DESIGN METHODS FOR AN UNCONVENTIONAL SPLIT WING AIRCRAFT CONFIGURATION

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Abstract

Conventional aircraft typically feature low wing and low tail or T-tail design which inherently have some drawbacks. For instance, the downwash from the wing onto the horizontal tail especially with flaps deployed result in higher control power required, incurring extra drag and thus fuel consumption increase. At high angles of attack, in T-tail configurations, the horizontal tail could be completely blanketed by the turbulent wake of the wing thereby rendering the tail ineffective and potentially causing deep stall problems. Flap deployment causes nose down pitching moments which must be overcome by elevator and/or horizontal tail deflection causing the overall maximum lift to go down. The current study proposes methods to analyze unconventional split wing designs that eliminate these issues.

1 Introduction

As mentioned in the abstract, conventional aircraft have some inherent drawbacks. Figure 1 shows the three-view and an isometric view of a three surface airplane which has the advantage of that the canard (forward lifting surface) is lifting. It still has the drawback of a T-tail with its associated issues. This configuration is chosen as the baseline because of the landing gear mounted in the fuselage. This way no special consideration for mounting the landing gear is needed.

Figure 2 shows the proposed split wing design where the left and right wing are no longer connected.

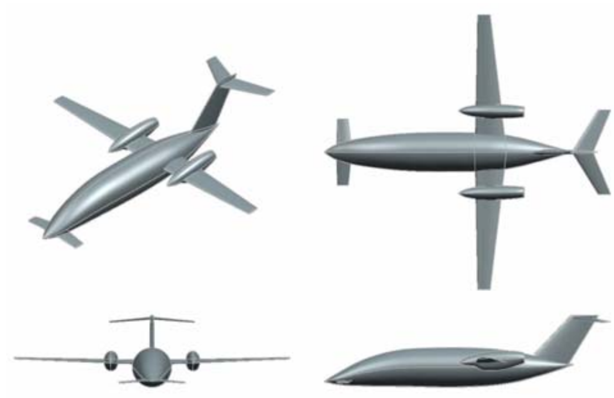

Fig. 1. Three-Surface Configuration

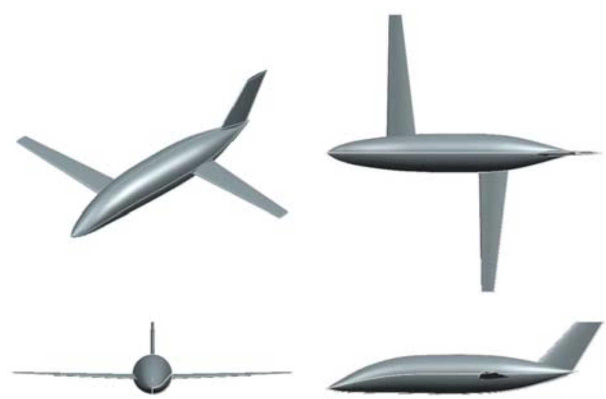

Fig. 2. Split Wing Configuration

There are several benefits to a split wing aircraft. Some of these are as listed below:

- Weight savings since the horizontal tail is eliminated. The fuselage weight will go up due to increased fuselage torque [1].

- Savings in drag which will result in better speed and range characteristics (lower fuel consumption).

The derivatives are estimated based on the so-called component build-up method. In this method, the airplane is assumed to be built up from a number of components. The total forces and moments which act on the airplane are then assumed to follow from summing the forces and moments which act on these components [2].

2 Aerodynamics

The body fixed axis system is defined as a right-hand axis system with the origin located at the airplane center of gravity, the X-axis is pointing forward, and the Y-axis pointing starboard. The stability axis system is also originated at the airplane center of gravity, and is defined by rotating the body fixed axis system through the airplane angle of attack, as shown in Figure 3. Figure 4 and Figure 5 illustrate the remaining axes definitions

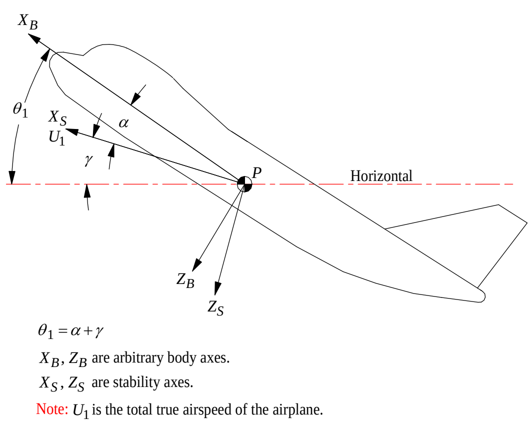

Fig. 3. Side View: Body & Stability Axis Systems

The stability and control forces, coefficients and derivatives are always defined in terms of the stability axis system. This can be seen by the lift, drag, and side forces always coinciding with the respective stability axis (see Figures 3, 4 and 5). The moments are always defined as a positive rotation about respective axis following the 'right hand rule'.

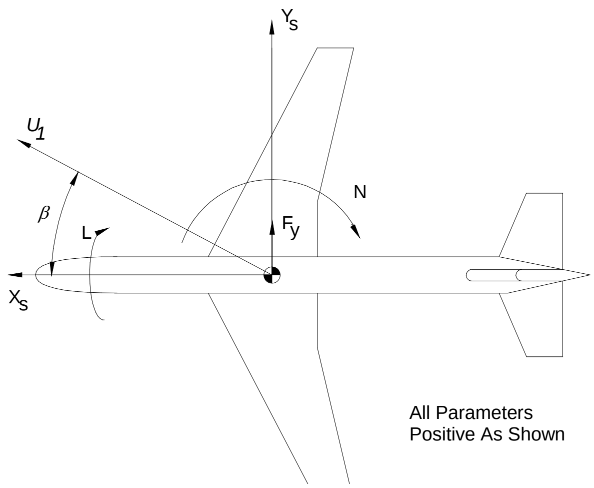

Fig. 4. Top View: Stability Axis System

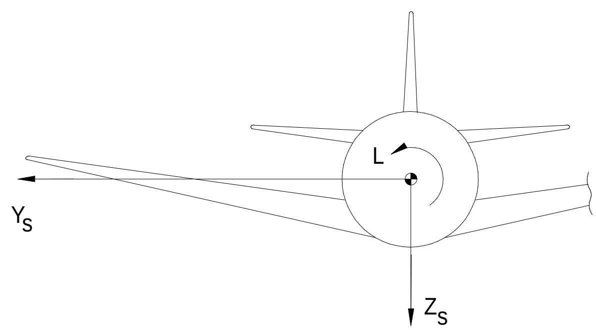

Fig. 5. Front View: Stability Axis System

Figure 6 shows the nomenclature used in the following equations.

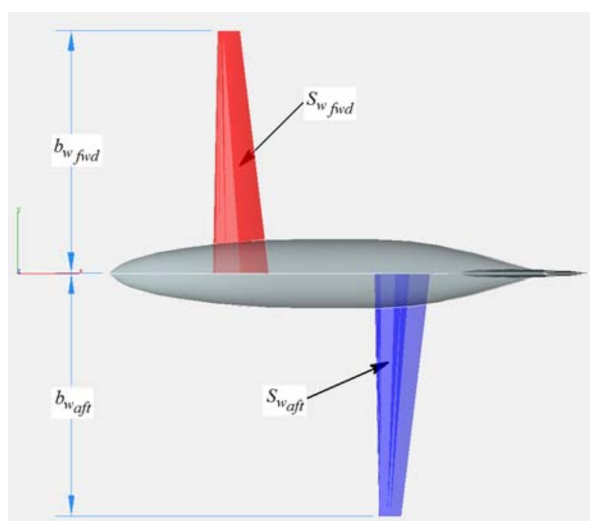

Fig. 6. Split Wing Nomenclature

For simplicity purposes, nacelles, pylons and landing gear are ignored in the equations for split wing configurations.

The aspect ratio of the split wings is calculated as shown below:

$$AR_{w_{fwd}} = \frac{b^2_{w_{fwd}}}{S_{w_{fwd}}} \quad (1)$$

$$AR_{w_{aft}} = \frac{b^2_{w_{aft}}}{S_{w_{aft}}} \quad (2)$$

Where:

$S_{w_{fwd}}$ = Area of Forward Wing
$S_{w_{aft}}$ = Area of Aft Wing
$b_{w_{fwd}}$ = Span of the Forward Wing
$b_{w_{aft}}$ = Span of the Forward Wing
$AR_{w_{fwd}}$ = Aspect Ratio of the Forward Wing
$AR_{w_{aft}}$ = Aspect Ratio of the Aft Wing

2.1 Lift

The lift coefficient for a conventional airplane with a horizontal tail is calculated as follows:

$$C_{L_1} = C_{L_0} + C_{L_\alpha}\alpha + C_{L_{\delta_{e0}}} K'_e \delta_e + C_{L_{i_h}} i_h \quad (3)$$

Where:

C_{L_0} = Airplane Zero Angle of Attack Lift Coefficient
C_{L_α} = Airplane Lift Curve Slope
α = Angle of Attack
$C_{L_{\delta_{e0}}}$ = Airplane Lift Coefficient due to Elevator Deflection Derivative at Zero Deflection
δ_e = Elevator Deflection
K'_e = Elevator Effectiveness Factor
$C_{L_{i_h}}$ = Airplane Lift Coefficient due to Horizontal Tail Incidence Derivative
i_h = Horizontal Tail Incidence

The lift coefficient for a split wing configuration is calculated as shown below:

$$\begin{aligned} C_{L_1} = {} & C_{L_{0_{wf_{fwd}}}} + C_{L_{\alpha_{wf_{fwd}}}}\alpha \\ & + C_{L_{0_{wf_{aft}}}} + C_{L_{\alpha_{wf_{aft}}}}\alpha \\ & + C_{L_{\delta_{el_{aft}}}}\delta_{el_{aft}} \end{aligned} \quad (4)$$

Where:

$C_{L_{0_{wf_{fwd}}}}$ = Forward Wing Zero-Angle of Attack Lift Coefficient
$C_{L_{0_{wf_{aft}}}}$ = Aft Wing Zero-Angle of Attack Lift Coefficient
$C_{L_{\alpha_{wf_{fwd}}}}$ = Forward Wing-Fuselage Lift Curve Slope
$C_{L_{\alpha_{wf_{aft}}}}$ = Aft Wing-Fuselage Lift Curve Slope
α = Angle of Attack

C_{L_α} = Airplane Lift Curve Slope

The airplane zero angle of attack lift coefficient for a conventional airplane with a horizontal tail is calculated as follows:

$$C_{L_0} = C_{L_{0_{wf}}} + C_{L_{0_h}} + C_{L_{0_n}} + C_{L_{0_{py}}} \quad (5)$$

Where:

$C_{L_{0_{wf}}}$ = Wing-Fuselage contribution to the Airplane Zero Angle of Attack Lift Coefficient
$C_{L_{0_h}}$ = Horizontal Tail contribution to the Airplane Zero Angle of Attack Lift Coefficient

$C_{L_{0_n}}$ = Nacelle contribution to the Airplane Zero Angle of Attack Lift Coefficient
$C_{L_{0_{py}}}$ = Pylon contribution to the Airplane Zero Angle of Attack Lift Coefficient

The airplane zero angle of attack lift coefficient for a split wing configuration is calculated as follows:

$$C_{L_0} = C_{L_{o_{wf\,fwd}}} + C_{L_{o_{wf_{aft}}}} \quad (6)$$

Where:
$C_{L_{o_{wf\,fwd}}}$ = Forward Wing-Fuselage contribution to the Airplane Zero Angle of Attack Lift Coefficient
$C_{L_{o_{wf_{aft}}}}$ = Aft Wing-Fuselage contribution to the Airplane Zero Angle of Attack Lift Coefficient

The airplane lift curve slope for a conventional airplane with a horizontal tail is calculated as shown below:

$$C_{L_\alpha} = C_{L_{\alpha_{wfn}}} + C_{L_{\alpha_h}} \quad (7)$$

Where:
$C_{L_{\alpha_{wfn}}}$ = Wing-Fuselage-Nacelle Lift Curve Slope
$C_{L_{\alpha_h}}$ = Horizontal Tail Contribution to Airplane Lift Curve Slope
C_{L_α} = Airplane Lift Curve Slope

The airplane lift curve slope for the split wing configuration is calculated as follows:

$$C_{L_\alpha} = C_{L_{\alpha_{wf\,fwd}}} + C_{L_{\alpha_{wf_{aft}}}} \quad (8)$$

Where:
$C_{L_{\alpha_{wf\,fwd}}}$ = Forward Wing-Fuselage Lift Curve Slope
$C_{L_{\alpha_{wf_{aft}}}}$ = Aft Wing-Fuselage Lift Curve Slope

The airplane lift coefficient due to elevon deflection derivative is calculated as follows:

$$C_{L_{\delta_{el_{aft}}}} = C_{L_{\delta_{el_{0_{aft}}}}} K''_{aft} \quad (9)$$

Where:
$C_{L_{\delta_{el_{0_{aft}}}}}$ = Airplane Lift-Coefficient-due-to-Aft-Elevon-Deflection Derivative at Zero Deflection
K''_{aft} = Slope at the given Elevon Deflection
$C_{L_{\delta_{el_{aft}}}}$ = Airplane Lift-Coefficient-due-to-Aft-Elevon-Deflection Derivative

2.2 Drag

The drag coefficient for a conventional airplane with a horizontal tail is calculated as shown below:

$$\begin{aligned} C_{D_1} = {} & C_{D_{0_w}} + C_{D_{L_w}} + C_{D_{0_h}} + C_{D_{L_h}} + C_{D_{0_v}} + C_{D_{L_v}} \\ & + C_{D_{0_f}} + C_{D_{L_f}} + C_{D_n} + C_{D_{py}} + C_{D_{flap}} + \\ & C_{D_{gear}} + C_{D_{wind\,shield}} + C_{D_{trim}} + C_{D_{prop}} + C_{D_{misc}} \end{aligned} \quad (10)$$

Where:
$C_{D_{0_w}}$ = Wing Zero-Lift Drag Coefficient
$C_{D_{L_w}}$ = Wing Drag Coefficient due to Lift
$C_{D_{0_h}}$ = Horizontal Tail Zero-Lift Drag Coefficient

$C_{D_{L_h}}$ = Horizontal Tail Drag Coefficient due to Lift
$C_{D_{0_v}}$ = Vertical Tail Zero-Lift Drag Coefficient
$C_{D_{L_v}}$ = Vertical Tail Drag Coefficient due to Lift
$C_{D_{0_f}}$ = Fuselage Zero-Lift Drag Coefficient
$C_{D_{L_f}}$ = Fuselage Drag Coefficient due to Lift
C_{D_n} = Nacelle Drag Coefficient
$C_{D_{py}}$ = Pylon Drag Coefficient
$C_{D_{flap}}$ = Flap Drag Coefficient
$C_{D_{gear}}$ = Gear Drag Coefficient
$C_{D_{wind\ shield}}$ = Windshield Drag Coefficient
$C_{D_{trim}}$ = Trim Drag Coefficient
$C_{D_{prop}}$ = Propeller Drag Coefficient
$C_{D_{misc}}$ = Miscellaneous Drag Coefficient
C_{D_1} = Airplane Drag Coefficient

The drag coefficient for a split wing configuration is calculated as shown below:

$$C_{D_1} = C_{D_{0_{w\,fwd}}} + C_{D_{L_{w\,fwd}}} + C_{D_{0_{waft}}} + C_{D_{L_{waft}}} + C_{D_{0_v}} + C_{D_{L_v}} + C_{D_{0_f}} + C_{D_{L_f}} \tag{11}$$

Where:

$C_{D_{0_{w\,fwd}}}$ = Forward Wing Zero-Lift Drag Coefficient
$C_{D_{0_{waft}}}$ = Aft Wing Zero-Lift Drag Coefficient
$C_{D_{L_{w\,fwd}}}$ = Forward Wing Drag Coefficient due to Lift
$C_{D_{L_{waft}}}$ = Aft Wing Drag Coefficient due to Lift
$C_{D_{0_v}}$ = Vertical Tail Zero-Lift Drag Coefficient
$C_{D_{L_v}}$ = Vertical Tail Drag Coefficient due to Lift
$C_{D_{0_f}}$ = Fuselage Zero-Lift Drag Coefficient
$C_{D_{L_f}}$ = Fuselage Drag Coefficient due to Lift
C_{D_1} = Airplane Drag Coefficient

2.3 Pitching Moment

The pitching moment coefficient for a conventional airplane with a horizontal tail is calculated as shown below:

$$C_{m_1} = C_{m_0} + C_{m_\alpha}\alpha + C_{m_{\delta_{e0}}} K'_e \delta_e + C_{m_{i_h}} i_h \tag{12}$$

Where:

C_{m_0} = Airplane Zero Angle of Attack Pitching Moment Coefficient
C_{m_α} = Airplane Pitching Moment Curve Slope
α = Angle of Attack
$C_{m_{\delta_{e0}}}$ = Airplane Pitching Moment Coefficient due to Elevator Deflection Derivative at Zero Deflection
δ_e = Elevator Deflection
K'_e = Elevator Effectiveness Factor
$C_{m_{i_h}}$ = Airplane Pitching Moment Coefficient due to Horizontal Tail Incidence Derivative
i_h = Horizontal Tail Incidence

The pitching moment coefficient for a split wing configuration is calculated as shown below:

$$C_{m_1} = C_{m_{o_{wf\,fwd}}} + C_{m_{\alpha_{wf\,fwd}}}\alpha + C_{m_{o_{wf\,aft}}} + C_{m_{\alpha_{wf\,aft}}}\alpha + C_{m_{\delta_{el_{aft}}}}\delta_{el_{aft}} \quad (13)$$

Where:

$C_{m_{o_{wf\,fwd}}}$ = Forward Wing-Fuselage contribution to the Airplane Zero Angle of Attack Pitching Moment Coefficient

$C_{m_{o_{wf\,aft}}}$ = Aft Wing-Fuselage contribution to the Airplane Zero Angle of Attack Pitching Moment Coefficient

$C_{m_{\alpha_{wf\,fwd}}}$ = Forward Wing-Fuselage Pitching Moment Curve Slope

$C_{m_{\alpha_{wf\,aft}}}$ = Aft Wing-Fuselage Pitching Moment Curve Slope

α = Angle of Attack

The pitching moment coefficient at zero angle of attack for a conventional airplane with a horizontal tail is calculated as shown below:

$$C_{m_0} = C_{m_{0_{wf}}} + C_{m_{0_h}} + C_{m_{0_n}} + C_{m_{0_{py}}} \quad (14)$$

Where:

$C_{m_{0_{wf}}}$ = Wing-Fuselage contribution to the Airplane Zero Angle of Attack Pitching Moment Coefficient

$C_{m_{0_h}}$ = Horizontal Tail contribution to the Airplane Zero Angle of Attack Pitching Moment Coefficient

$C_{m_{0_n}}$ = Nacelle contribution to the Airplane Zero Angle of Attack Pitching Moment Coefficient

$C_{m_{0_{py}}}$ = Pylon contribution to the Airplane Zero Angle of Attack Pitching Moment Coefficient

The airplane zero angle of attack pitching moment coefficient for a split wing configuration is calculated as follows:

$$C_{m_0} = C_{m_{o_{wf\,fwd}}} + C_{m_{o_{wf\,aft}}} \quad (15)$$

Where:

$C_{m_{o_{wf\,fwd}}}$ = Forward Wing-Fuselage contribution to the Airplane Zero Angle of Attack Pitching Moment Coefficient

$C_{m_{o_{wf\,aft}}}$ = Aft Wing-Fuselage contribution to the Airplane Zero Angle of Attack Pitching Moment Coefficient

The airplane pitching moment curve slope for a conventional airplane with a horizontal tail is calculated as shown below:

$$C_{m_\alpha} = C_{m_{\alpha_{wf}}} + C_{m_{\alpha_h}} + C_{m_{\alpha_n}} + C_{m_{\alpha_{py}}} \quad (16)$$

Where:

$C_{m_{\alpha_{wf}}}$ = Wing-Fuselage Pitching Moment Curve Slope

$C_{m_{\alpha_h}}$ = Horizontal Tail Contribution to Airplane Pitching Moment Curve Slope

$C_{m_{\alpha_n}}$ = Nacelle Contribution to Airplane Pitching Moment Curve Slope

$C_{m_{\alpha_{py}}}$ = Pylon Contribution to Airplane Pitching Moment Curve Slope

The airplane pitching moment curve slope for the split wing configuration is calculated as follows:

$$C_{m_\alpha} = C_{m_{\alpha_{wf\,fwd}}} + C_{m_{\alpha_{wf_{aft}}}} \qquad (17)$$

Where:

$C_{m_{\alpha_{wf\,fwd}}}$ = Forward Wing-Fuselage Pitching Moment Curve Slope

$C_{m_{\alpha_{wf_{aft}}}}$ = Aft Wing-Fuselage Pitching Moment Curve Slope

The airplane pitching moment coefficient due to elevon deflection derivative is calculated as follows:

$$C_{m_{\delta_{el_{aft}}}} = C_{m_{\delta_{el_{0_{aft}}}}} K''_{aft} \qquad (18)$$

Where:

$C_{m_{\delta_{el_{0_{aft}}}}}$ = Airplane Pitching-Moment-Coefficient-due-to-Aft-Elevon-Deflection Derivative at Zero Deflection

K''_{aft} = Slope at the given Elevon Deflection

$C_{m_{\delta_{el_{aft}}}}$ = Airplane Pitching-Moment - Coefficient-due-to-Aft-Elevon-Deflection Derivative

2.4 Sideforce

The sideforce coefficient due to sideslip derivative for a conventional airplane with a horizontal tail is calculated as shown below:

$$C_{y_\beta} = C_{y_{\beta_w}} + C_{y_{\beta_f}} + C_{y_{\beta_v}} + C_{y_{\beta_{py}}} \qquad (19)$$

Where:

$C_{y_{\beta_w}}$ = Wing Contribution to the Airplane Sideforce-Coefficient-due-to-Sideslip Derivative

$C_{y_{\beta_f}}$ = Fuselage Contribution to Airplane Sideforce-Coefficient-due-to-Sideslip Derivative

$C_{y_{\beta_v}}$ = Vertical Tail Contribution to Airplane Sideforce-Coefficient-due-to-Sideslip Derivative

$C_{y_{\beta_{py}}}$ = Airplane Sideforce-Coefficient-due-to-Sideslip

The sideforce coefficient due to sideslip derivative for a split wing configuration is calculated as shown below:

$$C_{y_\beta} = C_{y_{\beta_{w\,fwd}}} + C_{y_{\beta_{w_{aft}}}} + C_{y_{\beta_v}} \qquad (20)$$

Where:

$C_{y_{\beta_{w\,fwd}}}$ = Forward Wing Contribution to Airplane Sideforce-Coefficient-due-to-Sideslip Derivative

$C_{y_{\beta_{w_{aft}}}}$ = Aft Wing Contribution to Airplane Sideforce-Coefficient-due-to-Sideslip Derivative

$C_{y_{\beta_v}}$ = Vertical Tail Contribution to Airplane Sideforce-Coefficient-due-to-Sideslip Derivative

C_{y_β} = Airplane Sideforce-Coefficient-due-to-Sideslip

2.5 Rolling Moment

The rolling moment coefficient due to sideslip derivative for a conventional airplane with a horizontal tail is calculated as shown below:

$$C_{l_\beta} = C_{l_{\beta_{wf}}} + C_{l_{\beta_h}} + C_{l_{\beta_v}} \qquad (21)$$

Where:

$C_{l_{\beta_{wf}}}$ = Wing-Fuselage Contribution to the Airplane Rolling Moment-Coefficient-due-to-Sideslip Derivative

$C_{l_{\beta_h}}$ = Horizontal Tail Contribution to Airplane Rolling Moment - Coefficient-due-to-Sideslip Derivative

$C_{l_{\beta_v}}$ = Vertical Tail Contribution to Airplane Rolling Moment - Coefficient-due-to-Sideslip Derivative

The rolling moment coefficient due to sideslip derivative for a split wing configuration is calculated as shown below:

$$C_{l_\beta} = C_{l_{\beta_{wf\,fwd}}} + C_{l_{\beta_{wf_{aft}}}} + C_{l_{\beta_v}} \quad (22)$$

Where:

$C_{l_{\beta_{wf\,fwd}}}$ = Forward Wing-Fuselage Contribution to Airplane Rolling-Moment-Coefficient-due-to-Sideslip Derivative

$C_{l_{\beta_{wf_{aft}}}}$ = Aft Wing-Fuselage Contribution to Airplane Rolling-Moment-Coefficient-due-to-Sideslip Derivative

$C_{l_{\beta_v}}$ = Vertical Tail Contribution to Airplane Rolling-Moment-Coefficient-due-to-Sideslip Derivative

C_{l_β} = Airplane Rolling-Moment-Coefficient-due-to-Sideslip

3 CFD Investigations

To compare the semi empirical methods with physics based methods, a Computational Fluid Dynamics (CFD) analysis is performed. CFD simulations are carried out using RNG turbulence model with nodes ranging from 250,000 to 1 million [3].

Fuselage only, Fuselage and Symmetric Wing and Fuselage and Split Wing cases were analyzed at angles of attack of 2.0 and 5.0 degrees. The steady state flight speed is set to 150 kts at an altitude of 10,000 ft.

Figure 7 shows the static pressure contours on a conventional symmetric wing aircraft.

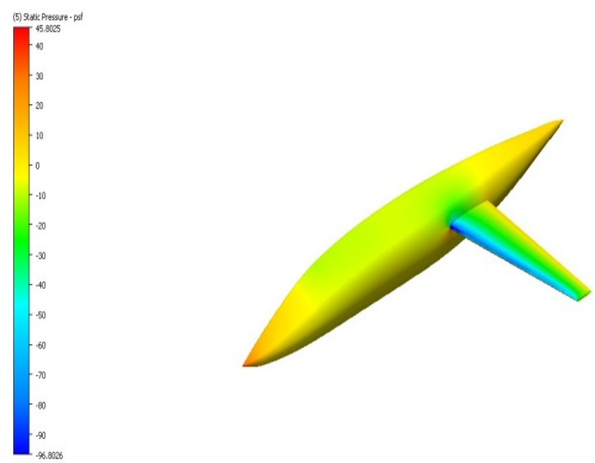

Fig. 7. Static Pressure Contours: Symmetric Wing Aircraft

Figure 8 shows the static pressure contours on a split wing aircraft.

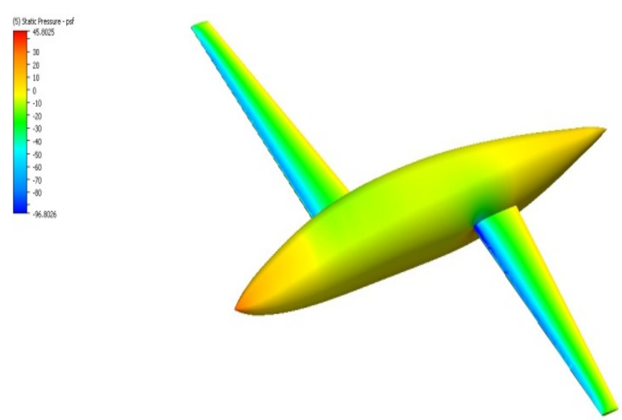

Fig. 8. Static Pressure Contours: Split Wing Aircraft

Table 1 shows the summary of the airplane lift curve slope from CFD and Component Build Up methods.

Table 1. C_{L_α} Comparison in deg^{-1}

Component	CFD	Component Build Up Method
Fuselage	0.0005	0.0005
Fuselage + Symmetric Wing	0.0122	0.0110
Fuselage + Split Wing	0.0120	0.0105

Table 2 shows the summary of the airplane pitching moment curve slope from CFD and Component Build Up methods.

Table 2. C_{m_α} Comparison in deg^{-1}

Component	CFD	Component Build Up Method
Fuselage	0.0144	0.0144
Fuselage + Symmetric Wing	0.0172	0.0228
Fuselage + Split Wing	0.0573	0.0366

It is seen from Table 1 that the variation in the Component Build Up method from CFD is around 10% to 12% in the calculation of C_{L_α} for the model with symmetric wing and split wing respectively.

It is also seen from Table 2 that the variation in the Component Build Up method from CFD is 33% to 36% in the calculation of C_{m_α} for the model with symmetric wing and split wing respectively.

The CFD cases are currently being analyzed in detail and more accurate empirical methods are being developed to determine the downwash from the wing and sidewash from the fuselage.

4 Controllability & Trim

A split wing aircraft has an inherent pitch-roll coupling. Whenever the elevons on the aft wing are deployed to pitch the airplane up or down, the airplane will tend to roll because of the asymmetric lift from the aft wing. This will have to be compensated by a simultaneous aileron deflection on the forward wing. This configuration has to be controlled by a fly-by-wire system [4].

Figure 9 shows the top view of a split wing airplane with ailerons, elevons and flaps.

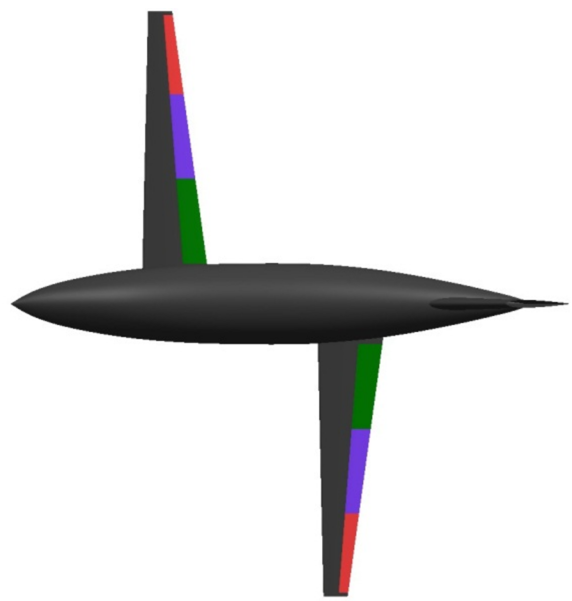

Fig. 9. Split Wing Control Surfaces

The rolling moment caused because of the asymmetric lift when the aft elevons are deployed is calculated as shown below:

$$\pounds = C_{l_{\delta_{el0_{aft}}}} K'_{el_{aft}} \delta_{el_{aft}} \bar{q} S_{w_{aft}} b_{w_{aft}} \quad (23)$$

Where:

$C_{l_{\delta_{el0_{aft}}}}$ = Airplane Rolling-Moment-Coefficient-due-to-Aft Elevon-Deflection Derivative at Zero Deflection

$K'_{el_{aft}}$ = Aft Wing Elevon Effectiveness Factor

$S_{w_{aft}}$ = Area of Aft Wing

$\delta_{el_{aft}}$ = Aft Wing Elevon Deflection

$b_{w_{aft}}$ = Aft Wing Span

$\overline{q}$ = Dynamic Pressure

£= Airplane Rolling Moment

This rolling moment is compensated by an aileron deflection on the forward wing as follows:

$$\delta_{a_{fwd}} = \delta_{el_{aft}} \left(\frac{C_{l_{\delta_{el0_{aft}}}}}{C_{l_{\delta_{a0_{fwd}}}}} \right) \left(\frac{K'_{el_{aft}}}{K'_{a_{fwd}}} \right) \left(\frac{b_{w_{aft}}}{b_{w_{fwd}}} \right) \left(\frac{S_{w_{aft}}}{S_{w_{fwd}}} \right) \quad (24)$$

Where:

$C_{l_{\delta_{el0_{aft}}}}$ = Airplane Rolling-Moment-Coefficient-due-to-Aft Elevon-Deflection Derivative at Zero Deflection

$K'_{el_{aft}}$ = Aft Wing Elevon Effectiveness Factor

$S_{w_{aft}}$ = Area of Aft Wing

$b_{w_{aft}}$ = Aft Wing Span

$\delta_{el_{aft}}$ = Aft Wing Elevon Deflection

$C_{l_{\delta_{a0_{fwd}}}}$ = Airplane Rolling-Moment-Coefficient-due-to-Aileron-Deflection Derivative at Zero Deflection

$b_{w_{fwd}}$ = Forward Wing Span

$S_{w_{fwd}}$ = Area of Forward Wing

$K'_{a_{fwd}}$ = Aileron Effectiveness Factor

$\delta_{a_{fwd}}$ = Aileron Deflection Angle

The elevon deflection angle for trimmed lift condition is found from the following equation:

$$\delta_{el_{aft}} = \frac{-\left(C_{m_0} + C_{m_\alpha} \alpha \right)}{C_{m_{\delta_{el0_{aft}}}} \; K'_{el_{aft}}} \quad (25)$$

Where:

C_{m_0} = Airplane Zero-Lift-Pitching-Moment-Coefficient

C_{m_α} = Airplane Pitching Moment Curve Slope

$C_{m_{\delta_{el0_{aft}}}}$ = Airplane Pitching-Moment-Coefficient-due-to-Aft-Elevon-Deflection Derivative at Zero Deflection

$K'_{el_{aft}}$ = Aft Elevon Effectiveness Factor

α = Angle of Attack

$\delta_{el_{aft}}$ = Aft Elevon Deflection Angle

Figure 10 shows an isometric view of a pitch down maneuver on a split wing airplane with flaps retracted.

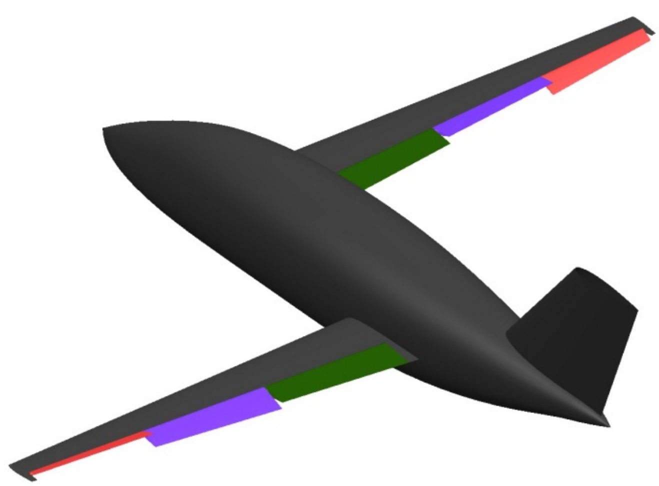

Fig. 10. Pitch Down Maneuver, Flaps Up

Figure 11 shows an isometric view of a pitch up maneuver on a split wing airplane with flaps retracted.

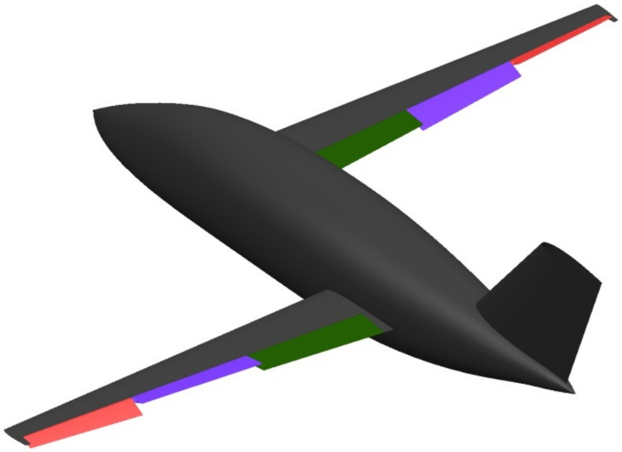

Fig. 11. Pitch Up Maneuver, Flaps Up

Figure 12 shows an isometric view of a pitch down maneuver on a split wing airplane with flaps deployed.

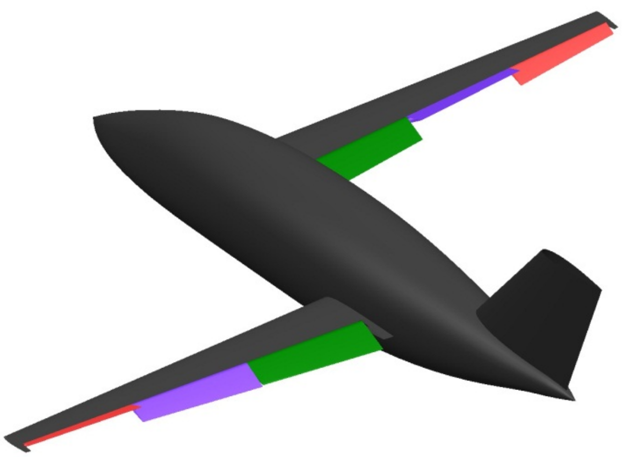

Fig. 12. Pitch Down Maneuver, Flaps Down

Figure 13 shows an isometric view of a pitch up maneuver on a split wing airplane with flaps deployed.

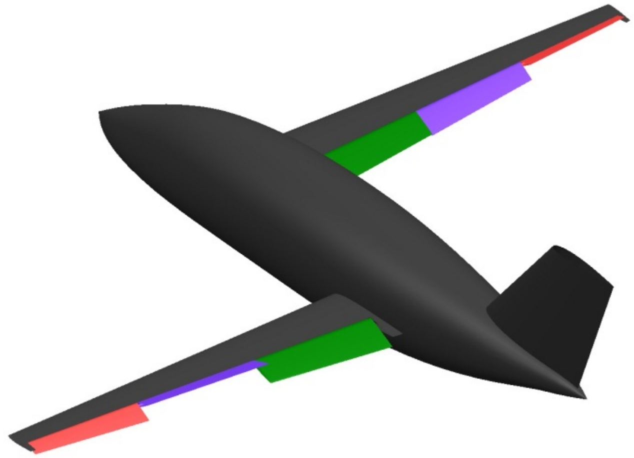

Fig. 13. Pitch Up Maneuver, Flaps Down

In a roll maneuver, an aileron deflection will cause the airplane to pitch up or down. Another important consideration that is being investigated is the need for differential aileron deflections on a split wing configuration.

5 Weight Investigation

Buckling is identified to be the most critical governing factor in the weight determination of a split wing aircraft [1]. In this reference weight estimations are carried out for a passenger type airliner. It is seen that for the passenger airliner, the fuselage weight increased significantly. Typically General Aviation (GA) type airplanes are over designed from a structural weight stand point because of minimum gauge thickness. A split wing concept on a GA type airplane could have significant weight savings. Because of the over design factor, a split wing GA aircraft could have weight savings because of the elimination of the horizontal tail. This will be further investigated.

6 Landing Gear Integration

For a Split Wing Aircraft, it is necessary to locate the landing gear on the fuselage. This will eliminate pitch over issues associated with an asymmetric landing gear configuration.

7 Take-off Rotation

As already mentioned in the Controllability & Trim Section, whenever the elevons on the aft wing are deployed, they have to be compensated by an aileron deflection on the forward wing.

Figure 14 shows the nomenclature associated with the equations for take-off rotation. The following equations govern the airplane equilibrium at the instant of rotation:

$$T - D - \mu_g R_g = \frac{W}{g}\dot{U} \tag{26}$$

Where:

T = Thrust
D = Drag
μ_g =Wheel-to-Ground Friction Coefficient
R_g = Ground Reaction
W = Aircraft Weight
g = Acceleration due to Gravity
$\dot{U}$ = Aircraft Acceleration

$$L_{wf\,fwd} + L_{wf_{aft}} + R_g = W \tag{27}$$

Where:

$L_{wf\,fwd}$ = Lift on Forward Wing
$L_{wf_{aft}}$ = Lift on Aft Wing
R_g = Ground Reaction
W = Aircraft Weight

$$\begin{aligned} &-W\left(X_{mg} - X_{cg}\right) + D\left(Z_D - Z_{mg}\right) \\ &-T\left(Z_T - Z_{mg}\right) + L_{wf\,fwd\,wing}\left(X_{mg} - X_{ac_{wf\,fwd}}\right) \\ &+M_{ac_{wf\,fwd}} + L_{wf_{aft}}\left(X_{ac_{wf_{aft}}} - X_{mg}\right) \\ &+\frac{W}{g}\dot{U}\left(Z_{cg} - Z_{mg}\right) = I_{yy_{mg}}\ddot{\Theta}_{mg} \end{aligned} \tag{28}$$

Where:

$L_{wf\,fwd}$ = Lift on Forward Wing
$L_{wf_{aft}}$ = Lift on Aft Wing
W = Aircraft Weight
g = Acceleration due to Gravity
$\dot{U}$ = Aircraft Acceleration
T = Thrust
D = Drag
$I_{yy_{mg}}$ = Airplane Moment of Inertia in Y-axis about the Main Gear
$\ddot{\Theta}_{mg}$ = Steady State Pitch Attitude Acceleration around Main Gear
X_{mg} = X-location of the Main Gear
Z_{mg} = Z-location of the Main Gear
X_{cg} = X-location of the Airplane Center of Gravity
Z_T = Z-location of the Thrust Vector Origin
Z_D = Z-location of the Drag Vector Origin
$X_{ac_{wf\,fwd}}$ = X-location of the Forward Wing-Fuselage Aerodynamic Center
$X_{ac_{wf_{aft}}}$ = X-location of the Aft Wing-Fuselage Aerodynamic Center
$M_{ac_{wf\,fwd}}$ = Forward Wing-Fuselage Pitching Moment

These equations are solved in ground effect to obtain a minimum area for the aft wing to initiate take-off rotation.

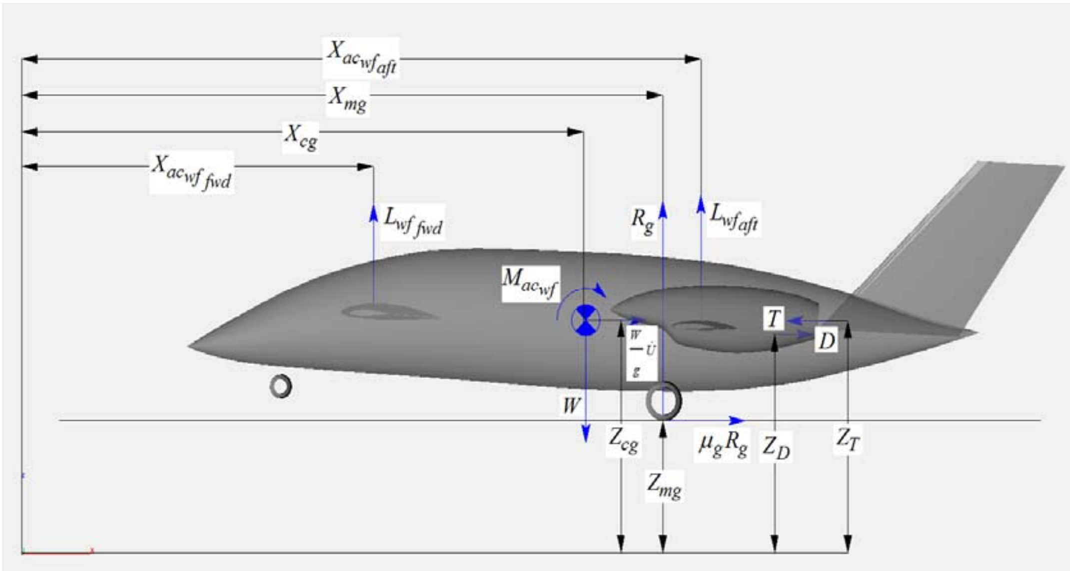

Fig. 14. Take-off Rotation Nomenclature

8 Conclusions

An attempt has been made to understand the aerodynamics and stability & control of split wing aircraft. Methods are currently being implemented in the Advanced Aircraft Analysis (AAA) software [5]. A component build-up technique is used to derive equations for the stability & control derivatives. A detailed study where these methods are applied on different size aircraft is underway.

References

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