ) is -1° . In the simulation, the setting angle is modified to 0° and 1° and tested for the angles-of-attack of 0° , 10° and 20° .

The simulation results show that the vehicle lift and drag only change slightly with the change in the setting angle of the vertical tail. In relationship with the sideslip angle, the contribution of the vertical tail is significant to the VTOL MAV. Increasing the setting angle of the vertical tail results in a less positive yawing moment (right yawing) and a more negative yawing moment (left yawing), and vice versa. However, the changes in the yawing moment magnitude do not affect the directional stability of the vehicle, as can be concluded from the relatively constant C_{n_β} (Figure 4-14).

Figure 4-14 Effect of vertical tail deflection

4.1.3.4 Effect of the stator

A study is conducted to analyze the effects of stators on the aerodynamic performances and static stability properties of the VTOL MAV. In the final design configuration, three stators with asymmetrical setting angle of 10° for the right part and -10° for the left are added. Two computational models are examined; one with the stators (Figure 4-15a) and the other without them (Figure 4-15b). Both models are evaluated from the angle-of-attack of 0° to 20° and from the sideslip angle of -10° to 10° . The resulting drag, lift, as well as pitching, rolling and yawing moments are analyzed to understand the effects of the stators.

Figure 4-15 VTOL MAV: with and without stators

The lift curve shows that the stators have a very small influence on lift (Figure 4-16). This result is easily predicted since the right and left stators are set asymmetrically at the same angle. Both parts of the stators produce the same amount of lift but in the opposite directions, which result in zero/very small increase of the vehicle's lift. The stators increase the drag of the VTOL MAV as they block the flow at the duct's outlet.

Figure 4-16 Effect of stator on the VTOL MAV lift, drag and pitching moment

The VTOL MAV is statically stable in the longitudinal mode with and without the stators, as can be seen from the C_m vs. α curve, which has a negative slope (Figure 4-16). The results also show that the stators do not affect considerably the pitching moment of the vehicle. This characteristic can be predicted after the lift data are obtained. The simulation results show that the stators only generate a very small amount of lift and thus will only have a small contribution on the overall pitching moment.

Figure 4-17 Effect of stator on the VTOL MAV rolling and yawing moments

The stators have a negligible influence on the directional static stability of the VTOL MAV as observed in the C_n vs. β curve (Figure 4-17). The asymmetrical stator is predicted to have a stabilizing effect on the vehicle static stability in the lateral mode. However, the numerical results show that the vehicle is statically unstable ($C_{l_\beta} > 0$) in the lateral mode with and without the stators. An explanation about the ineffectiveness of the stator will be given in Chapter 5.

4.1.3.5 Effect of stator blade-cascade configuration

Three layers of stator-blades with asymmetrical setting angle of 10° for the right part and -10° for the left are added to the VTOL MAV. The number of blade-layers and the setting angle of the stators are determined during the conceptual design phase [8]. Based on the empirical calculation conducted in the conceptual design phase, this configuration is expected to be able to generate sufficient rolling moment to counteract the propeller torque.

The objective of the numerical analysis is to study the influence of the number of blade-layers to the aerodynamics of the stators. Single, double and triple-layered stators are tested from the angle-of-attack of 0° to 20° and from the sideslip-angle of -10° to 10° (Figure 4-18). Each model of the stators is tested individually as an isolated component. Both in the full configuration and individual models, the stators maintain the same position relative to the other MAV's components. The moment coefficients are computed with respect to the center of gravity (cg) of the vehicle. The analysis is performed in the longitudinal and lateral-directional modes.

Figure 4-18 Geometry of one, two and three-bladed stators

The computational models of the stator are similar to the one explained in Section 3.2.2. The full domain is tube-shaped and has a sphere-inner-domain which surrounds the stator (Figure 4-19). The domain is meshed using unstructured grid of around 370,000, 500,000 and 627,000 tetrahedral cells for single, double and triple-layered stators respectively. These grid sizes are used because they give nearly uniform, for all stator models, wall y^+ value of 30, which is well within the recommended range [51].

Figure 4-19 Computational model and mesh of the triple-layered-stator

The lift of stators with different numbers of blade-layers is plotted in Figure 4-21. The lift coefficients of the multi-layered stators are higher than the single-layered stator at the angles-of-attack of 10° and 20° . This phenomenon occurs due to the cascade configuration of the multi-layered stators. The small gap between the layers increases the velocity of the freestream flow when it passes the stator, just like the flow in *venturi* pipe (Figure 4-20). The higher velocity of the flow results in a higher lift produced by the multi-layered stators. The single-layered stator with a low velocity flow around it shows a more nonlinear lift curve compared to the multi-layered stators.

Figure 4-20 Axial velocity of flow around single and double-bladed stators

The drag of the multi-layered stators is higher than the single-layered stator in the region of the angles-of-attack tested. The higher velocity caused by the cascade configuration of the stator results in a higher lift and also a higher drag. The multi-layered stators create a larger wake region, which leads to a higher pressure difference between the front and back parts of the component. The larger pressure difference between the upstream and downstream regions creates a higher pressure-drag. The drag increment of the double-layered stator compared to the single-layered one is the same as the drag increment of the triple-layered stator compared to the double-layered (Figure 4-21).

Figure 4-21 Effect of stator blade-cascade configuration on lift, drag and pitching moment

The C_m vs. α curve shows that all three stators generate a stabilizing pitching moment to counteract a pitch disturbance. The simulation results for the pitching moment coefficients are in agreement with the lift characteristics of the stators. In comparison with the single-layered stator, the multi-layered stators produce a higher lift and thus generate a higher nose down pitching moment. The pitching moment curve of the single-layered stator shows nonlinearity just like the lift curve.

The stators are tested from the sideslip angle of -10° to 10° , with 5° intervals. The C_l vs. β curve indicates that for the sideslip angles tested the stators generate a negative rolling moment (Figure 4-22). The production of rolling moment is dominantly influenced by the stator's lift. Therefore, the higher lift produced by the multi-layered stators creates more negative rolling moment as well. The pattern of the C_n vs. β curve can be associated to the C_D vs. α curve analyzed previously because drag is the dominant source of yawing moment. As the drag increases with the number of layers, the yawing moment produced by the stators also increases.

Figure 4-22 Effect of stator blade-cascade configuration on rolling and yawing moments

4.1.3.6 Effect of the wing aspect ratio

A further study is conducted to understand the flow around the coupled duct-wing. The influence of wing aspect ratio to the aerodynamic forces and moments of the wing as well as to the coupled duct-wing is analyzed numerically. It is predicted that the influence of tip vortex becomes less intense as the aspect ratio of the wing increases. The information about the computational model, procedure, and results are presented and analyzed next.

For both the wing and the duct-wing coupled configurations, four aspect ratios are tested in this study; 2, 4, 6 and 8 (Table 4-2). For each aspect ratio, there are three computational models need to be analyzed. These models are the individual wing, the individual duct and the duct-wing coupled (Figure 4-23). The wing has a rectangular platform with a constant chord of 0.1 m, while the span changes according to the aspect ratio tested (Figure 4-24). The duct is an annular airfoil with an aspect ratio of 1, which means it has an equal length of chord and diameter of 0.1 m. Both the duct and the wing use a NACA symmetrical airfoil, NACA 0009. The duct-wing coupled model is obtained by joining the wing and the duct with the x-body axis of both components is aligned together.

There are in total nine computational models and each model is tested for the angles-of-attack of 0° and 10° . The reference area (A) is the duct projection-area ($A = d \times c$), 0.01 m^2 . The reference length is the chord of the duct (c), 0.1 m. The aerodynamic coefficients are calculated with respect to these reference values.

Aspect Ratio (AR)	Chord (c) in m	Span (b) in m
2	0.1	0.2
4	0.1	0.4
6	0.1	0.6
8	0.1	0.8

Table 4-2 Detail geometry of duct and wing configurations

Figure 4-23 Scheme of individual and coupled configurations of duct-wing

Figure 4-24 Geometry of duct (left view) and wing (top view)

Hybrid type of grid with tetrahedral-hexahedral-cell combination is used in the duct and the duct-wing coupled models, while the individual wing is meshed using unstructured grid with only tetrahedral cells in the domain (Figure 4-25). The grid size used for the simulation models is between 350,000 and 1.2 million cells. The near wall grid distance (wall y^+) for these grid sizes is 30-70, which is within the suggested range [51].

Figure 4-25 Mesh of duct, wing and duct-wing coupled configurations

The individual simulation results of the wing are shown in Figure 4-26. It is found that the lift and drag of the wing increase as the aspect ratio of the wing becomes higher. The increase in the lift is higher than the drag rise; therefore the lift-to-drag ratio (L/D) of high-aspect-ratio wings is higher than the low-aspect-ratio ones. Similar findings are observed in the duct-wing coupled simulations as in the individual wing. Similar findings had also been observed in several works on the aerodynamics of low Re and/or low aspect ratio lifting surfaces [24, 25]

Figure 4-26 Effect of wing-aspect-ratio on lift and drag of wing

The maximum lift-to-drag ratio, $\left(\frac{L}{D}\right)_{\max}$, for the coupled configuration is lower than the wing alone. At first, the reason behind this result is predicted to be the reduced wing's lift due to the decrease of the wing's surface area after the integration with the duct. However, the results show that the lift of the wing as an individual component and as part of the duct-wing coupled configuration is nearly the same. What probably happens is that the drag rises when the wing is joined with the duct. The interference between the duct and the wing in the coupled configuration results in a rise in the drag but not in the lift, which leads to a lower lift-to-drag ratio (Figure 4-27).

Figure 4-27 Effect of wing-aspect-ratio on lift and drag of duct-wing coupled

4.1.4 Computational result discussion of the operating conditions

The MAV is designed to operate in certain “ideal” conditions. These ideal conditions are the standard operating conditions of the VTOL MAV. This study is conducted to analyze the non-standard operating conditions since there is a high probability that the MAV will operate in this region. The main parameters investigated here are related to velocity and the vehicle’s attitude. The full configuration model is used in all simulations to predict the aerodynamic forces and moments of the MAV flying outside the standard region. The complete results of the tests, which are under propeller-off conditions, can be found in Appendix D.

4.1.4.1 Effect of velocity

The forward velocity of the vehicle varies according to the flight conditions and also the maneuver performed during flight. The velocity variations might also be triggered by gust/wind disturbances. These situations show the need to investigate the effect of velocity to the aerodynamic and stability characteristics of the VTOL MAV.

The design cruise speed is 15 m/s, while at extreme high angles-of-attack (80° – 90°) the velocity can drop to as low as 1 m/s. Therefore, four different velocities are tested, namely 15 m/s, 10 m/s, 5 m/s and 1 m/s. These velocities correspond to Re_c of 1.54×10^5 , 1.02×10^5 , 5.13×10^4 and 1.02×10^4 respectively. The vehicle is simulated from the angle-of-attack of 0° to 90° with 10° intervals for each velocity tested. In order to have a clear display of the graphs, the simulation results shown here are limited to the angles-of-attack of 0° – 20° .

The C_L vs. α curve shows that the lift increases proportionally to the velocity. This behavior is also observed in other researches on the aerodynamics of low aspect ratio (AR) lifting surfaces at low Reynolds numbers (Re) [4, 26]. The computational results show that the drag decreases as the velocity increases. The results also suggest that velocity has only a small influence on the drag when the velocity is between 5 m/s and 15 m/s, where the points are overlapping. However, there is a significant increase in drag when the velocity decreases to 1 m/s (Figure 4-28).

The lift and drag results can be analyzed by creating two categories of chord-Reynolds number (Re_c): $Re_c < 5 \times 10^4$ and $Re_c \geq 5 \times 10^4$. The first category is for the velocity of 1 m/s, while the others are included in the second category. It is observed that a lifting surface with Re_c less than 5×10^4 will not experience laminar separation bubbles because the flow does not have enough energy to reattach to the surface after separation occurs [16]. A flow without laminar separation bubbles is expected to have a large separated region over the lifting surfaces. This separated region results in a significant loss of lift and increase in drag, as observed from the results for the velocity of 1 m/s ($Re_c < 5 \times 10^4$). On the contrary, for the second-category ($Re_c \geq 5 \times 10^4$) where the flow has enough energy to reattach, the lift is higher and the drag is lower than the first category.

The lift-to-drag ratio increases with velocity. A significant drop is observed when the velocity decreases to 1 m/s. This behavior is consistent with the physical phenomenon of the

corresponding Re region. At this low velocity, the large separated region creates a lower lift and a much higher drag, which leads to a poor L/D.

Figure 4-28 Effect of velocity variation

4.1.4.2 Effect of velocity during hover

The influences of velocity variations in the x-body-axis direction of the VTOL MAV at perfect hover conditions are investigated here. The velocity in the direction of the x-body axis of the vehicle is commonly termed as u (in lowercase or uppercase). Perfect hover conditions are

achieved when the vehicle is at hover attitude, as what is shown in Figure 4-29 and that the vehicle's velocity (translational and angular) is zero. In other words, the vehicle is in a motionless/stationary hover flight.

The vehicle with the propeller off is tested for five U variations of 1 m/s, 2 m/s, 3 m/s, 5 m/s, 8 m/s, and 10 m/s. These velocities correspond to Re_c of 1.02×10^4 , 2.05×10^4 , 3.08×10^4 , 5.13×10^4 , 8.21×10^4 , and 1.02×10^5 respectively. This Reynolds numbers range is set to cover the region before and after Re_c of 5×10^4 . This particular Re is deemed to be the critical Reynolds number, below which the flow around airfoil does not have sufficient energy to form laminar separation bubbles [16]. The formation of bubbles influences the size of separation region on the vehicle's surface and consequently the aerodynamic properties of the VTOL MAV.

Figure 4-29 Computation model of the VTOL MAV at perfect hover conditions

In this study, the forces and moment coefficients for different velocities are analyzed to understand the vehicle behavior at the perfect hover conditions. In order to do the analysis correctly, we need to consider the differences between the Fluent and the body-axis coordinate-system notations. Both Fluent and body-axis coordinate systems are Cartesian coordinate system, however the notations used are different. At hover flight, the body-axis coordinate system notation is shown in Figure 4-30a, while the Fluent notation is shown in Figure 4-30b.

In this study, lift is defined as the force which acts on x^+ axis according to the Fluent coordinate notation, whereas drag is defined as the force which acts on the y^- axis of the same coordinate

notation. The rolling moment is measured with respect to y^+ axis in the Fluent, whereas the yawing and pitching moments must be measured with respect to x^+ axis and z^+ axis respectively.

Figure 4-30 Comparison between body-axis (a) and Fluent (b) coordinate notation at hover-flight

The C_L vs. U curve shows that the lift of the vehicle increases when U rises from 1 to 3 m/s. The lift starts to drop when U increases above 3 m/s; however it rises again after 8 m/s. These results probably indicate that the laminar separation bubbles for the vehicle at perfect hover conditions occur when U is higher than 8 m/s. The lift increases in the beginning of the curve because the flow is still attached to the surface. The flow starts to separate when the velocity reaches 3 m/s, leading to the formation of a separated region and subsequently a drop in lift. Once the flow velocity reaches 8 m/s, it has enough energy to reattach to the surface and laminar separation bubbles are formed. The formation of laminar separation bubbles reduces the size of the separated region over the lifting surfaces, hence leads to the creation of a higher lift. This phenomenon is possibly the reason behind the rise in lift when U reaches 8 m/s (Figure 4-31).

The C_D vs. U curve shows that the drag drops with the increase in U . This trend appears possibly because the flow of higher velocities is less viscous than the lower one. Less viscous flow results in less friction, thus the drag generated by the vehicle is also lower. Another explanation behind this trend is that the laminar bubbles formed at these velocities create narrower separated region which result in a lower drag of the VTOL MAV.

Figure 4-31 Effect of velocity at perfect hover condition on lift and drag

The rolling moment of the VTOL MAV becomes more negative (left rolling) as U becomes higher. However, after U of 3 m/s, the rolling moment tends to be constant as what is shown by a nearly straight line in Figure 4-32. The C_m vs. U curve reveals that a less negative (nose down) pitching moment is produced when U increases from 1 m/s to 8 m/s. The yawing moment coefficient drops when U increases from 1 m/s to 2 m/s, after that it increases again. The positive values of C_n indicate that the VTOL MAV tends to yaw to the right side when U increases.

Figure 4-32 Effect of velocity at perfect hover condition on aerodynamic moments

4.1.4.3 Effect of cross-wind/side-wind

There are four different orientations (with respect to the wind direction) tested to study the crosswind effects at hover mode. The orientations are termed frontside, backside, left and right wing (Figure 4-33). For each orientation, three angles-of-attack are tested to obtain the slope of pitching moment curve. The slope of the C_m vs. α curve is then used to determine the longitudinal static stability of the VTOL MAV.

The computational results show negative values of C_{m_α} for all orientations (Figure 4-34). These results indicate that the VTOL MAV is statically stable in the longitudinal mode, under sidewind disturbances. The more negative value of C_{m_α} for the front and backside configurations show that they are relatively more stable compared to the others. This is due to the fact that the front and backside configurations generate higher drag which results in more negative (nose down) pitching moment.

Figure 4-33 Four orientations (with respect to wind direction) at the angle-of-attack of 90°

Figure 4-34 C_m vs. α of the VTOL MAV under the sidewind disturbances

The high drag for these two configurations is easily understood by observing the direction of the freestream velocity relative to the VTOL MAV. For the front and backside orientation, the freestream velocity is perpendicular to the wing. This condition creates a larger wake region which results in a higher drag compared to the left and right wing orientations where the wing is parallel to the flow (Figure 4-35).

Figure 4-35 Flow path line of the frontside and left wing orientation

4.2 Dynamic stability derivatives

The results and analysis of the MAV dynamic stability can be divided into three categories based on the type of moments: rolling, pitching and yawing. Complete simulation results are given in Appendix E.

4.2.1 Rolling-moment derivatives

In this category, there are C_{l_p} and C_{l_r} , the first is termed as damping-in-roll derivative and the later is rolling moment due to yawing derivative. The damping-in-roll derivative describes the vehicle resistance to rolling moment. In conventional aircraft, the component which has a strong influence on this derivative is wing. Wing in conventional aircraft as well as in the VTOL MAV is the main lift generator, thus it affects the rolling moment the most. A negative damping-in-roll (C_{l_p}) describes the tendency of the vehicle to generate rolling moment to counteract roll rate disturbances. The C_{l_r} is affected mainly by the position and the size of the vertical tail. This stability derivative becomes higher with the increase in the vertical tail size or the moment-arm-length.

Simulations of the VTOL MAV for two rates of roll (p) show that the vehicle has very small negative damping-in-roll derivatives (C_{l_p}) in the order of 10^{-4} (Figure 4-36). The C_{l_p} values, although they are negatives, are too small to determine the MAV's stability quality. The rolling moment coefficients due to yawing derivative (C_{l_r}) are shown in Figure 4-37. The curves show a positive slope for the negative yaw rate and vice versa.

Figure 4-36 Rolling moment vs. rate of roll (p) of the VTOL MAV

Figure 4-37 Rolling moment vs. rate of yaw (r) of the VTOL MAV

4.2.2 Pitching-moment derivatives

The coefficient C_{m_q} is termed as damping-in-pitch derivative. This stability derivative is dominantly influenced by the horizontal tail, although wing also contributes to it. This derivative is a negative constant for a stable aircraft. The negative sign indicates that the vehicle has a tendency to recover from a pitch disturbance by generating a pitching moment. The important parameters for this derivative are the size of the horizontal tail and the distance between the aerodynamic center of the tail and the aircraft's center of gravity (cg).

The values of C_m are very small with the average order of 10^{-4} (Figure 4-38). However, C_{m_q} is of a high order due to the very low pitch rates. The low rates of pitch are used because higher rates result in simulation divergence, which is possibly caused by the sliding mesh method used in this analysis. The sphere rotation along the y-body axis results in misalignment of the grid nodes between the inner and outer parts of the sphere. This misalignment might cause a problem in propagating information between the inner and the outer-sphere grid domains, which leads to the simulation divergence.

The simulation results show that C_{m_q} is a negative constant of -4.5/deg for $q = 0.0002$ deg/s and -1.28/deg for $q = 0.0005$ deg/s when a linear approximation is used. These C_{m_q} values are within the accuracy range of the flow solver, which has a maximum limit of 10^{-3} , and thus can be used to determine the vehicle stability. The negative values of damping-in-pitch derivatives suggest that the vehicle has pitch dynamic stability.

Figure 4-38 Pitching moment vs. rate of pitch (q) of the VTOL MAV

4.2.3 Yawing-moment derivatives

The two important stability derivatives are the damping-in-yaw derivative (C_{n_r}) and the yawing moment due to rolling derivative (C_{n_p}). Under normal circumstances, both the wing and the horizontal tail contribute to these derivatives. The damping-in-yaw derivative is negative when the aircraft has the tendency to generate a yawing moment to recover from a yaw disturbance. The yawing moment due to rolling derivative (C_{n_p}) and the rolling moment due to yawing derivative (C_{l_r}) are some of the cross derivatives which create the coupling between rolling and yawing moments.

The C_n vs. r curve show a negative slope for both rates of yaw tested (Figure 4-39). The damping-in-yaw derivative for $r = 0.0005$ deg/s is -1.4/deg, whereas for $r = 0.001$ deg/s is -0.707/deg. It can be observed that double increment of the yaw rate results in the drop of C_{n_r} into half of its original value. These negative values of C_{n_r} indicate that the VTOL MAV has yaw dynamic stability. The yawing moment due to rolling derivative (C_{n_p}) of the vehicle is very small, in the order of 10^{-5} (Figure 4-40). Therefore, the C_{n_p} values are not meaningful and should not be used to analyze the dynamic stability of the MAV.

Figure 4-39 Yawing moment vs. rate of yaw (r) of the VTOL MAV

Figure 4-40 Yawing moment vs. rate of roll (p) of the VTOL MAV

4.3 Summary

The five main components of the VTOL MAV are analyzed separately due to their distinct influence on the vehicle aerodynamic and stability characteristics. The numerical analysis of each component might include several design parameters such as setting angle and aspect ratio. Rebuilding and testing of the full-vehicle computational model and the corresponding grid domain are required to analyze a change in any of the design parameters. Rebuilding the full computational model is very complex and time-consuming. Moreover, solving the flow around each model simulated is too costly computationally. For that reason, an alternative method that can provide fast and reasonably accurate predictions on the aerodynamic forces and moments of the VTOL MAV is necessary. To meet such need, an innovative approach called the Improved-Component Buildup Method (I-CBM) is proposed in this study. A detailed explanation on I-CBM is presented in the next chapter.

Chapter 5 Improved-Component Buildup Method (I-CBM) for Fast Prediction of the Aerodynamic Properties and Stability Derivatives of the VTOL MAV

5.1 Background

One of the most common tools used in the aerodynamic prediction during the conceptual design phase of a conventional aircraft is the Component Buildup Method (CBM). In this method, each component of an aircraft is investigated individually and independently and then the aerodynamic coefficients of the vehicle are calculated by adding the aerodynamic coefficients of its components.

The early formulation of CBM is presented by Pitts, Nielsen, and Kaattari [60] in their paper about the interactions of circular-cylindrical bodies with various types of wings or tails. In the analysis, the authors suggest that the lift of the wing-body-tail combination can be estimated by the sum of lift from its principal components. The interference effects between the components are estimated using an interference factor called the foregoing ratio. CBM is intensively applied and developed for missiles and conventional-aircraft designs. CBM is used to predict the missile aerodynamics with circular and noncircular cross sections [61]. The method has also been used to investigate the interactions between a canard and thick bodies for missile configurations [62]. A group of researcher developed an integrated aerodynamic analysis tool, a CBM-based tool, to analyze rocket systems [63]. In the conventional-aircraft design process, Roskam [6] uses CBM to predict the aerodynamic properties of conventional aircraft.

Research in the application of CBM in the conceptual design phase of a MAV has only been done recently. CBM-based software, AVID OAV, is developed to predict the aerodynamics of ducted fan MAVs [10]. In predicting the aerodynamic properties, AVID OAV uses empirical data instead of computational results. This software is also designed only for ducted-fan MAVs with no wing and inside-the-duct control vanes. In this work, a case where the MAV may be equipped with the wing and outside-the-duct control vanes is considered.

5.2 Proposed Method

The lack of analytical and empirical models to predict the aerodynamic properties of a VTOL MAV makes a quick design analysis difficult. Thus, the aerodynamic design analysis is usually

based more on computational approach (CFD). The VTOL MAV aerodynamics can be simulated fully or per component (CBM) computationally.

Full configuration simulations are usually complex and costly to account for various degrees of freedom in the design envelope. A change in any of the component configuration, setting angle for example, will require rebuilding of the full computational model and the associated mesh. Moreover, solving for the flow around each new configuration investigated will be extremely costly computationally, which may restrict the number of design cases evaluated. The individual and independent (component) simulations (CBM), which will be referred as individual simulation/analysis for the rest of the report, are less complex and less costly than the full configuration. The whole configuration aerodynamics can then be calculated by summing the individual/component aerodynamics. Although interesting in terms of simplicity and significantly lower cost, the results obtained using such method is found to be inaccurate due to the negligence of the mutual aerodynamic interferences between the lifting surfaces.

A more sophisticated approach, called the Improved Component Buildup Method (I-CBM), is proposed here. In comparison to the original method (CBM), the I-CBM includes the interference effects by introducing the correction models (Figure 5-1). In I-CBM, similar to the CBM, the VTOL MAV is decomposed into its major components: duct, wing, and stator, horizontal and vertical tails. Three computational tests are set up to validate the method. The first set is for the full configuration, the second one is for the duct-wing (coupled), whereas the last set consists of five cases; one for each component of the VTOL MAV. In I-CBM, the aerodynamic properties of the VTOL MAV are calculated by adding the duct-wing (coupled) results and the corrected-CBM results of the rest of the components.

Figure 5-1 Schematic diagram of different computation methods

5.3 I-CBM for prediction of aerodynamic properties

Three types of analysis are performed in this study: full configuration, individual and coupled duct-wing. In the individual analysis, each component of the VTOL MAV is simulated individually and independently without the influence of other components. For that reason, each component in the individual analysis experiences flows with freestream angles-of-attack (α). The full configuration analysis includes all the components of the VTOL MAV, such that the flow around a component is perturbed by others. The angle-of-attack of the flow over a component is affected by the interference effects with other components. For analysis purpose, the perturbed angle-of-attack is termed as local angles-of-attack (α_{local}) in this study. The duct-wing coupled analysis is performed to study the mutual aerodynamic interference effects between the two components.

In the full configuration analysis, the interference effects between the duct-wing and the stator and also between the stator and the empennage are the main interests. The duct has an aligning effect to the vehicle's x-body axis for the flow that passes through it. Therefore, the local flow leaving the duct and entering the stator is straightened in the direction of the x-body axis, despite of the angle-of-attack of the freestream flow (Figure 5-2). The flow passes the stator is subjected to a lower local angle-of-attack compared to the freestream one, thus the lift and the drag coefficients are lower than the values provided by the individual analysis at the freestream angle of attack. The stator which has a cascade configuration will further change the direction of the flow before it passes the horizontal and vertical tails. As a result of lower local angles-of-attack, the tails produce lower drag and also lower lift compared to the individual analysis results at the same free-stream angle of attack. Consequently, a correction model is required to use the CBM accurately in predicting the aerodynamic properties of the VTOL MAV.

Figure 5-2 Flow over the stator in individual and full configuration computational models

The VTOL MAV configuration is divided into three major sections for the analysis purpose. The first section is the coupled duct and wing. The second section is the stator, while the rest of the components are grouped into the third section. Following the division of the VTOL MAV into three sections, three parameters are introduced in the analysis: α , γ and δ . The freestream angle-of-attack (α) of the VTOL MAV is the relative angle between the x-body axis and the freestream velocity vector. The second parameter (γ) is the setting angle of the stators, whereas γ_L is for the left plane and γ_R is for the right. The setting angle of the horizontal tail is symbolized as δ . The VTOL MAV with the propeller off is tested for different α , γ and δ to check the consistency of I-CBM in various configurations (Equation 5-1).

$$C_{L_{VTOL_MAV}}(\alpha, \gamma_L, \gamma_R, \delta) = C_{L_{DUCT+WING}}(\alpha) + C_{L_{STATOR}}(\alpha, \gamma_L, \gamma_R) + C_{L_{HORIZONTAL_TAIL}}(\alpha, \gamma_L, \gamma_R, \delta)$$

$$C_{D_{VTOL_MAV}}(\alpha, \gamma_L, \gamma_R, \delta) = C_{D_{DUCT+WING}}(\alpha) + C_{D_{STATOR}}(\alpha, \gamma_L, \gamma_R) + C_{D_{HORIZONTAL_TAIL}}(\alpha, \gamma_L, \gamma_R, \delta) \quad (5-1)$$

There are three configurations tested to study the interference effects between the duct and the wing. The first configuration is the full configuration, which is used as the reference for all the simulations (Figure 5-3). For the duct-wing interference study, only the aerodynamic properties of the duct and the wing are taken from the full configuration simulation results. The second configuration is the coupled duct-wing, whereas the last one is the duct and the wing individually.

Figure 5-3 Full configuration breakdown of the VTOL MAV

Based on γ , the computational model for stator can be divided into three different categories: symmetrical, asymmetrical-equal and asymmetrical-unequal. Symmetrical stator model has the

same γ_L and γ_R . For example, if γ_L is 0° then γ_R is also 0° . The asymmetrical-equal model has the same magnitude of γ but with different sign for the left and right planes of the stator. For the asymmetrical-equal model, if γ_L is -10° , then γ_R is 10° . The last model is the asymmetrical-unequal with different setting angle for the two planes of the stators. The right plane might have γ of 0° while the left plane's γ is 5° . A total of six configurations are tested. Two configurations for the symmetrical category are $0^\circ - 0^\circ$ and $10^\circ - 10^\circ$ ($\gamma_L - \gamma_R$). For the asymmetrical-equal category, two configurations with γ variations of $-5^\circ - 5^\circ$ and $-10^\circ - 10^\circ$. The last category, the asymmetrical unequal, is represented by two configurations $0^\circ - 5^\circ$ and $10^\circ - 0^\circ$. The horizontal tail tested has δ of -2° , 0° and 2° . Both individual and full configuration simulations are performed to study the influence of δ to the aerodynamic properties of the VTOL MAV.

5.3.1 Validation

A total of 23 computational models which consist of 12 individual and 11 full configuration models are created to validate CBM (Table 5-1). Using FLUENT, each model is tested for three different angles-of-attack of 0° , 10° and 20° (Appendix F). The individual and full configuration simulation results of a particular component are compared to build the correction models. The corrected equations are taken from the linear approximation of the full configuration analysis and marked with asterisk (*) for the rest of the chapter. In this work, the corrected equations are for the angle-of-attack range from 0° to 10° .

Configuration	Setting angle
Duct	0°
Wing	0°
Duct-Wing (coupled)	0°
Stators ($\gamma_L - \gamma_R$)	
Symmetrical	$0^\circ - 0^\circ$
	$10^\circ - 10^\circ$
Asymmetrical-equal	$-5^\circ - 5^\circ$
	$-10^\circ - 10^\circ$
Asymmetrical-unequal	$0^\circ - 5^\circ$
	$10^\circ - 0^\circ$
Horizontal tail (δ)	-2°
	0°
	2°

Table 5-1 various configurations of computational models

5.3.2 Interference effects of the duct and the wing

The computational results show that the lift and drag predictions for the individual simulations (CBM) are significantly higher than the coupled or the full configuration (Figure 5-4). The results

clearly show that the interference effects of the two components need to be taken into consideration. On the contrary, the lift predictions of the coupled duct-wing and the full configuration models are in a very good agreement. Both the coupled and full computational models include the interference effects, and thus the coupled configuration must be used to accurately predict the aerodynamic properties of the duct-wing configuration (Equation 5-2).

$$C_L(Duct _ and _ Wing) \neq C_{L_{DUCT}} + C_{L_{WING}}$$

$$C_L(Duct _ and _ Wing) = C_{L_{DUCT+WING (COUPLED)}} \quad (5-2)$$

Figure 5-4 Lift and drag of duct-wing coupled

The full configuration, which is the reference model, predicts the lift (C_D) and drag (C_L) coefficients of the duct-wing configuration equal to 0.0819 and 0.5233 respectively. As shown in Table 5-2, the predicted values of the aerodynamic coefficients using the I-CBM are significantly closer to the reference values than the counterpart values predicted using the individual/standard CBM. The total drag coefficient is overestimated with less than 12% via I-

CBM while the CBM prediction gives a value of more than 51% higher than the reference value. Similarly, the lift coefficient is over predicted with less than 2% using the I-CBM. On the contrary, C_L is overestimated by 52% when the standard CBM is used.

Method	Total C_D	Total C_L
Full Configuration	0.0819	0.5233
I-CBM (Coupled Duct & Wing)	0.0916	0.5294
CBM (Uncoupled/Individual Duct & Wing)	0.1244	0.7953

Table 5-2 Aerodynamic performance at the angle-of-attack of 10° for various methods of prediction

5.3.3 The correction model for the aerodynamic force coefficients of the stator

The lift and drag predictions of the individual asymmetrical-equal stator model (CBM) are higher than the full configuration (Figure 5-5). A consistent pattern of deviation of the individual (CBM) simulation results from the full configuration ones is also found in the aerodynamics coefficient predictions. This finding supports the preliminary observations that there is a distinct relationship between the individual and the full configuration predictions that can be generalized and applied for a wide range of design cases. Similar findings are observed in the computational results of the symmetrical (Figure 5-6) and asymmetrical-unequal (Figure 5-7) models as the one in the asymmetrical-equal model simulations.

Figure 5-5 Lift and drag of asymmetrical-equal stator model

Figure 5-6 Lift and drag of symmetrical stator model

Figure 5-7 Lift and drag of asymmetrical-unequal stator

The individual and full configuration simulation results for one of the asymmetrical-equal stator model ($-10^\circ - 10^\circ$), angle-of-attack from 0° to 10° for example, are approximated using linear relationship. The linear approximation results in linear equations shown in Figure 5-5. The second set of equations for the full configuration analysis, which is the corrected equation and marked by asterisk (*) sign, shows a significant decrease of the lift-curve-slope and drag-curve-slope compared to the individual one (Equation 5-3). These findings show that the freestream angle-of-attack has a small influence on the aerodynamic properties of the stator. It also validates our assumption that the duct redirects the flow that passes the stators, creates a much lower local angle-of-attack.

$$\frac{\partial C_{L_{STATOR}}^*}{\partial \alpha} \lll \frac{\partial C_{L_{STATOR}}}{\partial \alpha}$$

$$\frac{\partial C_{D_{STATOR}}^*}{\partial \alpha} \lll \frac{\partial C_{D_{STATOR}}}{\partial \alpha} \quad (5-3)$$

5.3.4 The correction model for the aerodynamic force coefficients of the horizontal tail

The computational results indicate that the angle-of-attack of the vehicle (freestream angle-of-attack) has a small influence on the aerodynamic properties of the horizontal tail. One can notice from the variations of C_L vs. α and C_D vs. α curves in Figure 5-8 that the drag and the lift of the horizontal tail with various setting angles (δ) only change slightly with the increase of the freestream angle-of-attack. These results show that the duct and the stator redirect the flow before it passes the horizontal tail. As a result, the local angle-of-attack of the horizontal tail is much smaller than the freestream one. A lower local angle-of-attack results in a lower drag and lift produced by the horizontal tail. These findings affirm the need of a correction model for the CBM to be used to accurately predict the contribution of the horizontal tail to the overall aerodynamic properties of the VTOL MAV.

Figure 5-8 Lift and drag of horizontal tail

The correction model is built by comparing the equations obtained from the linear approximation of the individual and the full configuration results. The equations from the full

configuration analysis, which are the corrected-equations and marked by asterisk (*) in Figure 5-8, show a significant decrease of the lift and drag curve-slope compared to the individual analysis. The C_L vs. α curve for the horizontal tail set at 0° for example, the curve-slope for the full configuration analysis is 5.24×10^{-5} much lower than 4.46×10^{-3} for the individual analysis. It shows that the aerodynamic properties of the horizontal tail in the VTOL MAV are not dictated by the magnitude of freestream angles-of-attack. The horizontal tail generates significantly lower lift and drag as part of the MAV than as an individual component due to the low local angles-of-attack caused by the duct and the stator's cascade configuration (Equation 5-4).

$$\frac{\partial C_{L_{HORIZONTAL_TAIL}}^*}{\partial \alpha} \lll \frac{\partial C_{L_{HORIZONTAL_TAIL}}}{\partial \alpha}$$

$$\frac{\partial C_{D_{HORIZONTAL_TAIL}}^*}{\partial \alpha} \lll \frac{\partial C_{D_{HORIZONTAL_TAIL}}}{\partial \alpha} \quad (5-4)$$

5.3.5 Conclusions

After the duct-wing coupled model is analyzed and the correction model for stator and horizontal tail are formulated, the I-CBM can be applied to predict the aerodynamic properties of the VTOL MAV. The corrected values for the angle-of-attack of 10° , highlighted in Table 5-3, are calculated using the correction model for the stator and the horizontal tail. The rest of the components are computed individually and independently. As shown in Table 5-3, I-CBM gives significantly more accurate results compared to CBM. The CBM over predicts the total lift and the total drag of the VTOL MAV by more than 60% and 40% respectively. Using the I-CBM, the total lift for the entire MAV is over predicted by less than 1% using the I-CBM, while the total drag is over predicted by 8%. The computational results for the angle-of-attack of 20° also show that the I-CBM provides significantly more accurate aerodynamic performance predictions compared to CBM (Table 5-4).

	CBM		I-CBM		Full configuration	
	CL	CD	CL	CD	CL	CD
Wing	0.527	0.065	0.529	0.092		
Duct	0.268	0.059				
Horizontal tail	0.045	0.013				
Vertical tail	0.003	0.009				
Stator	0.059	0.054	0.016	0.046		
Total	0.902	0.202	0.550	0.152	0.547	0.141
Error (%)	64.70	43.34	0.41	8.12		

Table 5-3 Comparison of results for freestream angle-of-attack of 10°

	CBM		I-CBM		Full configuration	
	CL	CD	CL	CD	CL	CD
Wing	0.814	0.263	0.826	0.285		
Duct	0.449	0.172				
Horizontal tail	0.087	0.029				
Vertical tail	0.006	0.009				
Stator	0.109	0.074	0.036	0.048		
Total	1.466	0.547	0.877	0.350	0.920	0.362
Error (%)	59.34	51.10	4.67	3.31		

Table 5-4 Comparison of results for freestream angle-of-attack of 20°

5.4 I-CBM for prediction of lateral-directional static stability derivatives

The application of the Improved Component Buildup Method (I-CBM) for fast prediction of the aerodynamic performances of the VTOL MAV has encouraged further analysis on the method. In the previous study, I-CBM has demonstrated its ability in providing fast and reasonably accurate predictions of the VTOL MAV lift and drag coefficients. The question now is whether the same method can be used to accurately estimate the lateral-directional static stability derivatives of the vehicle.

Similar to I-CBM for the aerodynamic performance predictions, there are three computational tests set up for this study. The first set is for the full configuration, the second set is for the duct-wing (coupled), and the last set consists of five tests for the five main components of the VTOL MAV. For these various configurations, two static stability derivatives, C_{l_β} and C_{n_β} , are analyzed. Similar to the previous works on the method, the assumption taken for this study is that the rolling and yawing moments of the VTOL MAV are calculated by adding the duct-wing (coupled) results and the corrected-CBM results of the rest of the components.

The interference effects between the duct-wing and the downstream components are the main interest in the full configuration analysis. The duct has an aligning effect to the vehicle's x-body axis for the flow that passes through it. Therefore, the local flow leaving the duct and entering the stator is straightened in the direction of the x-body axis, despite of the sideslip angle of the freestream flow and thus creates a loss of sensitivity of the downstream control surfaces (Figure 5-9). The flow passes the stator and the empennage is subjected to a lower local sideslip angle compared to the freestream one. Hence, the rolling and yawing moments produced by the stator and the tails are lower than the values provided by the individual simulations.

Figure 5-9 Freestream and local flow around the VTOL MAV at the sideslip angle

The analysis is divided into four sections. The first section is for the coupled duct and wing. The second section is for the stator, while the last two sections are for the horizontal and vertical tails. There are four important parameters in this study: β , γ , δ and ϵ . The sideslip angle (β) of the VTOL MAV is the relative angle between the x-body axis and the freestream velocity vector. The second parameter (γ) is the setting angle of the stators, with γ_L is for the left plane and γ_R is for the right. The setting angle of the horizontal tail is symbolized as δ . The last parameter, ϵ , is the setting angle of the vertical tail.

5.4.1 Validation

There are 19 computational models built for the validation process, which consists of 10 individual and 9 full configuration models (Table 5-5). These models are tested with the propeller off for four sideslip angles: -10° , -5° , 5° and 10° (Appendix F). The data for the sideslip angle of 0° are taken from the I-CBM results for the angle-of-attack of 0° obtained during the development of I-CBM for aerodynamic performance predictions. A total of 76 simulations are performed in FLUENT to obtain the static stability derivatives in the lateral-directional mode.

In this study, the reference data are the full configuration simulation results. In the full configuration simulations, the VTOL MAV is tested with all its components integrated in the model. The full configuration model includes the interference effects between the components and thus expected to better represent the actual flow around the MAV than other types of simulation. Otherwise mentioned, the full configuration model used as the reference consists of the duct-wing coupled, the stator set at $0^\circ - 0^\circ$, the horizontal tail set at 0° and the vertical tail set at 0° .

Similar to the lift and drag predictions, the individual and full configuration simulation results of a component are compared to build the correction model. The corrected equations are taken from the linear approximation of the full configuration simulation results and marked by asterisk (*). In this work, the corrected equations are for the sideslip angle range from -10° to 10° .

Configuration	Setting angle
Duct	0°
Wing	0°
Duct-Wing (coupled)	0°
Stators ($\gamma_L - \gamma_R$)	
Symmetrical	$0^\circ - 0^\circ$
Asymmetrical-equal	$-10^\circ - 10^\circ$
Asymmetrical-unequal	$0^\circ - 5^\circ$
	$10^\circ - 0^\circ$
Horizontal tail (δ)	-2°
	0°
Vertical tail (ϵ)	-2°
	0°

Table 5-5 various configurations of computational model

5.4.2 Interference effects between the duct and the wing

In comparison with the full-configuration and coupled simulations, the individual simulations fail to predict the rolling moment and C_{l_β} accurately. This result implies that the duct-wing coupled model must be used to accurately predict the lateral-static-stability derivatives of the vehicle. The yawing moment curve for the duct-wing coupled configuration exhibits nonlinearity with respect to the sideslip angles. The C_n vs. β curve shows that both individual and coupled simulations over-predict the yawing moment at negative sideslip angles. Moreover, both simulations also under-predict the yawing moment at positive sideslip angles (Figure 5-10).

Figure 5-10 Rolling and yawing moments of duct-wing coupled

To further study the coupled duct-wing interference effects in the lateral-directional mode, the simulation results for the duct and the wing are plotted separately (Figure 5-11 and 5-12). Important findings that can be taken from this process are as follows:

- The rolling moment of the coupled configuration is produced mainly by the wing, whereas the yawing moment is strongly influenced by the duct.
- The nonlinearity of the yawing moment is introduced by the duct.
- The coupled simulation must be used to accurately predict the rolling moment of the duct and the wing, also the yawing moment of the wing.
- The full configuration simulation must be used to accurately predict the yawing moment of the duct. Further analysis on the interference effects between the duct and other MAV's components will be given next.

Figure 5-11 Rolling moment vs. Sideslip angle for duct and wing

Figure 5-12 Yawing moment vs. Sideslip angle for duct and wing

5.4.3 Interference effects between the coupled duct-wing and the stator

A shifting pattern in the C_l vs. β curve can be observed with the change in the stator's configuration. Although the pattern appears consistent, the shift is in the order of 10^{-4} - 10^{-3} , which is in the tolerance range of the solver. Moreover, the configuration of the stators does not affect C_{l_β} of the coupled duct-wing. The lines for all stator configurations are overlapping in the C_n vs. β curve. These findings suggest that the stators' influence on the rolling and yawing moments of the coupled duct-wing is very small (Figure 5-13).

Figure 5-13 Interference effect between duct-wing coupled and stator

5.4.4 Interference effects between the coupled duct-wing and the horizontal tail

Further analysis is performed to investigate the interactions between the coupled duct-wing and the horizontal tail. The full configuration models used in this study includes the asymmetrical-equal stator ($-10^\circ - 10^\circ$), the vertical tail set at 0° , whereas the horizontal tail are set at -2° and 0° .

The large distance between the horizontal tail and the coupled duct-wing creates an initial prediction that the interference effects between the two components are minimal. The C_l vs. β and C_n vs. β curves of the coupled duct-wing for two horizontal tail configurations, confirm the initial prediction. The yawing and rolling moment curves for the two configurations tested are on top of each another (Figure 5-14).

Figure 5-14 Interference effects between duct-wing coupled and horizontal tail

5.4.5 Interference effects between the coupled duct-wing and the vertical tail

Numerical analysis on the influence of the vertical tail to the lateral-directional static stability of the coupled duct-wing is presented here. The full configuration models analyzed in this study consists of the coupled duct-wing, the asymmetrical-equal stator ($-10^\circ - 10^\circ$), the horizontal tail

set at 0° , and the vertical tail set at -2° and 0° .

The results show that the large distance between the vertical tail and the coupled duct-wing has nearly eliminated the interactions between the two components. The rolling and yawing moments produced by the coupled duct-wing remain nearly constant with the change in the vertical tail's setting angle (Figure 5-15).

Figure 5-15 Interference effects between duct-wing coupled and vertical tail

5.4.6 Summary on the study of the interference effects

The study on the interference effects between the duct and the wing reveal that the yawing moment of the coupled duct-wing can only be predicted accurately using the full configuration analysis. This finding seems to invalidate the assumption used in the I-CBM that the aerodynamic and stability properties of the VTOL MAV are calculated by adding the duct-wing (coupled) results and the corrected-CBM results of the rest of the components. However, further study on the interference effects among the MAV components show that the I-CBM is still capable of providing fast and reasonably accurate prediction on the aerodynamic and stability properties of the VTOL MAV. In order to use the I-CBM, an adjustment need to be made on the yawing moment prediction of the duct-wing coupled configuration.

A change in the configuration of the duct will require rebuilding, meshing and solving the full computational model to predict the yawing moment produced by the coupled duct-wing (Section 5.4.2). Nevertheless, this situation is unlikely and changes are usually made to the downstream components such as the stator and the empennage. Moreover, a change in the configuration of the stator, horizontal or vertical tails only have a small influence on the aerodynamic and stability properties of the coupled duct-wing (Section 5.4.3 – 5.4.5). It means that only one full-configuration analysis needs to be performed regardless the change in any of the downstream components. Unless the duct is modified, the full configuration analysis is only performed once to obtain the aerodynamic moments of the coupled duct-wing. After that, the I-CBM can be used in a similar way as the lift and drag predictions (Section 5.3).

5.4.7 The correction model for the aerodynamic moment coefficients of the stator

The slope of the C_l vs. β and C_n vs. β curves of the full configuration simulations are lower than the individual ones (Figure 5-16). These small values of C_{n_β} and C_{l_β} show that the freestream sideslip angle has a very small influence on the aerodynamic forces and moments of the stator. It also confirms our prediction that the duct redirects the flow. The flow leaving the duct is aligned in the direction of the x-body axis, creates much lower local sideslip-angle, which leads to a loss of sensitivity to sideslip-angles of the downstream control surfaces. For other stator configurations, similar findings of a very gentle slope of the C_l vs. β and C_n vs. β curves are also observed (Figure 5-17).

Figure 5-16 Rolling and yawing moments of asymmetrical-unequal stator

Figure 5-17 Rolling and yawing moments of symmetrical and asymmetrical-equal stators

5.4.8 The correction model for the aerodynamic moment coefficients of the horizontal tail

The full configuration and individual simulation results for the horizontal tail set at 0° and -2° are analyzed in this section. In the full vehicle configuration, the horizontal tail contributes more to the rolling moment ($O [10^{-3}]$) than to the yawing moment ($O [10^{-4}]$) (Figure 5-18). By observing

the slope of the C_l vs. β and C_n vs. β curves, it is found that the influence of sideslip angles is more pronounced for the yawing moment than it is for the rolling. However, it is also found that this strong influence on yawing moment deteriorates significantly when the horizontal tail is integrated with the other components in the full vehicle configuration. The horizontal tail in the full configuration simulations is influenced more by the duct exhaust flow than the freestream flow. The local sideslip-angles are very small compared to the freestream ones due to the aligning effect of the duct.

Figure 5-18 Rolling and yawing moments of horizontal tail

5.4.9 The correction model for the aerodynamic moment coefficients of the vertical tail

The main function of the vertical tail is to stabilize and to control the VTOL MAV in the directional mode. As expected, the vertical tail generates a very small rolling moment ($O [10^{-4}]$) and contributes significantly to the yawing moment ($O [10^{-2}]$) (Figure 5-19). It is found that the sideslip angle only influences the rolling moment slightly, although for the yawing moment it has a strong effect. The much lower yawing moment produced by the vertical tail in the full configuration simulations shows how the tail loses its sensitivity to sideslip angles. The flow

passes the vertical tail is subjected to a lower local sideslip-angle compared to the freestream one due to the aligning effect of the duct.

Figure 5-19 Rolling and yawing moments of vertical tail

The yawing moment created by the vertical tail of -2° setting angle is higher than the one set at 0° in both types of simulations. The setting angle of the vertical tail results in a higher or lower local sideslip-angle. For example at the sideslip angle of 10°, the local sideslip-angle for the vertical tail set at 0° is also 10°. However, for the vertical tail set at -2°, the local angle is 12° because the negative setting angle has an added effect to the freestream sideslip angle (Figure 5-20). A higher local sideslip-angle creates a higher side force (F_y), which leads to a higher yawing moment.

Figure 5-20 Flow scheme around vertical tail (top view) at the sideslip angle of 10°

5.4.10 Conclusions

There are three different approaches being compared in the predictions of the lift and drag coefficients (Section 5.3.5): CBM, I-CBM and the full configuration analysis. As explained previously in Section 5.4.6, an adjustment is needed for a more accurate prediction of the rolling and yawing moments of the VTOL MAV. Hence, the adjusted/calibrated I-CBM, which is marked by hash symbol (#), is included in Table 5-6. The difference between I-CBM and I-CBM[#] is the source of the coupled duct-wing data. In the I-CBM and I-CBM[#], the coupled duct-wing data are taken from the coupled and the full-configuration simulation results respectively. The corrected values for the sideslip angle of 5°, highlighted in Table 5-6, are calculated using the correction model for the stator and the empennage.

	CBM		I-CBM		I-CBM [#]		Full configuration	
	C _l	C _n	C _l	C _n	C _l	C _n	C _l	C _n
Wing	-0.0001	0.0003	-0.0007	-0.0002	-0.0017	0.0015		
Duct	-0.0002	0.0002						
Horizontal tail	-0.00005	0.0005	0.0010	0.00008	0.0010	0.00008		
Vertical tail	-0.00002	0.0119	0.0010	0.0023	0.0010	0.0023		
Stator	-0.0075	0.0015	-0.0059	-0.00002	-0.0059	-0.00002		
Total	-0.0078	0.0144	-0.0046	0.0022	-0.0055	0.0039	-0.0051	0.0036
Error (%)	52.82	303.28	11.11	38.40	7.71	7.87		

	Coupled duct-wing results from the coupled analysis
	Coupled duct-wing results from the full-configuration analysis
	Corrected-CBM results

Table 5-6 Comparison of results for freestream sideslip angle of 5°

Using the results from the full configuration analysis as a reference, it is found that the I-CBM gives significantly more accurate results compared to CBM. The CBM over predicts the total rolling moment and the total yawing moment of the VTOL MAV by more than 52% and 300% respectively. Using the I-CBM, the total rolling moment is under estimated with less than 12%, while the total yawing moment is around 39% lower than the reference value. The calibrated I-CBM (I-CBM[#]) overestimates both the total rolling moment and the total yawing moment by less

than 8%. Depending on the accuracy level desired, both the I-CBM and I-CBM[#] can be used in the conceptual design phase of a winged-ducted propeller MAV.

5.5 Summary

The proposed method, Improved-Component Buildup Method (I-CBM), has been developed and tested for fast and reasonably accurate prediction of the aerodynamic and stability properties of a winged-ducted propeller MAV. This method is developed from the Component Buildup Method (CBM), in which the aerodynamic coefficients of the vehicle are calculated by adding the aerodynamic coefficient of its components. In I-CBM, aerodynamic properties of the MAV are calculated by adding the duct-wing (coupled) results and the corrected-CBM results of the rest of the components. Compared to the original CBM method that is found to be inaccurate due to the negligence of the aerodynamic interferences between the MAV's components, I-CBM is capable of providing more accurate results. Moreover, this method is less complex and less costly computationally compared to the full vehicle simulations.

Analysis	Run Time (hours)
Full Configuration	14
Coupled Duct-Wing	4
Individual	2

Table 5-7 Run time for various computational models

Simulations conducted using workstations with 2.40 GHz and 2.00 GB of RAM show clearly the decrease in the computational costs (Table 5-7). The full configuration analysis requires 14 hours of simulation to account for a change in any of the component configuration. For the same purpose, I-CBM requires only 2 hours of simulation, which is 85% less than the computational cost required by the full configuration analysis. In I-CBM, 8 hours is the maximum run time needed to calculate the lift and the drag of the full MAV if there is a change in every component. This run time of 8 hours is made of 4 hours for the duct-wing coupled and 2 hours each for the stator and the horizontal tail.

For the prediction of the rolling and yawing moments using I-CBM, the maximum run time is 10 hours. Compared to the lift and drag predictions, there is an additional run time of 2 hours to account for a change in the vertical tail configuration. Using the I-CBM[#], the maximum run time, excluding the one-time full configuration analysis for the coupled duct-wing, is 6 hours with 2 hours each for the stator, horizontal and vertical tails.

Chapter 6 Conclusions and Future Works

6.1 Conclusions

Numerical method has been successfully used to calculate the aerodynamic forces and moments coefficients of a winged-ducted propeller VTOL MAV. Conclusions that can be drawn from this work are as follows:

On the computational approach:

- Validation of the code shows that the code and the simulation procedure used in this research can provide reasonably accurate predictions of aerodynamic forces and moments.
- Modeling of the ducted-propelled system by assuming the propeller as an actuator disc is proven to be sufficiently accurate and that the thrust produce in the computational model is the same as the experimental data obtained from the static thrust test.

On the aerodynamic and stability characteristics of the VTOL MAV:

- The computational results show that the current design of VTOL MAV has favorable aerodynamic performances as well as longitudinal and directional static stability quality. However, it tends to be statically unstable in the lateral mode. The constant-thrust propeller mounted inside the duct increases the lift and drag, and destabilizes the vehicle in the lateral mode.
- The numerical analysis of the lifting/control surfaces suggest that changing the size and/or geometrical properties of the wing, stator, vertical or horizontal tails influences the aerodynamic and static stability characteristics of the MAV. The numerical results show that increasing the wing setting-angle can improve the vehicle's performance only at low angles-of-attack. The results also suggest that deflecting the empennage has a small influence on the vehicle's performances and static stability quality.
- The operating condition analysis indicates that velocity has a small influence on the lift, while the drag increases significantly with the decrease in velocity. The perfect hover simulations show that the lift, drag and pitching moments of the vehicle behave nonlinearly with velocity variations. The sidewind simulations at hover-flight indicate that the MAV is statically stable in the longitudinal mode under the influence of sidewind/crosswind.

- The dynamic stability analysis shows that the MAV have favorable dynamic stability quality in the lateral, longitudinal, and directional modes. However, it needs to be acknowledged that the small values of the aerodynamic moments used in deducting the dynamic stability derivatives are very difficult to predict using the current code.

On the Improved-Component Buildup Method:

- The development and validation of the Improved-Component Buildup Method (I-CBM) show that the proposed method is capable of providing more accurate predictions of the aerodynamic and stability properties of the VTOL MAV, compared to the original CBM. I-CBM is also less complex and less costly computationally than the full MAV analysis.
- The development of I-CBM provides insight about the interference effects between the duct and the wing, between the duct and the stator, between the duct and horizontal tail, also between the duct and vertical tail. The duct has a significant influence on the aerodynamic and control characteristics of the downstream lifting/control surfaces.

6.2 Future works

The present study has revealed some topics that will need to be investigated further. These topics consist of experimental and computational studies as follows:

On aerodynamic and stability analysis:

- The interference effects between the MAV's components are an important topic to be investigated in the future. The MAV's components are usually low-aspect-ratio lifting/control surfaces that might exhibit complex flow phenomena when they are placed in the vicinity of the others. Focus should be given to study the lifting/control surfaces because their effectiveness is critical to the design of the MAV and its control system.
- The effect of propeller slipstream to the downstream lifting/control surfaces is a topic worth of detailed study. The rotating propeller increases the downstream velocity of the flow before it passes the stator, the horizontal and vertical tails.
- The MAV lateral static stability is recommended for further analysis. The results presented in this works indicate that the vehicle tends to be statically unstable in the lateral mode. An accurate study on this topic is currently hindered by the difficulty in predicting the flow at high sideslip-angles.

- The dynamic stability of the MAV needs to be investigated further. The preliminary works indicate the need for a more sophisticated code. The new code should be more robust and more accurate in predicting the small values of aerodynamic forces and moments.

On experimental approach:

- Validation of the VTOL MAV computation results
A complete prototype of the VTOL MAV is in the manufacturing process. Once the prototype is completed, wind or water-tunnel testing, for both the full configuration and the individual component, can be performed to investigate the aerodynamic and stability properties of the VTOL MAV. The experimental results can be used to validate the numerical aerodynamic and stability analysis as well as the Improved-Component Buildup Method (I-CBM).

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Appendix A Computational Results of the Aerodynamic and Static Stability Analysis of the VTOL MAV-baseline

A. 1 Baseline tests results

a) Effect of Convergence Limit

The VTOL MAV-baseline was tested for two residual criteria (Table A-1). As typically found in unstructured grid models, the computation results of the VTOL MAV-baseline were improved appreciably when the residual criterion was decreased from 10^{-3} to 10^{-6} .

Limit	C_L	C_D	C_l	C_m	C_n
10^{-3}	0.3042	1.3859	0.0005	0.2699	-0.0014
10^{-6}	0.3031	1.3801	0.0002	0.2701	-0.0012

Table A-1 Computation results for different convergence criterion

b) Grid independence study

Three models with grid sizes of around 800,000, 1 and 1.2 million cells were tested (Table A-2, A-3, and A-4). The computation results change slightly when the grid size is increased from 1 to 1.2 million. This trend shows that grid independence is already achieved. Considering the results accuracy, the model with the most cell numbers (1.2 million) is used for the VTOL MAV-baseline computations.

Angle-of-attack (deg)	C_D	C_L	C_l	C_m	C_n
20	0.3680	0.8827	-0.0056	-0.0354	0.0025
40	0.9806	1.1083	-0.0063	-0.1095	0.0067
60	1.3307	0.7478	0.0006	-0.1362	0.0042

Table A-2 Computation results for the VTOL MAV-baseline model with 800,000 cells

Angle-of-attack (deg)	C_D	C_L	C_l	C_m	C_n
20	0.3639	0.8864	-0.0067	-0.0351	0.0025
40	0.9096	1.0226	-0.0046	-0.0780	0.0033
60	1.2933	0.7276	0.0007	-0.1367	0.0045

Table A-3 Computation results for the VTOL MAV-baseline model with 1 million cells

Angle-of-attack (deg)	C_D	C_L	C_l	C_m	C_n
20	0.3628	0.8992	-0.0059	-0.0349	0.0020
40	0.9050	1.0219	-0.0001	-0.0777	0.0002
60	1.2956	0.7346	0.0008	-0.1361	0.0044

Table A-4 Computation results for the VTOL MAV-baseline model with 1.2 million cells

c) Near wall grid distance (wall y^+)

The numerical results for the adapted and non-adapted grids of the VTOL MAV-baseline are presented in Table A-5.

Grid	C_D	C_L
adapted	0.368	0.883
non-adapted	0.380	0.911

Table A-5 Lift and drag of adapted and non-adapted models

A. 2 Computational results

The VTOL MAV-baseline was tested at angles-of-attack 0° - 90° to study its aerodynamic and flight stability characteristics (Table A-6). It was also tested with the propeller operating (Table A-7). The actuator disk which represents the actual propeller is set at pressure jump equal of 34.8 Pa which corresponds to static thrust of 1.84 N.

Angle-of-attack (deg)	C_L	C_D	C_l	C_m	C_n
0	0.0873	0.0156	-0.0045	-0.0021	-4.8E-05
10	0.1404	0.5308	-0.0044	-0.0126	0.0009
20	0.3628	0.8992	-0.0059	-0.0527	0.0020
30	0.6477	0.9598	-0.0007	-0.0919	0.0034
40	0.9050	1.0219	-5.8E-05	-0.1411	0.0004
60	1.2956	0.7346	0.0008	-0.1991	0.0050
80	1.3859	0.3042	-0.0005	-0.2699	0.0014
90	1.4085	0.0912	0.0018	-0.2943	0.0001

Table A-6 Computation results of the VTOL MAV-baseline (without propeller)

Angle-of-attack (deg)	C_L	C_D	C_l	C_m	C_n
0	0.1235	0.0152	-0.0061	-0.0020	-9.6E-05
20	0.4084	0.9692	-0.0079	-0.0512	0.0025
60	1.3927	0.7918	0.0016	-0.1982	0.0043
90	1.4082	0.0867	-0.0008	-0.2870	-0.0003

Table A-7 Computation results of the VTOL MAV-baseline (with propeller)

Appendix B Computational Results of the Aerodynamic and Static Stability Analysis of the VTOL MAV

The numerical results for various angles-of-attack are presented in Table B-1 and B-2, while the results for various sideslip angles are displayed in Table B-3 and B-4.

Angle-of-attack (deg)	C_L	C_D	C_l	C_m	C_n
0	0.0728	0.0967	-0.0044	-0.0003	0.0019
10	0.6247	0.1728	-0.0045	-0.0124	0.0008
20	0.9994	0.4187	-0.0081	-0.0528	0.0002
30	1.0806	0.6883	-0.0018	-0.0853	0.0010
40	1.0718	0.9216	0.0002	-0.1103	0.0002
50	1.0027	1.1743	-0.0006	-0.1639	0.0008
60	0.7718	1.2843	-0.0032	-0.1889	0.0026
70	0.4910	1.3630	0.0032	-0.2254	0.0029
80	0.3047	1.4208	0.0009	-0.2655	0.0011
90	0.0857	1.4333	0.0021	-0.2907	-0.0009

Table B-1 Computation results for various angles-of-attack (without propeller)

Angle-of-attack (deg)	C_L	C_D	C_l	C_m	C_n
0	0.0703	0.1096	-0.0059	0.0004	0.0008
10	0.6543	0.1967	-0.0059	-0.0101	0.0010
20	1.0719	0.4475	-0.0099	-0.0522	0.0002
30	1.2202	0.7676	-0.0033	-0.1011	0.0022
40	1.1859	1.0157	-0.0011	-0.1262	0.0039
50	1.0893	1.2819	-0.0006	-0.1744	0.0003
60	0.8393	1.4213	-0.0024	-0.1990	0.0018
70	0.5538	1.4579	0.0008	-0.2164	0.0004
80	0.3529	1.4844	-0.0019	-0.2500	0.0004
90	0.1277	1.4471	-0.0003	-0.2796	-0.0003

Table B-2 Computation results for various angles-of-attack (with propeller)

Sideslip angle (deg)	C_L	C_D	C_l	C_m	C_n
-40	-0.1173	0.5320	0.0217	0.0149	-0.0664
-30	-0.0666	0.3862	0.0174	0.0161	-0.0449
-20	0.0620	0.2503	-0.0030	0.0003	-0.0346
-10	0.0765	0.1348	-0.0052	-0.0005	-0.0078
0	0.0728	0.0967	-0.0044	-0.0003	0.0019
10	0.0691	0.1358	-0.0032	-0.0003	0.0125
20	0.0493	0.2522	-0.0052	0.0077	0.0387
30	-0.0925	0.3906	-0.0246	0.0188	0.0462
40	-0.1360	0.5327	-0.0320	0.0195	0.0655

Table B-3 Computation results for various sideslip angles (without propeller)

Sideslip angle (deg)	C_L	C_D	C_l	C_m	C_n
-20	0.0878	0.2870	-0.0083	-0.0076	-0.0360
-10	0.0787	0.1595	-0.0065	-0.0007	-0.0072
0	0.0703	0.1096	-0.0059	0.0004	0.0008
10	0.0630	0.1610	-0.0044	0.0015	0.0131
20	0.0578	0.2898	-0.0031	0.0061	0.0419

Table B-4 Computation results for various sideslip angles (with propeller)

Appendix C Computational Results of the Aerodynamic and Static Stability Analysis of the Lifting/Control Surfaces

C. 1 Wing setting angle

Angle-of-attack (deg)	CD	CL	CI	Cm	Cn
0	9.67E-02	7.28E-02	-4.42E-03	-3.04E-04	1.95E-03
10	1.73E-01	6.25E-01	-4.55E-03	-1.24E-02	8.26E-04
20	4.19E-01	9.99E-01	-8.07E-03	-5.28E-02	2.21E-04
30	6.88E-01	1.08E+00	-1.84E-03	-8.53E-02	9.83E-04
40	9.22E-01	1.07E+00	1.60E-04	-1.10E-01	2.06E-04
50	1.17E+00	1.00E+00	-5.71E-04	-1.64E-01	7.88E-04
60	1.28E+00	7.72E-01	-3.21E-03	-1.89E-01	2.62E-03
70	1.36E+00	4.91E-01	3.16E-03	-2.25E-01	2.89E-03
80	1.42E+00	3.05E-01	8.91E-04	-2.66E-01	1.07E-03
90	1.43E+00	8.57E-02	2.11E-03	-2.91E-01	-9.32E-04

Table C-1 Computational results of the VTOL MAV with wing set at 6°

Angle-of-attack (deg)	CD	CL	CI	Cm	Cn
0	9.99E-02	1.16E-01	-3.93E-03	-9.17E-04	1.79E-03
10	1.90E-01	6.70E-01	-2.16E-03	-1.37E-02	6.32E-04
20	4.47E-01	1.02E+00	-3.45E-03	-5.48E-02	2.41E-04
40	9.43E-01	1.09E+00	4.49E-03	-1.10E-01	8.28E-04
90	1.42E+00	6.78E-02	2.57E-03	-2.89E-01	-5.91E-03

Table C-2 Computational results of the VTOL MAV with wing set at 10°

C. 2 Deflection of the horizontal tail

Angle-of-attack (deg)	CD	CL	CI	Cm	Cn
0	9.67E-02	7.28E-02	-4.42E-03	-3.04E-04	1.95E-03
10	1.73E-01	6.25E-01	-4.55E-03	-1.24E-02	8.26E-04
20	4.19E-01	9.99E-01	-8.07E-03	-5.28E-02	2.21E-04
30	6.88E-01	1.08E+00	-1.84E-03	-8.53E-02	9.83E-04
40	9.22E-01	1.07E+00	1.60E-04	-1.10E-01	2.06E-04
50	1.17E+00	1.00E+00	-5.71E-04	-1.64E-01	7.88E-04
60	1.28E+00	7.72E-01	-3.21E-03	-1.89E-01	2.62E-03
70	1.36E+00	4.91E-01	3.16E-03	-2.25E-01	2.89E-03
80	1.42E+00	3.05E-01	8.91E-04	-2.66E-01	1.07E-03
90	1.43E+00	8.57E-02	2.11E-03	-2.91E-01	-9.32E-04

Table C-3 Computational results of the VTOL MAV with horizontal tail set at -2°

Angle-of-attack (deg)	CD	CL	CI	Cm	Cn
0	9.49E-02	7.90E-02	-4.22E-03	-3.59E-03	1.44E-03
10	1.74E-01	6.32E-01	-2.54E-03	-1.60E-02	2.85E-04
20	4.21E-01	1.01E+00	-3.17E-03	-5.58E-02	-1.53E-04
30	6.93E-01	1.09E+00	1.92E-03	-8.72E-02	1.23E-04
40	9.26E-01	1.08E+00	4.36E-03	-1.13E-01	-5.46E-04

Table C-4 Computational results of the VTOL MAV with horizontal tail set at 0°

Angle-of-attack (deg)	CD	CL	CI	Cm	Cn
0	9.50E-02	8.43E-02	-4.13E-03	-6.68E-03	1.66E-03
10	1.75E-01	6.37E-01	-2.33E-03	-1.91E-02	4.17E-04
20	4.21E-01	1.01E+00	-4.23E-03	-5.88E-02	-9.32E-05
30	6.94E-01	1.09E+00	2.22E-03	-8.90E-02	2.65E-04
40	9.32E-01	1.08E+00	3.99E-03	-1.15E-01	1.20E-03

Table C-5 Computational results of the VTOL MAV with horizontal tail set at 2°

C. 3 Deflection of the vertical tail

Angle-of-attack (deg)	CD	CL	CI	Cn	Cm
0	9.67E-02	7.28E-02	-4.42E-03	1.95E-03	-3.04E-04
10	1.73E-01	6.25E-01	-4.55E-03	8.26E-04	-1.24E-02
20	4.19E-01	9.99E-01	-8.07E-03	2.21E-04	-5.28E-02
30	6.88E-01	1.08E+00	-1.84E-03	9.83E-04	-8.53E-02
40	9.22E-01	1.07E+00	1.60E-04	2.06E-04	-1.10E-01
50	1.17E+00	1.00E+00	-5.71E-04	7.88E-04	-1.64E-01
60	1.28E+00	7.72E-01	-3.21E-03	2.62E-03	-1.89E-01
70	1.36E+00	4.91E-01	3.16E-03	2.89E-03	-2.25E-01
80	1.42E+00	3.05E-01	8.91E-04	1.07E-03	-2.66E-01
90	1.43E+00	8.57E-02	2.11E-03	-9.32E-04	-2.91E-01

Table C-6 Computational results of the VTOL MAV with vertical tail set at -1°

Angle-of-attack (deg)	CD	CL	CI	Cm	Cn
0	9.48E-02	3.29E-02	-3.81E-03	6.42E-04	-7.74E-04
10	1.73E-01	6.24E-01	-2.40E-03	-1.22E-02	-1.73E-03
20	4.19E-01	9.98E-01	-3.94E-03	-5.30E-02	-1.86E-03

Table C-7 Computational results of the VTOL MAV with vertical tail set at 0°

Angle-of-attack (deg)	CD	CL	CI	Cm	Cn
0	9.50E-02	7.30E-02	-4.18E-03	-3.77E-04	-2.32E-03
10	1.72E-01	6.24E-01	-2.36E-03	-1.22E-02	-3.14E-03
20	4.20E-01	1.00E+00	-3.43E-03	-5.29E-02	-3.28E-03

Table C-8 Computational results of the VTOL MAV with vertical tail set at 1°

C. 4 Stator effect on the aerodynamic and stability properties of the VTOL MAV

Angle-of-attack (deg)	CD	CL	CI	Cm	Cn
0	0.0967	0.0728	-0.0044	-0.0003	0.0019
10	0.1728	0.6247	-0.0045	-0.0124	0.0008
20	0.4187	0.9994	-0.0081	-0.0528	0.0002
30	0.6883	1.0806	-0.0018	-0.0853	0.0010
40	0.9216	1.0718	0.0002	-0.1103	0.0002
50	1.1743	1.0027	-0.0006	-0.1639	0.0008
60	1.2843	0.7718	-0.0032	-0.1889	0.0026
70	1.3630	0.4910	0.0032	-0.2254	0.0029
80	1.4208	0.3047	0.0009	-0.2655	0.0011
90	1.4333	0.0857	0.0021	-0.2907	-0.0009

Table C-9 Computational results of the VTOL MAV with stator set at -10° - 10° for various angles-of-attack

Angle-of-attack (deg)	CD	CL	CI	Cm	Cn
0	0.0694	0.0680	0.0002	0.0014	0.0017
10	0.1482	0.6119	0.0021	-0.0109	0.0018
20	0.3839	0.9392	0.0009	-0.0537	0.0015
90	1.2432	0.0622	0.0005	-0.2251	-0.0043

Table C-10 Computational results of the VTOL MAV without stators for various angles-of-attack

Sideslip-angle (deg)	CI	Cm	Cn
-40	2.21E-02	1.50E-02	-6.67E-02
-30	1.79E-02	1.62E-02	-4.52E-02
-20	-2.21E-03	4.25E-04	-3.48E-02
-10	-4.66E-03	-4.12E-04	-8.02E-03
0	-4.42E-03	-3.04E-04	1.95E-03
10	-3.24E-03	-1.93E-04	1.22E-02
20	-5.57E-03	7.75E-03	3.85E-02
30	-2.56E-02	1.90E-02	4.59E-02
40	-3.33E-02	1.96E-02	6.51E-02

Table C-11 Computational results of the VTOL MAV with stator set at -10° - 10° for various sideslip-angles

Sideslip-angle (deg)	Cl	Cm	Cn
-30	1.88E-02	1.52E-02	-4.68E-02
-20	-1.93E-04	2.01E-03	-3.54E-02
-10	-1.47E-04	6.66E-04	-8.89E-03
0	1.65E-04	1.40E-03	1.72E-03
10	5.41E-04	9.44E-04	1.30E-02
20	3.26E-04	2.67E-03	3.97E-02
30	-1.93E-02	1.50E-02	4.82E-02

Table C-12 Computational results of the VTOL MAV without stators for various sideslip-angles

C. 5 Effect of stator blade-cascade configuration

Angle-of-attack (deg)	CD	CL	Cl	Cm	Cn
0	1.22E-02	5.46E-04	-5.80E-03	-1.63E-04	-3.81E-05
10	1.73E-02	4.70E-02	-5.79E-03	-1.47E-02	1.20E-04
20	3.17E-02	7.23E-02	-3.19E-03	-2.45E-02	1.17E-03

Table C-13 Computational results of the single-layered-stator set at $-10^\circ - 10^\circ$ for various angles-of-attack

Angle-of-attack (deg)	CD	CL	Cl	Cm	Cn
0	2.78E-02	-5.40E-05	-6.67E-03	4.98E-05	-9.06E-05
10	3.57E-02	5.03E-02	-5.52E-03	-1.51E-02	7.14E-04
20	5.31E-02	9.11E-02	-3.52E-03	-2.89E-02	1.77E-03

Table C-14 Computational results of the double-layered-stator set at $-10^\circ - 10^\circ$ for various angles-of-attack

Angle-of-attack (deg)	CD	CL	Cl	Cm	Cn
0	4.49E-02	5.57E-05	-7.42E-03	3.14E-05	-1.22E-04
10	5.42E-02	5.85E-02	-6.00E-03	-1.69E-02	8.12E-04
20	7.45E-02	1.09E-01	-3.93E-03	-3.32E-02	2.14E-03

Table C-15 Computational results of the triple-layered-stator set at $-10^\circ - 10^\circ$ for various angles-of-attack

Sideslip-angle (deg)	CD	CL	Cl	Cm	Cn
-20	1.29E-02	-8.05E-03	-6.42E-03	1.43E-03	-1.74E-03
-10	1.25E-02	-4.20E-03	-6.09E-03	7.11E-04	-9.35E-04
0	1.22E-02	5.46E-04	-5.80E-03	-1.63E-04	-3.81E-05
10	1.26E-02	4.16E-03	-6.07E-03	-9.15E-04	8.32E-04
20	1.29E-02	8.01E-03	-6.38E-03	-1.51E-03	1.63E-03

Table C-16 Computational results of the single-layered-stator set at $-10^\circ - 10^\circ$ for various sideslip-angles

Sideslip-angle (deg)	CD	CL	CI	Cm	Cn
-10	2.81E-02	-9.33E-03	-6.91E-03	1.13E-03	-2.09E-03
-5	2.80E-02	-4.69E-03	-6.77E-03	7.08E-04	-1.10E-03
0	2.78E-02	-5.40E-05	-6.67E-03	4.98E-05	-9.06E-05
5	2.79E-02	4.80E-03	-6.71E-03	-4.92E-04	9.38E-04
10	2.80E-02	9.33E-03	-6.83E-03	-8.73E-04	1.90E-03

Table C-17 Computational results of the double-layered-stator set at -10° – 10° for various sideslip-angles

Sideslip-angle (deg)	CD	CL	CI	Cm	Cn
-10	4.53E-02	-1.49E-02	-7.74E-03	1.04E-03	-3.31E-03
-5	4.51E-02	-7.77E-03	-7.53E-03	3.28E-04	-1.82E-03
0	4.49E-02	5.57E-05	-7.42E-03	3.14E-05	-1.22E-04
5	4.49E-02	7.80E-03	-7.50E-03	-7.15E-04	1.52E-03
10	4.52E-02	1.50E-02	-7.71E-03	-1.45E-03	3.04E-03

Table C-18 Computational results of the triple-layered-stator set at -10° – 10° for various sideslip-angles

Appendix D Computational Results of the Effect of the VTOL MAV Operating Conditions

D. 1 Velocity variations

Angle-of-attack (deg)	CD	CL	CI	Cm	Cn
0	1.60E-01	6.79E-02	-2.88E-03	-1.80E-03	1.57E-03
10	2.41E-01	6.02E-01	-1.85E-03	-1.42E-02	8.95E-04
20	4.58E-01	9.92E-01	-8.67E-04	-4.21E-02	4.13E-04
90	1.48E+00	6.99E-02	1.64E-03	-2.93E-01	-4.98E-03

Table D-1 Computational results of the VTOL MAV for freestream velocity of 1 m/s

Angle-of-attack (deg)	CD	CL	CI	Cm	Cn
0	1.07E-01	7.31E-02	-4.41E-03	-7.44E-04	1.88E-03
10	1.86E-01	6.21E-01	-4.52E-03	-1.19E-02	8.03E-04
20	4.29E-01	1.01E+00	-5.30E-03	-5.04E-02	-1.10E-05
30	7.01E-01	1.10E+00	-1.90E-03	-8.39E-02	9.99E-04
40	9.28E-01	1.08E+00	1.77E-05	-1.08E-01	5.05E-04
50	1.18E+00	1.01E+00	-7.07E-04	-1.63E-01	1.12E-03
60	1.30E+00	7.80E-01	-3.47E-03	-1.90E-01	1.44E-03
70	1.37E+00	4.90E-01	2.77E-03	-1.87E-01	2.80E-03
80	1.43E+00	3.04E-01	1.07E-03	-2.53E-01	1.35E-03

Table D-2 Computational results of the VTOL MAV for freestream velocity of 5 m/s

Angle-of-attack (deg)	CD	CL	CI	Cm	Cn
0	1.02E-01	7.16E-02	-4.34E-03	-4.26E-05	2.07E-03
10	1.77E-01	6.23E-01	-4.52E-03	-1.20E-02	8.46E-04
20	4.22E-01	1.01E+00	-6.34E-03	-5.14E-02	8.94E-05
30	6.93E-01	1.09E+00	-1.45E-03	-8.51E-02	1.10E-03
40	9.23E-01	1.07E+00	1.21E-04	-1.09E-01	1.89E-05
50	1.18E+00	1.00E+00	-6.22E-04	-1.64E-01	7.83E-04
60	1.29E+00	7.74E-01	-3.16E-03	-1.89E-01	2.16E-03
70	1.36E+00	4.90E-01	3.08E-03	-1.89E-01	2.60E-03
80	1.42E+00	3.04E-01	7.92E-04	-2.53E-01	1.37E-03

Table D-3 Computational results of the VTOL MAV for freestream velocity of 10 m/s

Angle-of-attack (deg)	CD	CL	CI	Cm	Cn
0	9.67E-02	7.28E-02	-4.42E-03	-3.04E-04	1.95E-03
10	1.73E-01	6.25E-01	-4.55E-03	-1.24E-02	8.26E-04
20	4.19E-01	9.99E-01	-8.07E-03	-5.28E-02	2.21E-04
30	6.88E-01	1.08E+00	-1.84E-03	-8.53E-02	9.83E-04
40	9.22E-01	1.07E+00	1.60E-04	-1.10E-01	2.06E-04
50	1.17E+00	1.00E+00	-5.71E-04	-1.64E-01	7.88E-04
60	1.28E+00	7.72E-01	-3.21E-03	-1.89E-01	2.62E-03
70	1.36E+00	4.91E-01	3.16E-03	-2.25E-01	2.89E-03
80	1.42E+00	3.05E-01	8.91E-04	-2.66E-01	1.07E-03
90	1.43E+00	8.57E-02	2.11E-03	-2.91E-01	-9.32E-04

Table D-4 Computational results of the VTOL MAV for freestream velocity of 15 m/s

D. 2 Velocity variations during hover

U (m/s)	CD	CL	CI	Cm	Cn
1	1.60E-01	6.79E-02	-2.87E-03	-1.90E-03	1.57E-03
2	1.28E-01	7.27E-02	-3.96E-03	-1.53E-03	1.19E-03
3	1.17E-01	7.37E-02	-4.10E-03	-1.13E-03	1.25E-03
5	1.07E-01	7.30E-02	-4.15E-03	-6.58E-04	1.48E-03
10	9.92E-02	7.26E-02	-4.16E-03	-3.39E-04	1.59E-03

Table D-5 Computational results of the VTOL MAV at perfect hover conditions for various freestream velocities

D. 3 Sidewind/crosswind effects

Angle-of-attack (deg)	CD	CL	CI	Cm	Cn
70	1.36E+00	4.91E-01	3.16E-03	-2.25E-01	2.89E-03
80	1.42E+00	3.05E-01	8.91E-04	-2.66E-01	1.07E-03
90	1.43E+00	8.57E-02	2.11E-03	-2.91E-01	-9.32E-04

Table D-6 Computational results of the VTOL MAV of front-side configuration

Angle-of-attack (deg)	CD	CL	CI	Cm	Cn
80	1.37E+00	3.41E-01	1.19E-03	-2.45E-01	-3.19E-03
90	1.40E+00	1.34E-01	1.32E-04	-2.87E-01	-5.19E-03
100	1.38E+00	-8.49E-02	4.00E-04	-2.93E-01	-6.47E-03

Table D-7 Computational results of the VTOL MAV of back-side configuration

Angle-of-attack (deg)	CD	CL	CI	Cm	Cn
80	5.22E-01	1.13E-01	-1.06E-03	-1.25E-01	-3.11E-03
90	5.01E-01	4.97E-02	-1.73E-04	-1.39E-01	-3.99E-03
100	5.04E-01	-3.61E-02	-9.70E-04	-1.51E-01	-4.87E-03

Table D-8 Computational results of the VTOL MAV of left-wing configuration

Angle-of-attack (deg)	CD	CL	CI	Cm	Cn
80	5.19E-01	1.17E-01	-9.98E-04	-1.30E-01	-4.20E-03
90	5.00E-01	5.37E-02	-1.67E-03	-1.39E-01	-3.09E-03
100	5.05E-01	-3.10E-02	-1.53E-03	-1.51E-01	-3.39E-03

Table D-9 Computational results of the VTOL MAV of right-wing configuration

Appendix E Computational Results of the Dynamic Stability Analysis of the VTOL MAV

E. 1 Baseline tests results

Time step size (s)	Cx	Cy	Cz	Cl	Cm	Cn
0.01	0.0951	0.0724	0.0035	-0.0041	-0.0002	0.0017
0.001	0.0952	0.0725	0.0035	-0.0041	-0.0002	0.0017
0.0001	0.0960	0.0731	0.0033	-0.0042	-0.0002	0.0017

Table E-1 Comparisons of numerical results for various time-step-sizes

Parameters	Steady	1st order	2nd order
Cx	0.0950	0.1182	0.0950
Cy	0.0718	0.0740	0.0717
Cz	0.0096	0.0103	0.0096
Cl	-0.0042	-0.0043	-0.0041
Cn	0.0017	0.0018	0.0017
Cm	-0.00001	0.0002	0.00002

Table E-2 Comparisons of numerical results for various discretization schemes

Parameters	Steady	SIMPLE	SIMPLEC
Cx	0.0950	0.0950	0.0950
Cy	0.0718	0.0718	0.0718
Cz	0.0096	0.0096	0.0098
Cl	-0.0042	-0.0040	-0.0040
Cn	0.0017	0.0017	0.0017
Cm	-1.00E-05	-1.68E-06	1.47E-06

Table E-3 Comparisons of numerical results for various pressure-velocity coupling

E. 2 Computational results of dynamic stability analysis

Angle (deg)	Cx	Cy	Cz	Cl	Cm	Cn
-5	0.0950	0.0717	0.0096	-0.0041	2.02E-05	0.0017
0	0.0951	0.0724	0.0034	-0.0042	-0.0002	0.0017
5	0.0950	0.0724	-0.0029	-0.0043	-0.0004	0.0016

Table E-4 Computational results for the rate of roll of 0.2 deg/s

Angle (deg)	Cx	Cy	Cz	Cl	Cm	Cn
-5	0.0950	0.0718	0.0096	-0.0950	-0.0096	-0.0718
0	0.0951	0.0724	0.0034	-0.0951	-0.0034	-0.0724
5	0.0951	0.0724	-0.0029	-0.0951	0.0029	-0.0724

Table E-5 Computational results for the rate of roll of 1 deg/s

Angle (deg)	Cx	Cy	Cz	Cl	Cm	Cn
-1	0.0940	0.0201	0.0039	-0.0044	0.0007	0.0017
0	0.0951	0.0724	0.0034	-0.0042	-0.0002	0.0017
1	0.0967	0.1252	0.0029	-0.0040	-0.0011	0.0016

Table E-6 Computational results for the rate of pitch of 0.0002 deg/s

Angle (deg)	Cx	Cy	Cz	Cl	Cm	Cn
-1	0.0941	0.0202	0.0038	-0.0043	0.0002	0.0014
0	0.0951	0.0724	0.0034	-0.0042	-0.0002	0.0017
1	0.0967	0.1251	0.0029	-0.0040	-0.0011	0.0016

Table E-7 Computational results for the rate of pitch of 0.0005 deg/s

Angle (deg)	Cx	Cy	Cz	Cl	Cm	Cn
-1	0.1187	0.0744	0.0292	-0.0043	-2.06E-05	0.0025
0	0.0951	0.0724	0.0034	-0.0042	-0.0002	0.0017
1	0.1186	0.0747	-0.0215	-0.0045	-2.65E-05	0.0011

Table E-8 Computational results for the rate of yaw of 0.0005 deg/s

Angle (deg)	Cx	Cy	Cz	Cl	Cm	Cn
-1	0.1187	0.0744	0.0292	-0.0043	-1.84E-05	0.0025
0	0.0951	0.0724	0.0034	-0.0042	-0.0002	0.0017
1	0.1186	0.0747	-0.0214	-0.0045	-3.05E-05	0.0011

Table E-9 Computational results for the rate of yaw of 0.001 deg/s

Appendix F Computational Results of the Improved-Component Buildup Method (I-CBM)

F.1 Improved-Component Buildup Method (I-CBM) for prediction of aerodynamic properties

FULL CONFIG			COUPLED			INDIVIDUAL			Duct
Angle-of-attack (deg)	CD	CL	Angle-of-attack (deg)	CD	CL	Angle-of-attack (deg)	CD	CL	
0	1.89E-02	4.86E-03	0	2.56E-02	1.47E-02	0	2.50E-02	4.00E-04	
10	5.33E-02	3.32E-01	10	6.33E-02	3.42E-01	10	5.91E-02	2.68E-01	
20	2.02E-01	5.45E-01	20	2.02E-01	5.38E-01	20	1.72E-01	4.49E-01	
Angle-of-attack (deg)	CD	CL	Angle-of-attack (deg)	CD	CL	Angle-of-attack (deg)	CD	CL	Wing
0	7.98E-03	9.84E-03	0	8.40E-03	9.88E-03	0	1.95E-02	3.33E-02	
10	2.86E-02	1.91E-01	10	2.83E-02	1.88E-01	10	6.54E-02	5.27E-01	
20	9.66E-02	3.24E-01	20	8.99E-02	2.99E-01	20	2.63E-01	8.14E-01	
Angle-of-attack (deg)	CD	CL	Angle-of-attack (deg)	CD	CL	Angle-of-attack (deg)	CD	CL	All
0	2.69E-02	1.47E-02	0	3.40E-02	2.46E-02	0	4.45E-02	3.37E-02	
10	8.19E-02	5.23E-01	10	9.16E-02	5.29E-01	10	1.24E-01	7.95E-01	
20	2.99E-01	8.69E-01	20	2.92E-01	8.37E-01	20	4.34E-01	1.26E+00	

Table F-1 Computational results of duct-wing for various angles-of-attack

FULL CONFIGURATION			INDIVIDUAL			Wing
Angle-of-attack (deg)	CD	CL	Angle-of-attack (deg)	CD	CL	
0	7.98E-03	9.84E-03	0	1.95E-02	3.33E-02	
10	2.86E-02	1.91E-01	10	6.54E-02	5.27E-01	
20	9.66E-02	3.24E-01	20	2.63E-01	8.14E-01	
Angle-of-attack (deg)	CD	CL	Angle-of-attack (deg)	CD	CL	Duct
0	1.89E-02	4.86E-03	0	2.50E-02	4.00E-04	
10	5.33E-02	3.32E-01	10	5.91E-02	2.68E-01	
20	2.02E-01	5.45E-01	20	1.72E-01	4.49E-01	
Angle-of-attack (deg)	CD	CL	Angle-of-attack (deg)	CD	CL	Horizontal tail
0	5.49E-03	1.67E-04	0	9.12E-03	4.37E-06	
10	5.49E-03	6.36E-04	10	1.34E-02	4.46E-02	
20	7.31E-03	7.91E-03	20	2.86E-02	8.74E-02	
Angle-of-attack (deg)	CD	CL	Angle-of-attack (deg)	CD	CL	Vertical tail
0	9.00E-03	2.14E-05	0	9.52E-03	-3.15E-06	
10	8.90E-03	2.16E-03	10	9.49E-03	3.20E-03	
20	8.19E-03	4.01E-03	20	9.46E-03	6.24E-03	
Angle-of-attack (deg)	CD	CL	Angle-of-attack (deg)	CD	CL	Stator
0	3.84E-02	1.37E-04	0	3.71E-02	6.24E-05	
10	3.91E-02	1.56E-02	10	4.39E-02	6.61E-02	
20	4.19E-02	3.39E-02	20	6.95E-02	1.15E-01	

Table F-2 Computational results of individual and full configuration models with stator set at 0° - 0° for various angles-of-attack

FULL CONFIGURATION		
Angle-of-attack (deg)	CD	CL
0	8.01E-03	1.52E-02
10	2.96E-02	1.99E-01
20	1.00E-01	3.38E-01

Angle-of-attack (deg)	CD	CL
0	1.83E-02	3.86E-03
10	5.19E-02	3.31E-01
20	1.99E-01	5.36E-01

Angle-of-attack (deg)	CD	CL
0	5.28E-03	-1.19E-02
10	3.29E-03	-1.14E-02
20	4.23E-03	-1.49E-03

Angle-of-attack (deg)	CD	CL
0	9.08E-03	4.44E-04
10	9.13E-03	2.57E-03
20	8.31E-03	4.31E-03

Angle-of-attack (deg)	CD	CL
0	4.62E-02	5.12E-02
10	5.68E-02	6.71E-02
20	6.90E-02	8.51E-02

INDIVIDUAL		
Angle-of-attack (deg)	CD	CL
0	1.95E-02	3.33E-02
10	6.54E-02	5.27E-01
20	2.63E-01	8.14E-01

Angle-of-attack (deg)	CD	CL
0	2.50E-02	4.00E-04
10	5.91E-02	2.68E-01
20	1.72E-01	4.49E-01

Angle-of-attack (deg)	CD	CL
0	9.12E-03	4.37E-06
10	1.34E-02	4.46E-02
20	2.86E-02	8.74E-02

Angle-of-attack (deg)	CD	CL
0	9.52E-03	-3.15E-06
10	9.49E-03	3.20E-03
20	9.46E-03	6.24E-03

Angle-of-attack (deg)	CD	CL
0	4.53E-02	5.65E-02
10	7.12E-02	1.10E-01
20	1.07E-01	1.52E-01

Wing

Duct

Horizontal tail

Vertical tail

Stator

Table F-3 Computational results of individual and full configuration models with stator set at 10° - 10° for various angles-of-attack

FULL CONFIGURATION		
Angle-of-attack (deg)	CD	CL
0	7.98E-03	9.97E-03
10	2.91E-02	1.91E-01
20	9.65E-02	3.23E-01

Angle-of-attack (deg)	CD	CL
0	1.88E-02	4.92E-03
10	5.38E-02	3.33E-01
20	2.02E-01	5.42E-01

Angle-of-attack (deg)	CD	CL
0	5.50E-03	2.13E-04
10	5.55E-03	9.29E-04
20	7.83E-03	9.34E-03

Angle-of-attack (deg)	CD	CL
0	9.00E-03	1.47E-05
10	8.91E-03	2.16E-03
20	8.18E-03	4.06E-03

Angle-of-attack (deg)	CD	CL
0	4.03E-02	3.27E-04
10	4.11E-02	1.63E-02
20	4.44E-02	3.86E-02

INDIVIDUAL		
Angle-of-attack (deg)	CD	CL
0	1.95E-02	3.33E-02
10	6.54E-02	5.27E-01
20	2.63E-01	8.14E-01

Angle-of-attack (deg)	CD	CL
0	2.50E-02	4.00E-04
10	5.91E-02	2.68E-01
20	1.72E-01	4.49E-01

Angle-of-attack (deg)	CD	CL
0	9.12E-03	4.37E-06
10	1.34E-02	4.46E-02
20	2.86E-02	8.74E-02

Angle-of-attack (deg)	CD	CL
0	9.52E-03	-3.15E-06
10	9.49E-03	3.20E-03
20	9.46E-03	6.24E-03

Angle-of-attack (deg)	CD	CL
0	3.89E-02	-1.51E-04
10	4.58E-02	6.66E-02
20	7.01E-02	1.16E-01

Wing

Duct

Horizontal tail

Vertical tail

Stator

Table F-4 Computational results of individual and full configuration models with stator set at - 5° - 5° for various angles-of-attack

FULL CONFIGURATION			INDIVIDUAL			Wing
Angle-of-attack (deg)	CD	CL	Angle-of-attack (deg)	CD	CL	
0	8.09E-03	1.02E-02	0	1.95E-02	3.33E-02	
10	2.88E-02	1.91E-01	10	6.54E-02	5.27E-01	
20	9.70E-02	3.27E-01	20	2.63E-01	8.14E-01	Duct
Angle-of-attack (deg)	CD	CL	Angle-of-attack (deg)	CD	CL	
0	1.85E-02	5.05E-03	0	2.50E-02	4.00E-04	
10	5.32E-02	3.32E-01	10	5.91E-02	2.68E-01	
20	2.02E-01	5.44E-01	20	1.72E-01	4.49E-01	Horizontal tail
Angle-of-attack (deg)	CD	CL	Angle-of-attack (deg)	CD	CL	
0	5.36E-03	1.28E-04	0	9.12E-03	4.37E-06	
10	5.35E-03	6.52E-04	10	1.34E-02	4.46E-02	
20	8.14E-03	1.00E-02	20	2.86E-02	8.74E-02	Vertical tail
Angle-of-attack (deg)	CD	CL	Angle-of-attack (deg)	CD	CL	
0	9.29E-03	3.12E-06	0	9.52E-03	-3.15E-06	
10	9.22E-03	2.20E-03	10	9.49E-03	3.20E-03	
20	8.48E-03	4.21E-03	20	9.46E-03	6.24E-03	Stator
Angle-of-attack (deg)	CD	CL	Angle-of-attack (deg)	CD	CL	
0	4.53E-02	2.29E-04	0	4.49E-02	5.67E-05	
10	4.56E-02	1.64E-02	10	5.42E-02	5.85E-02	
20	4.78E-02	3.65E-02	20	7.45E-02	1.09E-01	

**Table F-5 Computational results of individual and full configuration models with stator set at -
10° - 10° for various angles-of-attack**

FULL CONFIGURATION			INDIVIDUAL			Wing
Angle-of-attack (deg)	CD	CL	Angle-of-attack (deg)	CD	CL	
0	8.01E-03	1.12E-02	0	1.95E-02	3.33E-02	
10	2.91E-02	1.93E-01	10	6.54E-02	5.27E-01	
20	9.78E-02	3.28E-01	20	2.63E-01	8.14E-01	Duct
Angle-of-attack (deg)	CD	CL	Angle-of-attack (deg)	CD	CL	
0	1.88E-02	4.65E-03	0	2.50E-02	4.00E-04	
10	5.35E-02	3.32E-01	10	5.91E-02	2.68E-01	
20	2.01E-01	5.38E-01	20	1.72E-01	4.49E-01	Horizontal tail
Angle-of-attack (deg)	CD	CL	Angle-of-attack (deg)	CD	CL	
0	5.49E-03	-2.67E-03	0	9.12E-03	4.37E-06	
10	5.05E-03	-2.06E-03	10	1.34E-02	4.46E-02	
20	6.83E-03	6.10E-03	20	2.86E-02	8.74E-02	Vertical tail
Angle-of-attack (deg)	CD	CL	Angle-of-attack (deg)	CD	CL	
0	9.02E-03	1.36E-04	0	9.52E-03	-3.15E-06	
10	8.98E-03	2.28E-03	10	9.49E-03	3.20E-03	
20	8.29E-03	4.16E-03	20	9.46E-03	6.24E-03	Stator
Angle-of-attack (deg)	CD	CL	Angle-of-attack (deg)	CD	CL	
0	3.94E-02	1.27E-02	0	3.80E-02	1.49E-02	
10	4.26E-02	2.80E-02	10	4.85E-02	8.04E-02	
20	4.90E-02	4.98E-02	20	7.76E-02	1.26E-01	

**Table F-6 Computational results of individual and full configuration models with stator set at
0° - 5° for various angles-of-attack**

FULL CONFIGURATION			INDIVIDUAL			Wing
Angle-of-attack (deg)	CD	CL	Angle-of-attack (deg)	CD	CL	
0	8.01E-03	1.25E-02	0	1.95E-02	3.33E-02	
10	2.92E-02	1.96E-01	10	6.54E-02	5.27E-01	
20	9.85E-02	3.29E-01	20	2.63E-01	8.14E-01	Duct
Angle-of-attack (deg)	CD	CL	Angle-of-attack (deg)	CD	CL	
0	1.87E-02	4.38E-03	0	2.50E-02	4.00E-04	
10	5.26E-02	3.32E-01	10	5.91E-02	2.68E-01	
20	1.99E-01	5.35E-01	20	1.72E-01	4.49E-01	Horizontal tail
Angle-of-attack (deg)	CD	CL	Angle-of-attack (deg)	CD	CL	
0	5.36E-03	-5.92E-03	0	9.12E-03	4.37E-06	
10	4.37E-03	-5.25E-03	10	1.34E-02	4.46E-02	
20	5.95E-03	3.64E-03	20	2.86E-02	8.74E-02	Vertical tail
Angle-of-attack (deg)	CD	CL	Angle-of-attack (deg)	CD	CL	
0	9.04E-03	2.34E-04	0	9.52E-03	-3.15E-06	
10	9.01E-03	2.37E-03	10	9.49E-03	3.20E-03	
20	8.27E-03	4.21E-03	20	9.46E-03	6.24E-03	Stator
Angle-of-attack (deg)	CD	CL	Angle-of-attack (deg)	CD	CL	
0	4.22E-02	2.55E-02	0	4.11E-02	2.83E-02	
10	4.81E-02	4.13E-02	10	5.79E-02	8.74E-02	
20	5.63E-02	6.22E-02	20	8.68E-02	1.34E-01	

Table F-7 Computational results of individual and full configuration models with stator set at 10° - 0° for various angles-of-attack

FULL CONFIG			INDIVIDUAL			Wing
Angle-of-attack (deg)	CD	CL	Angle-of-attack (deg)	CD	CL	
0	7.98E-03	9.70E-03	0	1.95E-02	3.33E-02	
10	2.88E-02	1.91E-01	10	6.54E-02	5.27E-01	
20	9.69E-02	3.24E-01	20	2.63E-01	8.14E-01	Duct
Angle-of-attack (deg)	CD	CL	Angle-of-attack (deg)	CD	CL	
0	1.85E-02	4.61E-03	0	2.50E-02	4.00E-04	
10	5.31E-02	3.32E-01	10	5.91E-02	2.68E-01	
20	2.02E-01	5.43E-01	20	1.72E-01	4.49E-01	Horizontal tail
Angle-of-attack (deg)	CD	CL	Angle-of-attack (deg)	CD	CL	
0	5.19E-03	-5.46E-03	0	8.54E-03	-7.97E-03	
10	4.17E-03	-5.32E-03	10	1.10E-02	3.54E-02	
20	5.38E-03	2.97E-03	20	2.39E-02	7.81E-02	Vertical tail
Angle-of-attack (deg)	CD	CL	Angle-of-attack (deg)	CD	CL	
0	9.24E-03	1.36E-05	0	9.52E-03	-3.15E-06	
10	9.19E-03	2.24E-03	10	9.49E-03	3.20E-03	
20	8.51E-03	4.28E-03	20	9.46E-03	6.24E-03	Stator
Angle-of-attack (deg)	CD	CL	Angle-of-attack (deg)	CD	CL	
0	4.46E-02	3.06E-04	0	4.49E-02	5.57E-05	
10	4.49E-02	1.58E-02	10	5.42E-02	5.85E-02	
20	4.61E-02	3.50E-02	20	7.45E-02	1.09E-01	

Table F-8 Computational results of individual and full configuration models with horizontal tail set at -2° for various angles-of-attack

FULL CONFIG			INDIVIDUAL			Wing
Angle-of-attack (deg)	CD	CL	Angle-of-attack (deg)	CD	CL	
0	7.98E-03	1.02E-02	0	1.95E-02	3.33E-02	
10	2.89E-02	1.91E-01	10	6.54E-02	5.27E-01	Duct
20	9.70E-02	3.25E-01	20	2.63E-01	8.14E-01	
Angle-of-attack (deg)	CD	CL	Angle-of-attack (deg)	CD	CL	
0	1.85E-02	5.36E-03	0	2.50E-02	4.00E-04	Horizontal tail
10	5.35E-02	3.32E-01	10	5.91E-02	2.68E-01	
20	2.02E-01	5.43E-01	20	1.72E-01	4.49E-01	
Angle-of-attack (deg)	CD	CL	Angle-of-attack (deg)	CD	CL	Vertical tail
0	5.18E-03	5.79E-03	0	8.53E-03	7.67E-03	
10	6.12E-03	5.84E-03	10	1.48E-02	5.49E-02	
20	1.01E-02	1.54E-02	20	3.32E-02	8.83E-02	Stator
Angle-of-attack (deg)	CD	CL	Angle-of-attack (deg)	CD	CL	
0	9.27E-03	6.72E-05	0	9.52E-03	-3.15E-06	
10	9.23E-03	2.30E-03	10	9.49E-03	3.20E-03	
20	8.50E-03	4.24E-03	20	9.46E-03	6.24E-03	
Angle-of-attack (deg)	CD	CL	Angle-of-attack (deg)	CD	CL	
0	4.46E-02	1.13E-03	0	4.49E-02	5.57E-05	
10	4.50E-02	1.67E-02	10	5.42E-02	5.85E-02	
20	4.74E-02	3.74E-02	20	7.45E-02	1.09E-01	

Table F-9 Computational results of individual and full configuration models with horizontal tail set at 2° for various angles-of-attack

F. 2 Improved-Component Buildup Method (I-CBM) for prediction of stability derivatives

FULL CONFIG			COUPLED			INDIVIDUAL			Duct
Sideslip-angle (deg)	CI	Cn	Sideslip-angle (deg)	CI	Cn	Sideslip-angle (deg)	CI	Cn	
-10	3.02E-04	-2.61E-03	-10	3.02E-04	-1.18E-04	-10	3.00E-04	-1.15E-03	
-5	1.60E-04	-7.78E-04	-5	1.62E-04	6.55E-04	-5	1.53E-04	-2.82E-04	Wing
0	2.01E-05	1.50E-05	0	1.62E-05	-6.11E-05	0	-2.29E-07	1.58E-06	
5	-1.21E-04	7.75E-04	5	-1.30E-04	-8.44E-04	5	-1.51E-04	1.70E-04	
10	-2.64E-04	2.60E-03	10	-2.72E-04	-9.27E-05	10	-2.99E-04	1.05E-03	All
Sideslip-angle (deg)	CI	Cn	Sideslip-angle (deg)	CI	Cn	Sideslip-angle (deg)	CI	Cn	
-10	2.29E-04	-1.44E-03	-10	1.19E-03	-1.41E-03	-10	6.66E-04	-8.88E-04	
-5	-3.84E-04	-7.43E-04	-5	6.11E-04	-7.26E-04	-5	4.01E-04	-4.81E-04	
0	-9.72E-04	-2.09E-05	0	1.65E-05	-3.16E-05	0	1.41E-04	-6.90E-05	
5	-1.55E-03	6.99E-04	5	-5.78E-04	6.66E-04	5	-1.11E-04	3.42E-04	
10	-2.12E-03	1.37E-03	10	-1.16E-03	1.36E-03	10	-4.13E-04	7.51E-04	
Sideslip-angle (deg)	CI	Cn	Sideslip-angle (deg)	CI	Cn	Sideslip-angle (deg)	CI	Cn	
-10	5.31E-04	-4.04E-03	-10	1.50E-03	-1.53E-03	-10	9.66E-04	-2.03E-03	
-5	-2.23E-04	-1.52E-03	-5	7.74E-04	-7.07E-05	-5	5.53E-04	-7.63E-04	
0	-9.52E-04	-5.88E-06	0	3.27E-05	-9.28E-05	0	1.41E-04	-6.75E-05	
5	-1.67E-03	1.47E-03	5	-7.08E-04	-1.78E-04	5	-2.61E-04	5.12E-04	
10	-2.39E-03	3.97E-03	10	-1.43E-03	1.27E-03	10	-7.13E-04	1.80E-03	

Table F-10 Computational results of duct-wing for various sideslip-angles

FULL CONFIGURATION			INDIVIDUAL			Wing
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	
-10	1.21E-03	-1.45E-03	-10	6.66E-04	-8.88E-04	
-5	5.92E-04	-7.54E-04	-5	4.01E-04	-4.81E-04	
0	-1.51E-05	-2.70E-05	0	1.41E-04	-6.90E-05	
5	-6.00E-04	7.02E-04	5	-1.11E-04	3.42E-04	
10	-1.21E-03	1.40E-03	10	-4.13E-04	7.51E-04	
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	Duct
-10	2.97E-04	-2.45E-03	-10	3.00E-04	-1.15E-03	
-5	1.56E-04	-7.21E-04	-5	1.53E-04	-2.82E-04	
0	1.62E-05	-8.00E-06	0	-2.29E-07	1.58E-06	
5	-1.25E-04	6.56E-04	5	-1.51E-04	1.70E-04	
10	-2.68E-04	2.35E-03	10	-2.99E-04	4.62E-03	
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	All
-10	1.51E-03	-3.90E-03	-10	9.66E-04	-2.03E-03	
-5	7.48E-04	-1.47E-03	-5	5.53E-04	-7.63E-04	
0	1.09E-06	-3.50E-05	0	1.41E-04	-6.75E-05	
5	-7.25E-04	1.36E-03	5	-2.61E-04	5.12E-04	
10	-1.48E-03	3.75E-03	10	-7.13E-04	5.37E-03	

Table F-11 Computational results of duct-wing with stator set at 0° - 0° for various sideslip-angles

FULL CONFIGURATION			INDIVIDUAL			Wing
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	
-10	2.29E-04	-1.44E-03	-10	6.66E-04	-8.88E-04	
-5	-3.84E-04	-7.43E-04	-5	4.01E-04	-4.81E-04	
0	-9.72E-04	-2.09E-05	0	1.41E-04	-6.90E-05	
5	-1.55E-03	6.99E-04	5	-1.11E-04	3.42E-04	
10	-2.12E-03	1.37E-03	10	-4.13E-04	7.51E-04	
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	Duct
-10	3.02E-04	-2.61E-03	-10	3.00E-04	-1.15E-03	
-5	1.60E-04	-7.78E-04	-5	1.53E-04	-2.82E-04	
0	2.01E-05	1.50E-05	0	-2.29E-07	1.58E-06	
5	-1.21E-04	7.75E-04	5	-1.51E-04	1.70E-04	
10	-2.64E-04	2.60E-03	10	-2.99E-04	4.62E-03	
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	All
-10	5.31E-04	-4.04E-03	-10	9.66E-04	-2.03E-03	
-5	-2.23E-04	-1.52E-03	-5	5.53E-04	-7.63E-04	
0	-9.52E-04	-5.88E-06	0	1.41E-04	-6.75E-05	
5	-1.67E-03	1.47E-03	5	-2.61E-04	5.12E-04	
10	-2.39E-03	3.97E-03	10	-7.13E-04	5.37E-03	

Table F-12 Computational results of duct-wing with stator set at -10° - 10° for various sideslip-angles

FULL CONFIGURATION			INDIVIDUAL			Wing
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	
-10	1.73E-03	-1.45E-03	-10	6.66E-04	-8.88E-04	
-5	1.13E-03	-7.63E-04	-5	4.01E-04	-4.81E-04	
0	5.25E-04	-3.55E-05	0	1.41E-04	-6.90E-05	
5	-5.93E-05	6.92E-04	5	-1.11E-04	3.42E-04	
10	-6.56E-04	1.39E-03	10	-4.13E-04	7.51E-04	
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	Duct
-10	2.93E-04	-2.35E-03	-10	3.00E-04	-1.15E-03	
-5	1.51E-04	-5.96E-04	-5	1.53E-04	-2.82E-04	
0	1.13E-05	1.56E-04	0	-2.29E-07	1.58E-06	
5	-1.29E-04	8.58E-04	5	-1.51E-04	1.70E-04	
10	-2.73E-04	2.59E-03	10	-2.99E-04	4.62E-03	
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	All
-10	2.03E-03	-3.80E-03	-10	9.66E-04	-2.03E-03	
-5	1.28E-03	-1.36E-03	-5	5.53E-04	-7.63E-04	
0	5.36E-04	1.20E-04	0	1.41E-04	-6.75E-05	
5	-1.89E-04	1.55E-03	5	-2.61E-04	5.12E-04	
10	-9.29E-04	3.98E-03	10	-7.13E-04	5.37E-03	

Table F-13 Computational results of duct-wing with stator set at 10° - 0° for various sideslip-angles

FULL CONFIGURATION			INDIVIDUAL			Wing
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	
-10	1.01E-03	-1.44E-03	-10	6.66E-04	-8.88E-04	
-5	3.79E-04	-7.49E-04	-5	4.01E-04	-4.81E-04	
0	-2.29E-04	-2.58E-05	0	1.41E-04	-6.90E-05	
5	-8.14E-04	7.01E-04	5	-1.11E-04	3.42E-04	
10	-1.43E-03	1.40E-03	10	-4.13E-04	7.51E-04	
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	Duct
-10	2.98E-04	-2.50E-03	-10	3.00E-04	-1.15E-03	
-5	1.57E-04	-7.76E-04	-5	1.53E-04	-2.82E-04	
0	1.70E-05	-4.61E-05	0	-2.29E-07	1.58E-06	
5	-1.24E-04	6.23E-04	5	-1.51E-04	1.70E-04	
10	-2.68E-04	2.35E-03	10	-2.99E-04	4.62E-03	
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	All
-10	1.31E-03	-3.95E-03	-10	9.66E-04	-2.03E-03	
-5	5.37E-04	-1.53E-03	-5	5.53E-04	-7.63E-04	
0	-2.12E-04	-7.19E-05	0	1.41E-04	-6.75E-05	
5	-9.38E-04	1.32E-03	5	-2.61E-04	5.12E-04	
10	-1.70E-03	3.75E-03	10	-7.13E-04	5.37E-03	

Table F-14 Computational results of duct-wing with stator set at 0° - 5° for various sideslip-angles

FULL CONFIGURATION			INDIVIDUAL			Wing
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	
-10	1.88E-04	-1.43E-03	-10	6.66E-04	-8.88E-04	
-5	-4.18E-04	-7.43E-04	-5	4.01E-04	-4.81E-04	
0	-1.01E-03	-1.74E-05	0	1.41E-04	-6.90E-05	
5	-1.59E-03	7.11E-04	5	-1.11E-04	3.42E-04	
10	-2.19E-03	1.41E-03	10	-4.13E-04	7.51E-04	
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	Duct
-10	3.00E-04	-2.58E-03	-10	3.00E-04	-1.15E-03	
-5	1.59E-04	-7.65E-04	-5	1.53E-04	-2.82E-04	
0	1.84E-05	2.58E-06	0	-2.29E-07	1.58E-06	
5	-1.22E-04	7.38E-04	5	-1.51E-04	1.70E-04	
10	-2.65E-04	2.49E-03	10	-2.99E-04	4.62E-03	
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	All
-10	4.88E-04	-4.01E-03	-10	9.66E-04	-2.03E-03	
-5	-2.58E-04	-1.51E-03	-5	5.53E-04	-7.63E-04	
0	-9.87E-04	-1.48E-05	0	1.41E-04	-6.75E-05	
5	-1.71E-03	1.45E-03	5	-2.61E-04	5.12E-04	
10	-2.46E-03	3.90E-03	10	-7.13E-04	5.37E-03	

Table F-15 Computational results of duct-wing with horizontal tail set at -2° for various sideslip-angles

FULL CONFIGURATION			INDIVIDUAL			Wing
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	
-10	2.21E-04	-1.44E-03	-10	6.66E-04	-8.88E-04	
-5	-3.97E-04	-7.44E-04	-5	4.01E-04	-4.81E-04	
0	-9.99E-04	-1.76E-05	0	1.41E-04	-6.90E-05	
5	-1.59E-03	7.11E-04	5	-1.11E-04	3.42E-04	
10	-2.19E-03	1.40E-03	10	-4.13E-04	7.51E-04	
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	Duct
-10	3.00E-04	-2.54E-03	-10	-2.54E-03	-1.15E-03	
-5	1.59E-04	-7.72E-04	-5	-7.72E-04	-2.82E-04	
0	1.87E-05	1.19E-05	0	1.19E-05	1.58E-06	
5	-1.22E-04	7.50E-04	5	7.50E-04	1.70E-04	
10	-2.65E-04	2.51E-03	10	2.51E-03	4.62E-03	
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	All
-10	5.21E-04	-3.98E-03	-10	-1.88E-03	-2.03E-03	
-5	-2.37E-04	-1.52E-03	-5	-3.71E-04	-7.63E-04	
0	-9.80E-04	-5.69E-06	0	1.53E-04	-6.75E-05	
5	-1.71E-03	1.46E-03	5	6.39E-04	5.12E-04	
10	-2.46E-03	3.92E-03	10	2.10E-03	5.37E-03	

Table F-16 Computational results of duct-wing with vertical tail set at -2° for various sideslip-angles

FULL CONFIG			INDIVIDUAL			Wing
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	
-10	1.21E-03	-1.45E-03	-10	6.66E-04	-8.88E-04	
-5	5.92E-04	-7.54E-04	-5	4.01E-04	-4.81E-04	
0	-1.51E-05	-2.70E-05	0	1.41E-04	-6.90E-05	
5	-6.00E-04	7.02E-04	5	-1.11E-04	3.42E-04	
10	-1.21E-03	1.40E-03	10	-4.13E-04	7.51E-04	
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	Duct
-10	2.97E-04	-2.45E-03	-10	3.00E-04	-1.15E-03	
-5	1.56E-04	-7.21E-04	-5	1.53E-04	-2.82E-04	
0	1.62E-05	-8.00E-06	0	-2.29E-07	1.58E-06	
5	-1.25E-04	6.56E-04	5	-1.51E-04	1.70E-04	
10	-2.68E-04	2.35E-03	10	-2.99E-04	4.62E-03	
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	Horizontal tail
-10	-6.04E-05	-1.44E-04	-10	-5.39E-05	-1.09E-03	
-5	-2.63E-05	-8.23E-05	-5	-5.69E-05	-5.39E-04	
0	1.22E-05	-3.95E-05	0	-3.98E-05	-2.41E-05	
5	5.33E-05	3.11E-06	5	-4.60E-05	4.99E-04	
10	9.66E-05	6.04E-05	10	-5.59E-05	1.04E-03	
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	Vertical tail
-10	1.25E-05	-4.53E-03	-10	4.39E-05	-2.48E-02	
-5	5.72E-06	-1.86E-03	-5	3.30E-05	-1.20E-02	
0	-1.82E-05	1.93E-04	0	3.49E-06	7.47E-05	
5	-4.62E-05	2.23E-03	5	-2.27E-05	1.19E-02	
10	-7.24E-05	4.91E-03	10	-4.71E-05	2.47E-02	
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	Stator
-10	-1.31E-04	-3.73E-04	-10	-1.13E-04	-2.77E-03	
-5	-9.36E-05	-2.53E-04	-5	-8.40E-05	-1.41E-03	
0	-5.63E-05	-1.57E-04	0	-2.60E-05	-1.26E-04	
5	-1.95E-05	-6.08E-05	5	-7.00E-05	1.19E-03	
10	2.10E-05	7.31E-05	10	-8.58E-05	2.50E-03	

Table F-17 Computational results of individual and full configuration models with stator set at 0° - 0° for various sideslip-angles

FULL CONFIG			INDIVIDUAL			Wing
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	
-10	2.29E-04	-1.44E-03	-10	6.66E-04	-8.88E-04	
-5	-3.84E-04	-7.43E-04	-5	4.01E-04	-4.81E-04	
0	-9.72E-04	-2.09E-05	0	1.41E-04	-6.90E-05	
5	-1.55E-03	6.99E-04	5	-1.11E-04	3.42E-04	
10	-1.87E-03	1.27E-03	10	-4.13E-04	7.51E-04	
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	Duct
-10	3.02E-04	-2.61E-03	-10	3.00E-04	-1.15E-03	
-5	1.60E-04	-7.78E-04	-5	-1.53E-04	-1.53E-04	
0	2.01E-05	1.50E-05	0	2.29E-07	2.29E-07	
5	-1.21E-04	7.75E-04	5	1.51E-04	1.51E-04	
10	-2.63E-04	2.94E-03	10	2.99E-04	2.99E-04	
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	Horizontal tail
-10	1.68E-03	-1.83E-04	-10	-5.39E-05	-1.09E-03	
-5	1.79E-03	-9.00E-05	-5	-5.69E-05	-5.39E-04	
0	1.85E-03	-1.42E-05	0	-3.98E-05	-2.41E-05	
5	1.88E-03	6.19E-05	5	-4.60E-05	4.99E-04	
10	1.86E-03	1.25E-04	10	-5.59E-05	1.04E-03	
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	Vertical tail
-10	1.04E-03	-5.25E-03	-10	4.39E-05	-2.48E-02	
-5	9.92E-04	-2.36E-03	-5	3.30E-05	-1.20E-02	
0	9.47E-04	-1.43E-04	0	3.49E-06	7.47E-05	
5	9.20E-04	2.04E-03	5	-2.27E-05	1.19E-02	
10	8.74E-04	4.92E-03	10	-4.71E-05	2.47E-02	
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	Stator
-10	-6.22E-03	-3.93E-04	-10	-7.74E-03	-3.31E-03	
-5	-6.33E-03	-2.44E-04	-5	-7.53E-03	-1.82E-03	
0	-6.32E-03	-1.23E-04	0	-7.42E-03	-1.22E-04	
5	-6.25E-03	-7.12E-06	5	-7.50E-03	1.52E-03	
10	-6.01E-03	7.57E-05	10	-7.71E-03	3.04E-03	

Table F-18 Computational results of individual and full configuration models with stator set at -10° - 10° for various sideslip-angles

FULL CONFIG			INDIVIDUAL			Wing
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	
-10	1.73E-03	-1.45E-03	-10	6.66E-04	-8.88E-04	
-5	1.13E-03	-7.63E-04	-5	4.01E-04	-4.81E-04	
0	5.25E-04	-3.55E-05	0	1.41E-04	-6.90E-05	
5	-5.93E-05	6.92E-04	5	-1.11E-04	3.42E-04	Duct
10	-6.56E-04	1.39E-03	10	-4.13E-04	7.51E-04	
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	
-10	2.93E-04	-2.35E-03	-10	3.00E-04	-1.15E-03	
-5	1.51E-04	-5.96E-04	-5	1.53E-04	-2.82E-04	
0	1.13E-05	1.56E-04	0	-2.29E-07	1.58E-06	
5	-1.29E-04	8.58E-04	5	-1.51E-04	1.70E-04	Horizontal tail
10	-2.73E-04	2.59E-03	10	-2.99E-04	4.62E-03	
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	
-10	-8.78E-04	-1.44E-04	-10	-5.39E-05	-1.09E-03	
-5	-8.93E-04	-5.82E-05	-5	-5.69E-05	-5.39E-04	
0	-8.91E-04	7.34E-06	0	-3.98E-05	-2.41E-05	Vertical tail
5	-8.78E-04	6.09E-05	5	-4.60E-05	4.99E-04	
10	-8.44E-04	1.22E-04	10	-5.59E-05	1.04E-03	
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	
-10	-4.81E-04	-5.32E-03	-10	4.39E-05	-2.48E-02	
-5	-4.92E-04	-2.53E-03	-5	3.30E-05	-1.20E-02	Stator
0	-5.12E-04	-4.35E-04	0	3.49E-06	7.47E-05	
5	-5.41E-04	1.91E-03	5	-2.27E-05	1.19E-02	
10	-5.77E-04	4.37E-03	10	-4.71E-05	2.47E-02	
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	
-10	3.06E-03	-8.03E-04	-10	4.50E-03	-3.50E-03	
-5	3.21E-03	-6.65E-04	-5	4.13E-03	-2.04E-03	
0	3.24E-03	-5.64E-04	0	3.79E-03	-5.39E-04	
5	3.24E-03	-4.46E-04	5	3.56E-03	9.07E-04	
10	3.24E-03	-2.96E-04	10	3.45E-03	2.34E-03	

Table F-19 Computational results of individual and full configuration models with stator set at 10° - 0° for various sideslip-angles

FULL CONFIG			INDIVIDUAL			Wing
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	
-10	1.01E-03	-1.44E-03	-10	6.66E-04	-8.88E-04	
-5	3.79E-04	-7.49E-04	-5	4.01E-04	-4.81E-04	
0	-2.29E-04	-2.58E-05	0	1.41E-04	-6.90E-05	
5	-8.14E-04	7.01E-04	5	-1.11E-04	3.42E-04	
10	-1.43E-03	1.40E-03	10	-4.13E-04	7.51E-04	
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	Duct
-10	2.98E-04	-2.50E-03	-10	3.00E-04	-1.15E-03	
-5	1.57E-04	-7.76E-04	-5	1.53E-04	-2.82E-04	
0	1.70E-05	-4.61E-05	0	-2.29E-07	1.58E-06	
5	-1.24E-04	6.23E-04	5	-1.51E-04	1.70E-04	
10	-2.68E-04	2.35E-03	10	-2.99E-04	4.62E-03	
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	Horizontal tail
-10	3.65E-04	-1.25E-04	-10	-5.39E-05	-1.09E-03	
-5	4.13E-04	-6.96E-05	-5	-5.69E-05	-5.39E-04	
0	4.42E-04	-2.40E-05	0	-3.98E-05	-2.41E-05	
5	4.64E-04	2.60E-05	5	-4.60E-05	4.99E-04	
10	4.67E-04	9.42E-05	10	-5.59E-05	1.04E-03	
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	Vertical tail
-10	2.65E-04	-4.74E-03	-10	4.39E-05	-2.48E-02	
-5	2.55E-04	-1.98E-03	-5	3.30E-05	-1.20E-02	
0	2.46E-04	9.40E-06	0	3.49E-06	7.47E-05	
5	2.23E-04	2.09E-03	5	-2.27E-05	1.19E-02	
10	1.99E-04	4.86E-03	10	-4.71E-05	2.47E-02	
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	Stator
-10	-1.57E-03	-2.63E-04	-10	-1.59E-03	-2.75E-03	
-5	-1.56E-03	-1.37E-04	-5	-1.71E-03	-1.42E-03	
0	-1.54E-03	-3.50E-05	0	-1.81E-03	-6.77E-05	
5	-1.50E-03	9.58E-05	5	-1.96E-03	1.29E-03	
10	-1.40E-03	2.05E-04	10	-2.23E-03	2.65E-03	

Table F-20 Computational results of individual and full configuration models with stator set at 0° - 5° for various sideslip-angles

FULL CONFIG			INDIVIDUAL			Wing
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	
-10	1.88E-04	-1.43E-03	-10	6.66E-04	-8.88E-04	
-5	-4.18E-04	-7.43E-04	-5	4.01E-04	-4.81E-04	
0	-1.01E-03	-1.74E-05	0	1.41E-04	-6.90E-05	
5	-1.59E-03	7.11E-04	5	-1.11E-04	3.42E-04	
10	-2.19E-03	1.41E-03	10	-4.13E-04	7.51E-04	
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	Duct
-10	3.00E-04	-2.58E-03	-10	3.00E-04	-1.15E-03	
-5	1.59E-04	-7.65E-04	-5	1.53E-04	-2.82E-04	
0	1.84E-05	2.58E-06	0	-2.29E-07	1.58E-06	
5	-1.22E-04	7.38E-04	5	-1.51E-04	1.70E-04	
10	-2.65E-04	2.49E-03	10	-2.99E-04	4.62E-03	
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	Horizontal tail
-10	1.64E-03	-2.21E-04	-10	-2.22E-04	-1.13E-03	
-5	1.79E-03	-1.39E-04	-5	-1.51E-04	-5.67E-04	
0	1.90E-03	-5.91E-05	0	-2.98E-05	-2.84E-05	
5	1.97E-03	3.40E-05	5	6.84E-05	5.08E-04	
10	1.99E-03	1.47E-04	10	9.60E-05	1.08E-03	
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	Vertical tail
-10	1.04E-03	-5.18E-03	-10	4.39E-05	-2.48E-02	
-5	9.81E-04	-2.23E-03	-5	3.30E-05	-1.20E-02	
0	9.23E-04	-3.16E-05	0	3.49E-06	7.47E-05	
5	8.81E-04	2.13E-03	5	-2.27E-05	1.19E-02	
10	8.60E-04	5.01E-03	10	-4.71E-05	2.47E-02	
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	Stator
-10	-6.25E-03	-4.10E-04	-10	-7.74E-03	-3.31E-03	
-5	-6.37E-03	-2.67E-04	-5	-7.53E-03	-1.82E-03	
0	-6.38E-03	-1.36E-04	0	-7.42E-03	-1.22E-04	
5	-6.30E-03	-1.23E-05	5	-7.50E-03	1.52E-03	
10	-6.10E-03	1.48E-04	10	-7.71E-03	3.04E-03	

Table F-21 Computational results of individual and full configuration models with horizontal tail set at - 2° for various sideslip-angles

FULL CONFIG			INDIVIDUAL			Wing
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	
-10	2.21E-04	-1.44E-03	-10	6.66E-04	-8.88E-04	
-5	-3.97E-04	-7.44E-04	-5	4.01E-04	-4.81E-04	
0	-9.99E-04	-1.76E-05	0	1.41E-04	-6.90E-05	
5	-1.59E-03	7.11E-04	5	-1.11E-04	3.42E-04	Duct
10	-2.19E-03	1.40E-03	10	-4.13E-04	7.51E-04	
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	
-10	3.00E-04	-2.54E-03	-10	-2.54E-03	-1.15E-03	
-5	1.59E-04	-7.72E-04	-5	-7.72E-04	-2.82E-04	
0	1.87E-05	1.19E-05	0	1.19E-05	1.58E-06	
5	-1.22E-04	7.50E-04	5	7.50E-04	1.70E-04	Horizontal tail
10	-2.65E-04	2.51E-03	10	2.51E-03	4.62E-03	
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	
-10	1.77E-03	-1.95E-04	-10	-5.39E-05	-1.09E-03	
-5	1.89E-03	-1.01E-04	-5	-5.69E-05	-5.39E-04	
0	1.93E-03	-2.79E-05	0	-3.98E-05	-2.41E-05	
5	1.97E-03	5.37E-05	5	-4.60E-05	4.99E-04	Vertical tail
10	1.96E-03	1.57E-04	10	-5.59E-05	1.04E-03	
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	
-10	9.82E-04	-6.07E-04	-10	2.86E-05	-1.80E-02	
-5	9.50E-04	2.08E-03	-5	2.19E-06	-6.47E-03	
0	9.15E-04	4.25E-03	0	-2.64E-05	4.49E-03	Stator
5	8.90E-04	6.45E-03	5	-3.96E-05	1.55E-02	
10	9.02E-04	9.49E-03	10	-6.33E-05	2.84E-02	
Sideslip-angle (deg)	Cl	Cn	Sideslip-angle (deg)	Cl	Cn	
-10	-6.29E-03	-3.88E-04	-10	-7.74E-03	-3.31E-03	
-5	-6.40E-03	-2.37E-04	-5	-7.53E-03	-1.82E-03	
0	-6.38E-03	-1.37E-04	0	-7.42E-03	-1.22E-04	
5	-6.31E-03	-1.60E-05	5	-7.50E-03	1.52E-03	
10	-6.14E-03	1.62E-04	10	-7.71E-03	3.04E-03	

Table F-22 Computational results of individual and full configuration models with vertical tail set at - 2° for various sideslip-angles