

Wake Disturbance-Based Separation Evaluation for Urban Air Mobility Aircraft

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This paper builds upon previous research to evaluate Urban Air Mobility (UAM) flight dynamics in the presence of wake disturbances. A model for fixed-wing-generated wake effects is applied to a Revolutionary Vertical Lift Technology (RVLT) Lift+Cruise UAM trail aircraft operating in the fixed-wing Cruise configuration. The wake effects are evaluated at various longitudinal and latitudinal separation configurations to determine the most hazardous wake encounter situation. Trim calculations and closed-loop simulations show a trailing aircraft flying directly into a wake vortex as the most hazardous scenario. Furthermore, trailing behind a larger aircraft at similar speeds greatly increases the safe separation distance. These considerations indicate a need for dynamic pairwise separation criteria to enable dense Advanced Air Mobility (AAM) operations while maintaining safety.

I. Introduction

To match the growing transportation needs in densely populated areas, the vision of air taxiing people and cargo comes closer to reality. Concepts such as Urban Air Mobility (UAM) have seen much research in both aircraft and operation design.¹ These UAM aircraft often have vertical take-off and landing (VTOL) capabilities for air taxiing passengers in dense urban environments, with some level of autonomy onboard. To enable UAM operations over highly populated areas such as urban cities, the airspace must be able to support more flights at altitudes lower than typical commercial aircraft. NASA defines the UAM Maturity Level (UML)² to categorize the UAM vehicles, airspace, and community status. These levels begin at the pre-operational UML-1, which serves as the initial state with testing and demonstrations in limited environments. By UML-3, airspace management should provide some form of separation services to enable dense operations around vertiports. UML-6 is the most mature level and enables “ubiquitous UAM operations with system-wide automated optimization”.

Vascik et Hansman³ expect current Air Traffic Control (ATC) procedures to inadequately scale to the number of predicted UAM aircraft. The 8 nautical mile longitudinal separation between Instrument Flight

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Rules (IFR) aircraft would be opposing to a dense operational environment. Goodrich et Theodore² also propose a reduced separation minima to enable dense operations at variably distributed vertiports over populated areas in UML-6. While much research has been conducted on formulating new separation criteria, most of these studies focus on the airspace operational and technological solutions.³⁻⁶ This study focuses on the inter-UAM wake interactions that affect aircraft flight dynamics. By using a physics-based approach to testing safety, we aim to evaluate wake-based separation minima that can be scaled to varying UML levels.

A. FAA Standards

A wake-generating aircraft's vortex strength increases proportionally with its operating weight and inversely with its airspeed. Furthermore, an aircraft's ability to handle a wake encounter primarily depends on the aircraft's wingspan and counter control response. A longer wingspan enables higher wake tolerance for incoming aircraft. Therefore, the U.S. Department of Transportation Federal Aviation Administration (FAA) designates wake separation criteria based on the aircraft classifications in Table 1 and the lookup matrix in Table 2.

Table 1: FAA aircraft wake classes.⁷

Class	Takeoff Weight
Super	Above Heavy (e.g. Airbus A380 AN225)
Heavy	> 300,000 lbs
Large	> 41,000 lbs & < 300,000 lbs
Small	< 41,000 lbs

Table 2: FAA wake separation standards at the threshold (MRS=Minimum Radar Separation).⁷

		Follower				
		Super	Heavy	B757	Large	Small
Leader	Super	MRS	6	7	7	8
	Heavy	MRS	4	5	5	6
	B757	MRS	4	4	4	5
	Large	MRS	MRS	MRS	MRS	4
	Small	MRS	MRS	MRS	MRS	MRS

After decades of employing the standards shown in Tables 1 and 2, the FAA has approved a recategorization (RECAT) of the wake separation minimum. The main motivation of RECAT is to move away from the conservative separation minimum and increase operation throughput. RECAT will consist of three phases.⁸ Phase I recategorizes the aircraft classes shown in Table 1 into six classes based on weight, approach speed, wing characteristics, and roll counter response.⁷ Phase II moves away from aircraft classes entirely and instead defines pairwise wake separation minimum that depend on the specific leading and trailing aircraft. Phase III defines dynamics pairwise separation that consider atmospheric and aircraft data to set the minimum based on situational conditions.

A Phase 1.5 recategorization was released under Order JO 7110.126B⁹ in 2021 with additional aircraft categories along with corresponding separation minimum, shown in Table 4. Category I is defined as Lower Small aircraft with a maximum takeoff weight of 15,400 pounds or less, and is where we expect most UAM aircraft to be categorized. Table 3 shows the most common Category I aircraft currently in operation. It is important to note that Table 4 shows there is no set minimum for lead aircraft that fall under Categories F through I. Therefore, there is an opportunity to define safe separation standards for UAM operations.

Table 3: RECAT Phase 1.5 Category I (Lower Small) aircraft.⁹ Aircraft specifications from <https://skybrary.aero/>.

Aircraft	Full Name	MTOW (lbs)	Wingspan (feet)	Cruise Speed (kts)
BE10	BEECH 100 King Air	9,876.7	45.90	226
BE58	BEECH 58 Baron	5,099.3	37.73	180
C208	CESSNA 208 Grand Caravan	8,598.0	52.17	184
C210	CESSNA 210 Centurion	4,012.4	36.75	170
C25A	CESSNA 525A Citation CJ2	12,125.4	49.54	384
C25B	C441 CESSNA 441 Conquest 2	9,700.3	49.21	210
C550	CESSNA 550 Citation 2	13,227.7	52.17	384
PAY2	PIPER Cheyenne 2	9,259.4	42.65	275
PA31	PIPER PA-31	6,503.6	40.71	210
PC12	PILATUS PC-12 Eagle	9,700.3	53.15	232
SR22	CIRRUS SR-22	3,306.9	38.30	180
SW3	SWEARINGEN Merlin	12,566.4	46.26	260

Table 4: Wake turbulence separation (nautical miles) for an aircraft trailing directly behind a leading aircraft.⁹

		Follower								
		A	B	C	D	E	F	G	H	I
Leader	A		5	6	6	7	7	7	8	8
	B		3	4	4	5	5	5	5	5
	C					3.5	3.5	3.5	5	5
	D		3	4	4	5	5	5	5	5
	E									4
	F									
	G									
	H									
	I									

B. Contribution

This research is part of the System-Wide Safety (SWS) project under NASA’s Aeronautics Research Mission Directorate (ARMD) that is developing predictive and mitigative solutions to risks and hazards for novel small Unmanned Aircraft Systems (sUAS) and UAM operations,¹⁰ and builds upon previous work^{11,12} studying how UAM-generated wake vortices affect safety. Completed work includes preliminary simulations on the linearized flight dynamics of a Revolutionary Vertical Lift Technology (RVLT) Lift+Cruise aircraft trailing behind a fixed-wing wake-generating aircraft. The fixed-wing wake model and linearized RVLT Lift+Cruise dynamics model were previously developed by Baculi et al.¹² This study presents separation criteria evaluations based on the wake effects on the RVLT Lift+Cruise. We calculated trim conditions for various longitudinal and lateral separation configurations due to a wake-generating RVLT Lift+Cruise lead aircraft, as well as a Swearingen Merlin lead. To validate the trim solutions, we conducted closed-loop simulations of the RVLT Lift+Cruise linear dynamics in response to constant wake inputs.

II. Wake Vortex Model

A. Fixed-Wing Wake Model

To compute the wake generated by a leading aircraft, we utilize the fixed-wing wake model developed by Baculi et al.¹² Figure 1 shows the wake vortex V_θ centered on the aircraft wingtip, with the core radius r_c indicating where V_θ is the strongest. A key parameter for understanding the peak tangential vortex velocity is the initial circulation strength Γ_0 shown in Equation (1) which indicates that an aircraft’s induced wake is dependent on its lift L , airspeed V , wingspan b , and the air density ρ .

$$\Gamma_0 = \frac{2C_L V S}{\pi b} = \frac{4L}{\pi \rho V b} \quad (1)$$

Baculi et al.¹² incorporate the initial vortex strength with a Proctor decay and Boeing propagation model to calculate the circulation strength

$$\Gamma(t, \Delta x) = \Gamma_p e^{-\alpha \frac{t \Gamma_p}{2\pi b_0^2}} \quad (2)$$

where

$$\Gamma_p = \Gamma_0 \left(\frac{a_1 x + b_1}{1 + e^{k(x-n)}} + \frac{x^{a_2} e^{b_2}}{1 + e^{-k(x-n)}} \right) \quad (3)$$

and $\alpha = 0.04887$, $b_0 = \frac{\pi}{4}b$, $x = \Delta x/b$, $a_1 = 2.298 * 10^{-17}$, $b_1 = 1$, $a_2 = -1.002$, and $b_2 = 3.501$.

Equation (4) adds the Proctor decay and Boeing propagation model to a Lamb wake aging model for the complete wake vortex model. Here, the circulation strength calculated by $\Gamma(t, \Delta x)$ is adjusted based on the radial distance r from the wingtip and the time t since the wake’s creation.

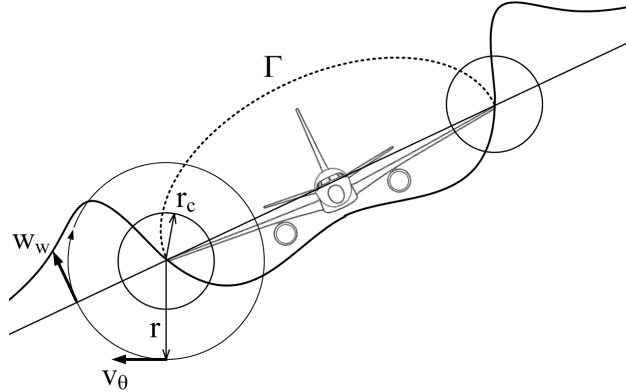


Figure 1: Wake vortex generated by a fixed-wing aircraft.

$$w_w = \begin{cases} \frac{\Gamma(t, \Delta x)}{2\pi r} \left[1 - e^{-10 \left(\frac{1.4r_c}{b} \right)^{0.75}} \right] \left[\frac{1 - e^{-\alpha_p \left(\frac{r}{r_c} \right)^2}}{1 - e^{-\alpha_p (1.4)^2}} \right] \left(1 - e^{-\frac{r^2}{4\nu t}} \right), & r \leq 1.4r_c \\ \frac{\Gamma(t, \Delta x)}{2\pi r} \left[1 - e^{-10 \left(\frac{|r|}{b} \right)^{0.75}} \right] \left(1 - e^{-\frac{r^2}{4\nu t}} \right), & r > 1.4r_c \end{cases} \quad (4)$$

The vortex core can be calculated as

$$r_c(t) = \sqrt{r_c^2(0) + r\alpha_L\nu t} \quad (5)$$

where $\alpha_L = 1.25643$ and ν is the atmospheric kinematic viscosity. The initial core radius is taken to be $r_c(0) = 0.014b$ based on Boeing 757 measurements.¹³

The wake tangential velocity calculated by Equation (4) can be transformed to a point on the inertia frame using the wake-generating aircraft's attitude $\{\phi, \theta, \psi\}$ and the trailing aircraft's down Δz , and right Δy distances:

$$U_w = \frac{w_w}{r} \left[-\Delta z(-\cos \phi \sin \psi + \sin \phi \sin \theta \cos \psi) + (\Delta y \mp \frac{b}{2})(\sin \phi \sin \psi + \cos \phi \sin \theta \cos \psi) \right] \quad (6)$$

$$V_w = \frac{w_w}{r} \left[-\Delta z(\cos \phi \cos \psi + \sin \phi \sin \theta \sin \psi) + (\Delta y \mp \frac{b}{2})(-\sin \phi \cos \psi + \cos \phi \sin \theta \sin \psi) \right] \quad (7)$$

$$W_w = \frac{w_w}{r} \left[-\Delta z \sin \phi \cos \theta + (\Delta y - \mp \frac{b}{2}) \cos \phi \cos \theta \right] \quad (8)$$

where the operand $\mp$ corresponds to the left and right wing tips, respectively. The total wake vortex effect at a specified point can be calculated by using Equations (6)–(8) to determine the wake generated by each wingtip and summing each contribution:

$$\begin{aligned} U_{w,tot} &= U_{w,L} + U_{w,R} \\ V_{w,tot} &= V_{w,L} + V_{w,R} \\ W_{w,tot} &= W_{w,L} + W_{w,R} \end{aligned} \quad (9)$$

III. Trailing Aircraft Model

To observe the wake effects on aircraft flight dynamics, we model the Revolutionary Vertical Lift Technology (RVLT) Lift+Cruise¹⁴ eVTOL aircraft in the Cruise configuration. Figure 2 shows the RVLT Lift+Cruise in this configuration where the vertical lift rotors are aligned with the aircraft's x -axis, thus enabling forward flight as a fixed-wing. The wake effects on the RVLT Lift+Cruise dynamics are calculated by applying the downwash from Equation (8) on the wing and horizontal tail three-quarter chord and the sidewash from Equation (7) on the vertical tail three-quarter chord. Table 5 shows the RVLT Lift+Cruise characteristics that are relevant to the trailing aircraft model.

A. Fixed-Wing Equations of Motion

For the RVLT Lift+Cruise aircraft in forward flight, we employ the standard 6DOF equations of motion for a fixed-wing. The angular body rate derivatives are given by Equations (10), (11), and (12):

$$I_{xx}\dot{p} - I_{xy}\dot{q} - I_{xz}\dot{r} - I_{xz}pq + I_{xy}pr + (I_{zz} - I_{yy})qr + I_{yz}(r^2 - q^2) = l \quad (10)$$

$$-I_{xy}\dot{p} + I_{yy}\dot{q} - I_{yz}\dot{r} + I_{yz}pq - I_{xy}qr + (I_{xx} - I_{zz})pr + I_{xz}(p^2 - r^2) = m \quad (11)$$

$$-I_{xz}\dot{p} - I_{yz}\dot{q} + I_{zz}\dot{r} - I_{yz}pr + I_{xz}qr + (I_{yy} - I_{xx})pq + I_{xy}(q^2 - p^2) = n \quad (12)$$

l , m , and n correspond to the aerodynamic rolling, pitching, and yawing moments, respectively. I_{xx} , I_{yy} , I_{zz} , I_{xz} , and I_{yz} are the aircraft inertias. For symmetric aircraft, $I_{xy} = I_{yz} = 0$ and Equations (10)–(12) can

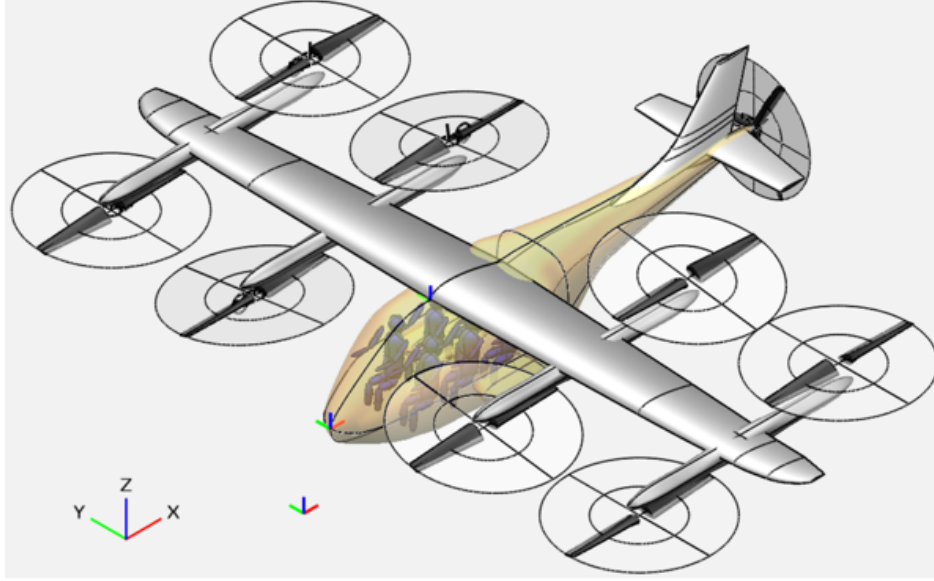


Figure 2: Revolutionary Vertical Lift Technology (RVLT) Lift+Cruise aircraft in cruise mode with the vertical lift rotors aligned with the aircraft's x -axis.¹⁴

Table 5: Physical parameters for the RVLT Lift+Cruise aircraft in forward flight.

Aircraft Parameter	Variable	Value
Weight	W	8395 lbs
Surface Area	S	232.9 ft ²
Mean Chord	$\bar{c}$	5.098 ft
Wingspan	b	47.69 ft
Trim Angle of Attack	$\bar{\alpha}$	4.1885°
Trim Sideslip Angle	$\bar{\beta}$	0.45884°
Altitude	h	4000 ft
Airspeed	V	200 ft/s

be simplified into a decoupled system of equations. However, the RVLT Lift+Cruise has some asymmetry, therefore we cannot make this simplification and Equations (10)–(12) are used for the nonlinear dynamics.

The stability frame states are given by

$$\begin{aligned}
 MV \cos \beta \dot{\alpha} = & -L - T \sin \alpha + Mg(\cos \theta \cos \phi \cos \alpha + \sin \theta \sin \alpha) \\
 & + MV \cos \beta q - MV \sin \beta(p \cos \alpha + r \sin \alpha)
 \end{aligned} \tag{13}$$

$$\begin{aligned}
 MV \dot{\beta} = & Y \cos \beta + D \sin \beta - T \cos \alpha \sin \beta \\
 & + Mg(\cos \theta \sin \phi \cos \beta + \sin \theta \cos \alpha \sin \beta - \cos \theta \cos \phi \sin \alpha \sin \beta) \\
 & + MV(p \sin \alpha - r \cos \alpha)
 \end{aligned} \tag{14}$$

$$\begin{aligned}
 MV \dot{\gamma} = & Y \sin \beta - D \cos \beta + Y_e \sin \beta + T \cos \alpha \cos \beta \\
 & - Mg(\sin \theta \cos \alpha \cos \beta - \cos \theta \sin \phi \sin \beta - \cos \theta \cos \phi \sin \alpha \cos \beta)
 \end{aligned} \tag{15}$$

where we use M = mass to distinguish from the pitching moment m . The lift, drag, and side forces are denoted by L , D , and Y , respectively. T represents the aircraft thrust and can be expanded as

$$T = T_0 + T_1 \Delta \theta_r \tag{16}$$

where $\Delta\theta_r$ is the rotor blade pitch control input relative to the trim blade pitch, T_0 is the trim thrust, and T_1 is the linear relationship between the thrust and rotor pitch input.

The aerodynamic forces and moments applied to the aircraft in Equations (10)–(15) can be rewritten as follows:

$$l = C_l \bar{q} S b \quad (17)$$

$$m = C_m \bar{q} S \bar{c} \quad (18)$$

$$n = C_n \bar{q} S b \quad (19)$$

$$L = C_L \bar{q} S \quad (20)$$

$$D = C_D \bar{q} S \quad (21)$$

$$Y = C_Y \bar{q} S \quad (22)$$

$\bar{q} = \frac{1}{2} \rho_\infty V^2$ is the dynamic air pressure, S is the aircraft lifting surface area, b is the wingspan, and $\bar{c}$ is the mean chord length. The C_* coefficients in Equations (17)–(22) consist of the stability, dynamics, and control derivatives shown in Equation (23):

$$\begin{aligned} C_* = & \bar{C}_* + C_{*\alpha} \Delta\alpha + (C_{*u} + 2\bar{C}_*)(\frac{u}{\bar{u}} - 1) + C_{*\beta} \Delta\beta + C_{*p} p \frac{b}{2\bar{V}} + C_{*q} q \frac{\bar{c}}{2\bar{V}} + C_{*r} r \frac{b}{2\bar{V}} \\ & + C_{*\dot{p}} \dot{p} (\frac{b}{2\bar{V}})^2 + C_{*\dot{q}} \dot{q} (\frac{\bar{c}}{2\bar{V}})^2 + C_{*\dot{r}} \dot{r} (\frac{b}{2\bar{V}})^2 + C_{*\dot{\alpha}} \dot{\alpha} \frac{\bar{c}}{2\bar{V}} - C_{*\dot{\beta}} \dot{\beta} \frac{b}{2\bar{V}} \\ & + C_{*\Delta\theta_r} \Delta\theta_r + C_{*\Delta\delta_e} \Delta\delta_e + C_{*\Delta\delta_a} \Delta\delta_a + C_{*\Delta\delta_r} \Delta\delta_r \\ & + C_{*\dot{\theta}_r} \dot{\theta}_r + C_{*\dot{\delta}_e} \dot{\delta}_e + C_{*\dot{\delta}_a} \dot{\delta}_a + C_{*\dot{\delta}_r} \dot{\delta}_r \\ & + C_{*\ddot{\theta}_r} \ddot{\theta}_r + C_{*\ddot{\delta}_e} \ddot{\delta}_e + C_{*\ddot{\delta}_a} \ddot{\delta}_a + C_{*\ddot{\delta}_r} \ddot{\delta}_r \end{aligned} \quad (23)$$

$\bar{C}_*$ is the coefficient at the trim state and input conditions. The control inputs δ_e , δ_a , and δ_r correspond to the elevator, aileron, and rudder deflections, respectively. Variables with a preceding Δ indicate the value is relative to same trim conditions used to calculate the $\bar{C}_*$ terms.

We use the standard kinematics equations to derive the Euler angles

$$\dot{\phi} = p + (\sin \phi \tan \theta) q + (\cos \phi \tan \theta) r \quad (24)$$

$$\dot{\theta} = (\cos \phi) q - (\sin \phi) r \quad (25)$$

$$\dot{\psi} = (\sin \phi \sec \theta) q + (\cos \phi \sec \theta) r \quad (26)$$

and North-East-Down (NED) position

$$\begin{aligned} \dot{x} = & V [\cos \theta \cos \psi \cos \alpha \cos \beta \\ & + (-\cos \phi \sin \psi + \sin \phi \sin \theta \cos \psi) \sin \beta \\ & + (\sin \phi \sin \psi + \cos \phi \sin \theta \cos \psi) \sin \alpha \cos \beta] \end{aligned} \quad (27)$$

$$\begin{aligned} \dot{y} = & V [(\cos \theta \sin \psi) \cos \alpha \cos \beta \\ & + (\cos \phi \cos \psi + \sin \phi \sin \theta \sin \psi) \sin \beta \\ & + (-\sin \phi \cos \psi + \cos \phi \sin \theta \sin \psi) \sin \alpha \cos \beta] \end{aligned} \quad (28)$$

$$\begin{aligned} \dot{z} = & V [(-\sin \theta) \cos \alpha \cos \beta \\ & + (\sin \phi \cos \theta) \sin \beta \\ & + (\cos \phi \cos \theta) \sin \alpha \cos \beta] \end{aligned} \quad (29)$$

B. Wake Effects on Fixed-Wing Dynamics

Figure 3 shows the aerodynamic lift, drag, and pitching moment vs. angle of attack for the wing and horizontal tail as generated by NASA's FUN3D* computational fluid dynamics (CFD) tool suite. The wing

*<https://fun3d.larc.nasa.gov/>

C_L vs. α plot shows that for positive values of α , the linear region lies within $0^\circ \leq \alpha \leq 6^\circ$, which is where the nominal derivatives shown in Equation (23) are constant. For $\alpha > 6^\circ$, C_L vs. α is nonlinear and peaks at the stall angle $\alpha_{\text{stall,wing}} = 15^\circ$ for the wing and $\alpha_{\text{stall,h-tail}} = 20^\circ$ for the horizontal tail. When the surfaces reach $C_{L_{\text{max}}}$ at their respective stall angles, the flow begins to separate and lift decreases as the angle of attack increases. Figure 3 shows that the drag increases with positive angle of attack. Therefore, a wake upwash applied to either the wing or horizontal tail will induce additional drag onto the aircraft. The C_m vs. α plots show the pitch break occurs at 20° for both the wing and horizontal tail. The positive slope after the pitch break angle of attack indicates C_{m_α} becomes positive, thus losing stability in pitch. These plots indicate a maximum tolerable angle of attack of 15° to minimize adverse flight performance.

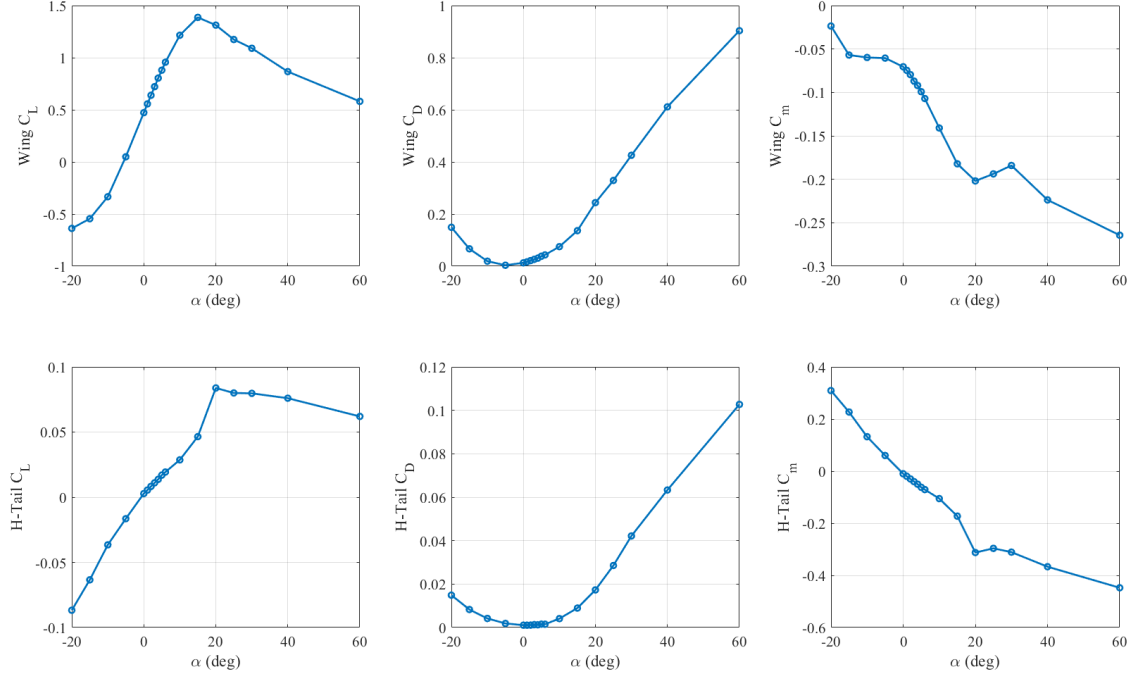


Figure 3: Lift, drag, and pitching moment coefficients vs. angle of attack for the RVLTLift+Cruise wing and horizontal tail.

The aircraft trim conditions are designed to operate within the linear range of C_L vs. α for performance and safety regions. However, wake vortex encounters with high levels of upwash may induce additional vertical velocity components that increase the effective angle of attack past the linear range, thus degrading the stability, control, and dynamics derivatives from their nominal values.

The wake-induced angle of attack $\alpha_w(\mathbf{p}, t)$ is estimated as

$$\alpha_w(\mathbf{p}, t) = \frac{w_w(\mathbf{p}, t)}{\bar{V}} \quad (30)$$

where $w_w(\mathbf{p}, t)$ is the wake-induced upwash as a function of aircraft position $\mathbf{p}$ and time t . In the event of a strong wake encounter, the nonlinear dynamics equations can no longer be approximated using a linear time-invariant state-space system. The nonlinear time-varying state-space system is then defined as

$$\Delta \dot{\mathbf{x}}_r = \mathbf{A}_{\mathbf{x}_r}(\mathbf{p}, t) \Delta \mathbf{x}_r + \mathbf{B}_{\mathbf{u}_r}(\mathbf{p}, t) \Delta \mathbf{u}_r \quad (31)$$

The derivation for $\mathbf{A}_{\mathbf{x}_r}(\mathbf{p}, t)$ and $\mathbf{B}_{\mathbf{u}_r}(\mathbf{p}, t)$ can be found in Baculi et al.¹²

The wake-induced aerodynamic loading is obtained by integrating the wake-induced lift contribution $c_{l_{\alpha_w}}^*(\mathbf{p}, t) \alpha_w(\mathbf{p}, t)$ and drag contribution $c_{d_{\alpha_w}}^*(\mathbf{p}, t) \alpha_w(\mathbf{p}, t)$.¹² The aerodynamic coefficients may be expressed

generally as

$$\begin{aligned}
C_F(\mathbf{p}, t) = & \bar{C}_F + C_{F_x}(c_{l_\alpha}^*(\mathbf{p}, t), c_{d_\alpha}^*(\mathbf{p}, t)) \mathbf{v} + C_{F_v}(c_{l_\alpha}^*(\mathbf{p}, t), c_{d_\alpha}^*(\mathbf{p}, t)) \dot{\mathbf{v}} \\
& + C_{F_\omega}(c_{l_\alpha}^*(\mathbf{p}, t), c_{d_\alpha}^*(\mathbf{p}, t)) \boldsymbol{\omega} + C_{F_{\dot{\omega}}}(c_{l_\alpha}^*(\mathbf{p}, t), c_{d_\alpha}^*(\mathbf{p}, t)) \dot{\boldsymbol{\omega}} \\
& + C_{F_\delta}(c_{l_\delta}^*(\mathbf{p}, t), c_{d_\delta}^*(\mathbf{p}, t)) \delta(t) + C_{F_{\alpha_w}}(c_{l_{\alpha_w}}^*(\mathbf{p}, t), c_{d_{\alpha_w}}^*(\mathbf{p}, t)) \alpha_w(\mathbf{p}, t) \\
& + C_{F_{\dot{\alpha}_w}}(c_{l_{\alpha_w}}^*(\mathbf{p}, t), c_{d_{\alpha_w}}^*(\mathbf{p}, t)) \dot{\alpha}_w(\mathbf{p}, t)
\end{aligned} \tag{32}$$

where $F = L, D, Y, l, m$, or n . When the wake-induced angle of attack pushes the effective angle of attack outside the linear range shown in Figure 3, a nonlinear stall model must be implemented at every time step to accurately represent the changes in derivatives due to the wake.

C. Control Design

To design the controllers for the RVLTLift+Cruise model, we use the nominal stability, control, and dynamics derivatives and build the LTI state-space system. We linearize Equations (10)–(15) and (24)–(26) to the rigid-body state vector $\mathbf{x}_r = [p \ q \ r \ \phi \ \Delta\alpha \ \Delta\beta \ \Delta V/\bar{V} \ \Delta h/\bar{h} \ \Delta\theta \ \psi]^\top$ and control inputs $\mathbf{u}_r = [\theta_r \ \delta_e \ \delta_a \ \delta_r]^\top$. The flight dynamics equations can then be written as

$$\mathbf{M}_r \dot{\mathbf{x}}_r = \mathbf{F}_{\dot{\mathbf{x}}_r} \dot{\mathbf{x}}_r + \mathbf{F}_{\mathbf{x}_r} \mathbf{x}_r + \mathbf{F}_{\ddot{\mathbf{u}}_r} \ddot{\mathbf{u}}_r + \mathbf{F}_{\dot{\mathbf{u}}_r} \dot{\mathbf{u}}_r + \mathbf{F}_{\mathbf{u}_r} \mathbf{u}_r \tag{33}$$

where $\mathbf{M}_r$ is the mass matrix, $\mathbf{F}_{\dot{\mathbf{x}}_r}$ is the aerodynamic forces and moments derivative matrix, and $\mathbf{F}_{\mathbf{x}_r}$ is the aerodynamic forces and moments matrix. $\mathbf{F}_{\ddot{\mathbf{u}}_r}$, $\mathbf{F}_{\dot{\mathbf{u}}_r}$, and $\mathbf{F}_{\mathbf{u}_r}$ are the control, control first derivative, and control second derivative matrices, respectively. Solving for $\dot{\mathbf{x}}_r$ rewrites Equation (33) into the standard state-space form:

$$\dot{\mathbf{x}}_r = (\mathbf{M}_r - \mathbf{F}_{\dot{\mathbf{x}}_r})^{-1}(\mathbf{F}_{\mathbf{x}_r} \mathbf{x}_r + \mathbf{F}_{\ddot{\mathbf{u}}_r} \ddot{\mathbf{u}}_r + \mathbf{F}_{\dot{\mathbf{u}}_r} \dot{\mathbf{u}}_r + \mathbf{F}_{\mathbf{u}_r} \mathbf{u}_r) \tag{34}$$

$$= \mathbf{A}_{\mathbf{x}_r} \mathbf{x}_r + \mathbf{B}_{\ddot{\mathbf{u}}_r} \ddot{\mathbf{u}}_r + \mathbf{B}_{\dot{\mathbf{u}}_r} \dot{\mathbf{u}}_r + \mathbf{B}_{\mathbf{u}_r} \mathbf{u}_r \tag{35}$$

where $\mathbf{A}_{\mathbf{x}_r} = (\mathbf{M}_r - \mathbf{F}_{\dot{\mathbf{x}}_r})^{-1} \mathbf{F}_{\mathbf{x}_r}$, $\mathbf{B}_{\ddot{\mathbf{u}}_r} = (\mathbf{M}_r - \mathbf{F}_{\dot{\mathbf{x}}_r})^{-1} \mathbf{F}_{\ddot{\mathbf{u}}_r}$, $\mathbf{B}_{\dot{\mathbf{u}}_r} = (\mathbf{M}_r - \mathbf{F}_{\dot{\mathbf{x}}_r})^{-1} \mathbf{F}_{\dot{\mathbf{u}}_r}$, and $\mathbf{B}_{\mathbf{u}_r} = (\mathbf{M}_r - \mathbf{F}_{\dot{\mathbf{x}}_r})^{-1} \mathbf{F}_{\mathbf{u}_r}$.

The linear model in Equation (35) is used to build a Linear Quadratic Integrator (LQI) to track the output $\mathbf{y} = [\Delta V/\bar{V} \ \Delta h/\bar{h} \ \mu \ \phi]^\top$ where $\mu = \psi + \beta$ is the course variable to control the direction of the aircraft's flight. These outputs were selected to enable altitude and direction tracking for scenarios such as following a leading aircraft. With these outputs, the reference commands are $\mathbf{r} = [(\Delta V/\bar{V})_r \ (\Delta h/\bar{h})_r \ \mu_r \ \phi_r]^\top$.

The LQI controller is designed as

$$\begin{aligned}
\Delta \mathbf{u} &= -\mathbf{K} \mathbf{z} \\
&= -\begin{bmatrix} \mathbf{K}_p & \mathbf{K}_i \end{bmatrix} \begin{bmatrix} \Delta \mathbf{x}_r \\ \mathbf{e} \end{bmatrix}
\end{aligned} \tag{36}$$

where the $\mathbf{K}_p$ terms are the proportional gains that regulate the states to 0 and $\mathbf{K}_i$ is the integrator state that tracks to the integral error term $\mathbf{e}$ to 0. The integral error term is derived by

$$\mathbf{e} = \int (\mathbf{y} - \mathbf{r}) dt \tag{37}$$

D. Actuator Dynamics

The actuator dynamics are also modeled in state-space form as

$$\dot{\mathbf{x}}_a = \mathbf{A}_{\mathbf{x}_a} \mathbf{x}_a + \mathbf{B}_{\mathbf{x}_a} \mathbf{u}_a \tag{38}$$

$$\mathbf{u} = \mathbf{C}_{\mathbf{x}_a} \mathbf{x}_a \tag{39}$$

where $\mathbf{x}_a = [\theta_r \quad \dot{\theta}_r \quad \ddot{\theta}_r \quad \delta_e \quad \dot{\delta}_e \quad \ddot{\delta}_e \quad \dots]^\top$ and $\mathbf{u}_a = [\theta_{r_c} \quad \delta_{e_c} \quad \delta_{a_c} \quad \delta_{r_c}]$ is the command vector. The state-space matrices are

$$\mathbf{A}_{\mathbf{x}_a} = \text{diag} \left(\begin{bmatrix} \mathbf{A}_{\theta_r} \\ \mathbf{A}_{\delta_e} \\ \mathbf{A}_{\delta_a} \\ \mathbf{A}_{\delta_r} \end{bmatrix} \right), \mathbf{B}_{\mathbf{x}_a} = \text{diag} \left(\begin{bmatrix} \mathbf{B}_{\theta_r} \\ \mathbf{B}_{\delta_e} \\ \mathbf{B}_{\delta_a} \\ \mathbf{B}_{\delta_r} \end{bmatrix} \right), \mathbf{C}_{\mathbf{x}_a} = \text{diag} \left(\begin{bmatrix} \mathbf{C}_{\theta_r} \\ \mathbf{C}_{\delta_e} \\ \mathbf{C}_{\delta_a} \\ \mathbf{C}_{\delta_r} \end{bmatrix} \right) \quad (40)$$

The $\mathbf{A}_*$, $\mathbf{B}_*$, and $\mathbf{C}_*$ matrices for the four control inputs take the same form:

$$\mathbf{A}_* = \begin{bmatrix} 0 & 1 & 0 \\ -900 & -30 & 0 \\ 1 & 0 & 0 \end{bmatrix}, \mathbf{B}_* = \begin{bmatrix} 0 \\ 900 \\ 0 \end{bmatrix}, \quad (41)$$

$$\mathbf{C}_* = \begin{bmatrix} 1 & 0 & 0 \end{bmatrix} \quad (42)$$

Figure 4 shows the Bode plot of the closed-loop system. The system has a gain margin of 5.4448 and a minimum phase margin of 39.9821° . Figure 5 shows the closed-loop response with actuator dynamics to an altitude command while keeping the airspeed at 200 ft/s and the course and roll angles regulated to 0° . For a smoother reference model, the altitude command is shaped by a first-order command shaping filter.

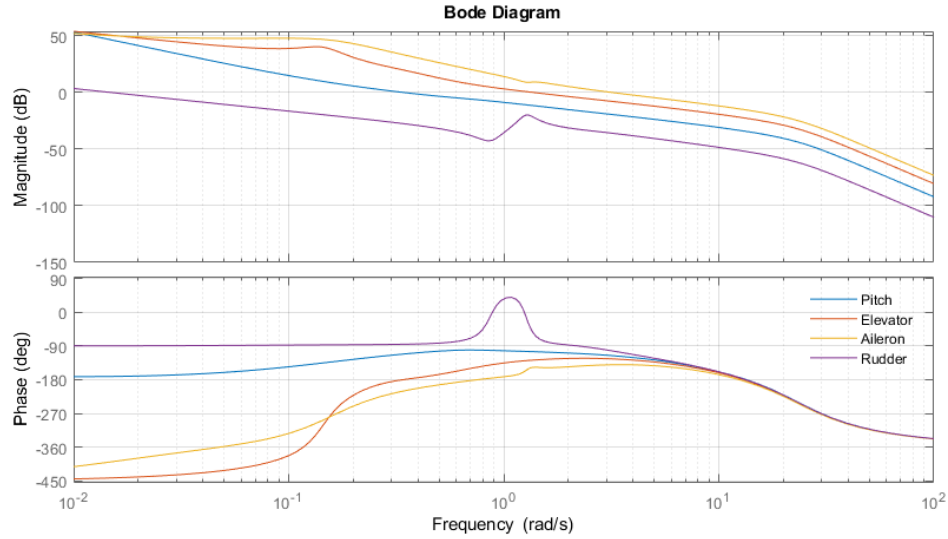


Figure 4: Bode plot of the four control inputs (rotor pitch, elevator, aileron, and rudder deflections) and four outputs (height, airspeed, course angle, and roll angle).

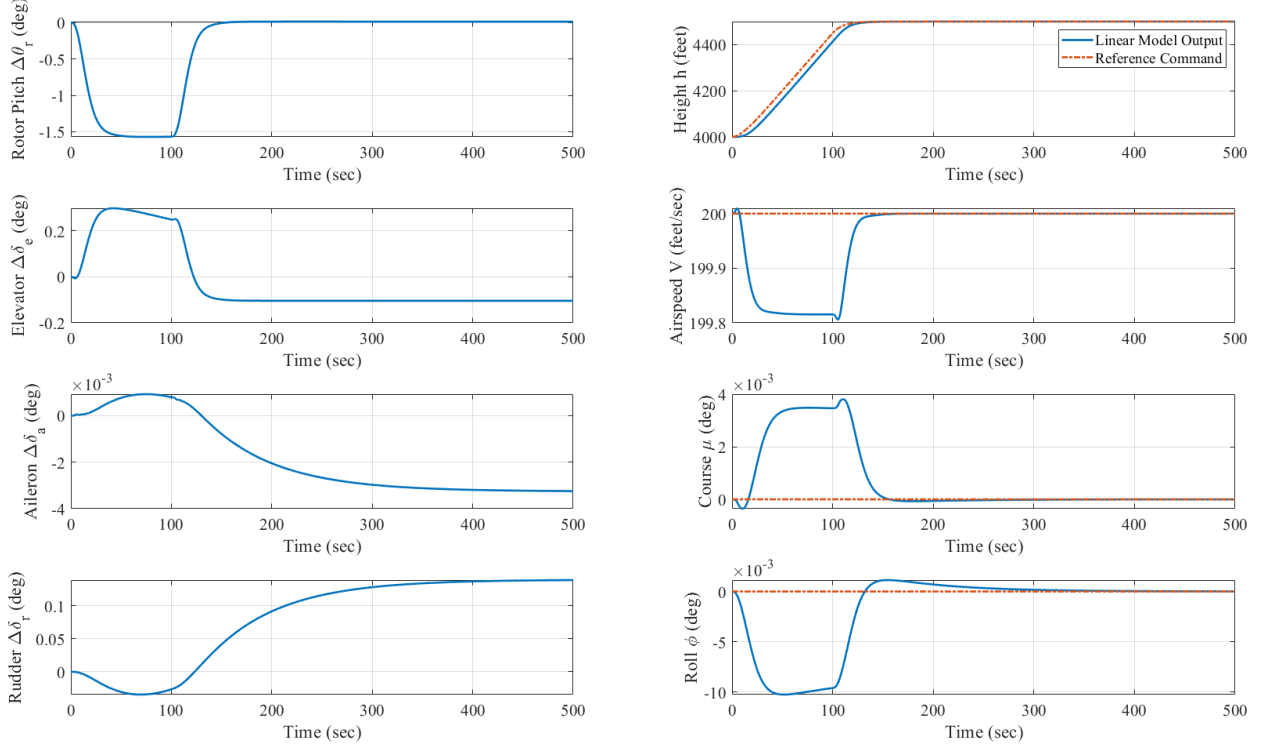


Figure 5: Control inputs (left) and outputs (right) of the closed-loop system in response to altitude, airspeed, course, and bank angle reference models. The altitude reference model is designed to achieve a 500 feet increase in altitude.

IV. Simulation Results

A. Wake-Induced Trim Conditions

We generated wake profiles along the trailing aircraft wing, horizontal tail, and vertical tail at various longitudinal separation distances. We began our wake separation evaluation with the 130 wingspan threshold studied by Baculi et al.,¹² which is around 6200 ft for the RVLTL Lift+Cruise with its 47.69 ft wingspan. We then incrementally closed the gap until a final distance of 500 ft. The wake profiles were then used to determine the wake-induced angle of attack, sideslip, forces, and moments on the trailing aircraft and the subsequent trim conditions to cancel out these effects. Equation (43) shows the linear system of equations used to determine the incremental angle of attack and sideslip angles and the elevator, aileron, and rudder deflections. The incremental angle of attack and sideslip are relative to the trim conditions shown in Table 5. The stability and control derivatives for Equation (43) are shown in Table 6 and generated from the trimming the aircraft using Table 5.

For the trim solution and closed-loop simulation in the following sections, we do not update derivatives at every time step as described in Section III.B. Keeping the derivatives constant allows us to observe the best-case scenario as an initial separation evaluation. Future work will include adding real-time derivative updates as a function of the wake-induced angle of attack and sideslip.

$$\begin{bmatrix} \Delta\alpha \\ \Delta\beta \\ \delta_e \\ \delta_a \\ \delta_r \end{bmatrix} = \begin{bmatrix} C_{L_\alpha} \bar{q}S & C_{L_\beta} \bar{q}S & C_{L_{\delta_e}} \bar{q}S & C_{L_{\delta_a}} \bar{q}S & C_{L_{\delta_r}} \bar{q}S \\ C_{Y_\alpha} \bar{q}S & C_{Y_\beta} \bar{q}S & C_{Y_{\delta_e}} \bar{q}S & C_{Y_{\delta_a}} \bar{q}S & C_{Y_{\delta_r}} \bar{q}S \\ C_{l_\alpha} \bar{q}Sb & C_{l_\beta} \bar{q}Sb & C_{l_{\delta_e}} \bar{q}Sb & C_{l_{\delta_a}} \bar{q}Sb & C_{l_{\delta_r}} \bar{q}Sb \\ C_{m_\alpha} \bar{q}S\bar{c} & C_{m_\beta} \bar{q}S\bar{c} & C_{m_{\delta_e}} \bar{q}S\bar{c} & C_{m_{\delta_a}} \bar{q}S\bar{c} & C_{m_{\delta_r}} \bar{q}S\bar{c} \\ C_{n_\alpha} \bar{q}Sb & C_{n_\beta} \bar{q}Sb & C_{n_{\delta_e}} \bar{q}Sb & C_{n_{\delta_a}} \bar{q}Sb & C_{n_{\delta_r}} \bar{q}Sb \end{bmatrix}^{-1} \begin{bmatrix} L_w \\ Y_w \\ l_w \\ m_w \\ n_w \end{bmatrix} \quad (43)$$

Table 6: Stability and control derivatives for the RVLTLift+Cruise trimmed at $V = 200$ ft/s and $h = 4000$ ft.

	α	β	δ_a	δ_e	δ_r
C_L	4.6655	0.013489	0.0006	0.3398	0.0149
C_D	0.33525	0.00077698	0.0011	0.0086	-0.0003667
C_Y	-0.0002642	-0.26532	0.02289	0.004995	0.2292
C_l	0.00043372	-0.022251	0.363	-0.00286	0.0271
C_m	-0.52536	-0.002221	0.0199	-1.137	-0.0173
C_n	-0.003379	0.093796	-0.02634	0.0002205	-0.0772

B. RVLTLift+Cruise Lead Aircraft

Using the fixed-wing wake model in Equation (4), we simulated the wake upwash and sidewash generated by an RVLTLift+Cruise aircraft flying North at a trimmed airspeed of 200 feet/sec. While separation minimum dictates a longitudinal distance between the leading and trailing aircraft, we also evaluated the effects of different lateral separation configurations. Figures 6–11 show the total angle of attack, sideslip, forces, and moments, and the subsequent trim conditions for aligned, half wingspan offset, and full wingspan offset lateral configurations with an RVLTLift+Cruise leading and trailing aircraft. The total effective angle of attack and sideslip can be computed by adding the wake-induced angles to trim:

$$\alpha_t = \bar{\alpha} + \alpha_w \quad (44)$$

$$\beta_t = \bar{\beta} + \beta_w \quad (45)$$

The leading aircraft is simulated to fly at the same airspeed as the trailing aircraft, $V = 200$ ft/s, to maintain a fixed separation.

Figure 6 shows the aligned configuration where the trailing aircraft is directly behind the leading aircraft. This creates a near symmetrical effective angle of attack along the trailing aircraft wing and horizontal tail. There is a very small induced sideslip on the vertical tail due to the trailing aircraft having a slight heading angle to offset the non-zero trim sideslip angle shown in Table 5. The greatest wake effect in this configuration is the negative lift force shown in Figure 7 which causes an increase in the incremental angle of attack. There is also a small change in incremental sideslip and a subsequent rudder deflection. Ultimately, these trim states and control inputs are within the typical saturation limits for a fixed-wing aircraft such as the RVLTLift+Cruise operating in the Cruise configuration.

Figure 8 shows the effective angle of attack and sideslip due to the half wingspan lateral offset where the trailing aircraft is aligned with the leading aircraft left wingtip, thus flying directly into the vortex. This configuration creates an upwash along the left side of the aircraft and a downwash along the right that induce a positive and negative angle of attack, respectively. There is both left and right induced sidewash along the vertical tail due to the aircraft aligning with the vortex. The combination of the positive α_t on the left wing and the negative α_t on the right wing creates the large positive rolling moment and negative yawing moment shown in Figure 9. The negative yawing moment is due to the increased drag on the left wing caused by its high α_t , which is consistent with the relationship between drag and angle of attack shown in Figure 3. Despite the total angle of attack lying just below the 15° stall angle of attack, the yawing moment results in rudder deflections that exceed the typical fixed-wing $\pm 10^\circ$ rudder saturation limit as close as 4000 ft. The trimmed incremental sideslip is also larger than desired for cruise flight.

Figure 10 shows the full wingspan lateral separation where the trailing aircraft right wingtip is aligned with the leading aircraft left wingtip. Due to the alignment of opposite wingtips, there is a large angle of attack on the trailing aircraft right wing. Figure 9 shows how this high angle of attack close to the right wingtip creates a negative rolling moment and positive yawing moment. Just as with the half wingspan offset, Figure 11 shows the greatest change in the trim conditions is in the sideslip and rudder deflection, which are above -4° up to 2000 ft until dropping to approximately -10° at 500 ft and 1000 ft. Figure 10 shows the total angle of attack just below stall at 500 ft separation, but the trimmed incremental sideslip and rudder deflection approach undesirable values.

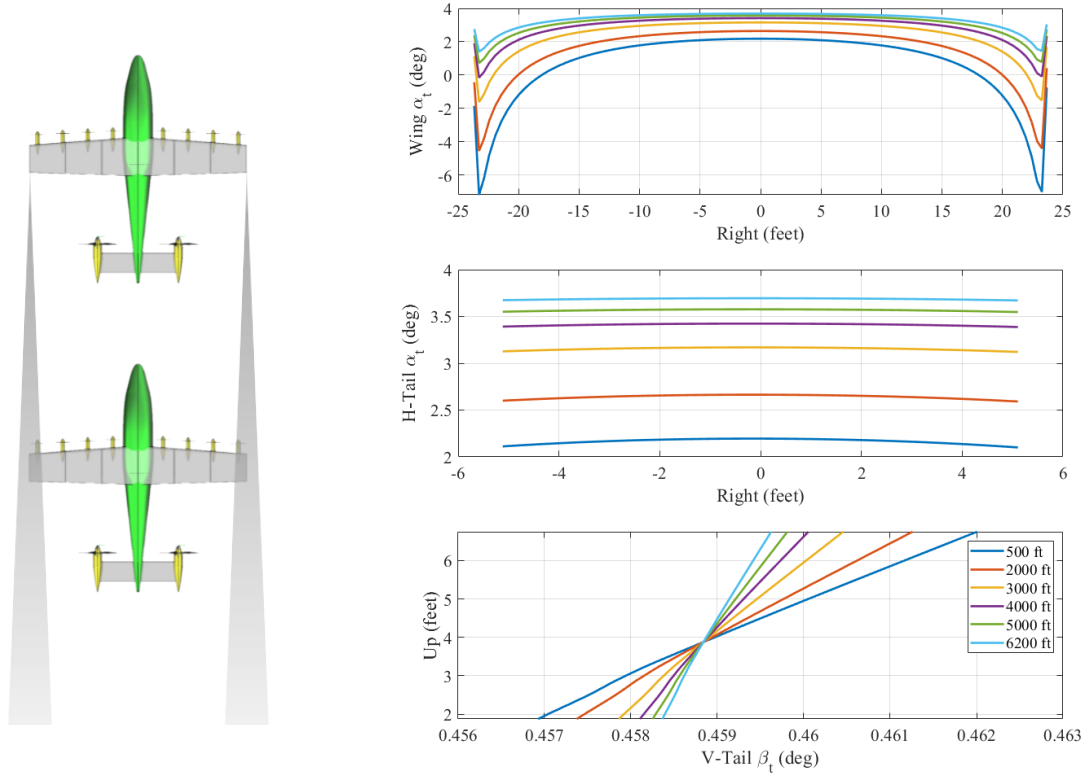


Figure 6: Total angle of attack along the wing and horizontal tail and total sideslip on the vertical tail due to an RVLT Lift+Cruise following another RVLT Lift+Cruise in the aligned configuration.

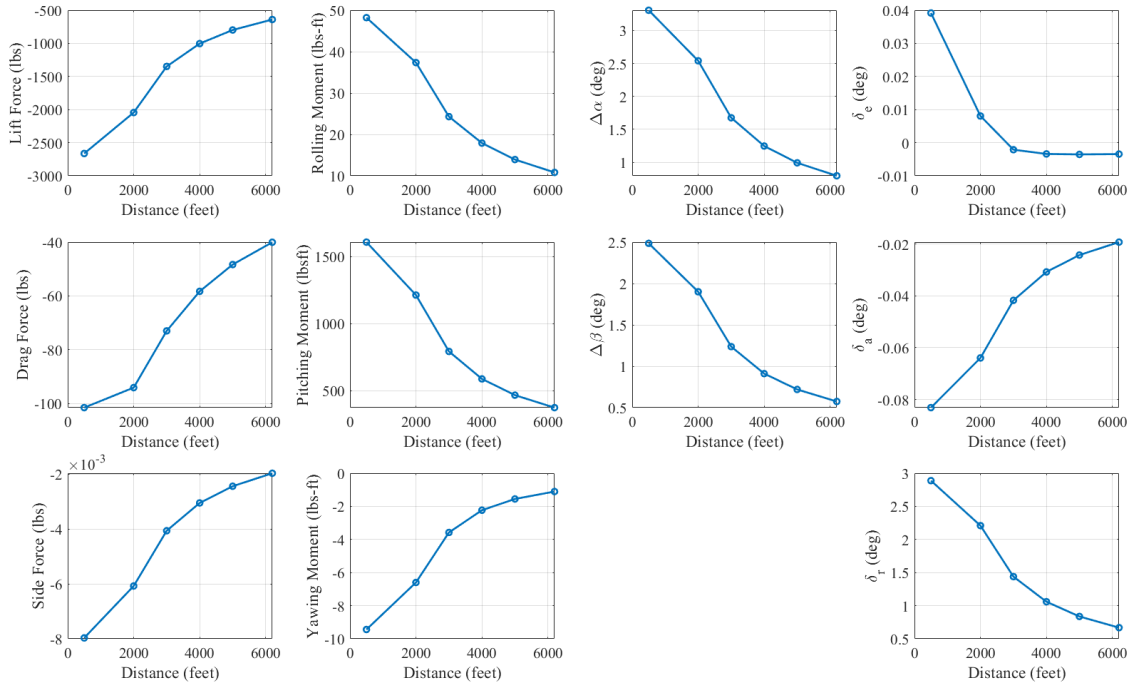


Figure 7: Wake-induced forces and moments and the subsequent trim conditions due to an RVLT Lift+Cruise following another RVLT Lift+Cruise in the aligned configuration.

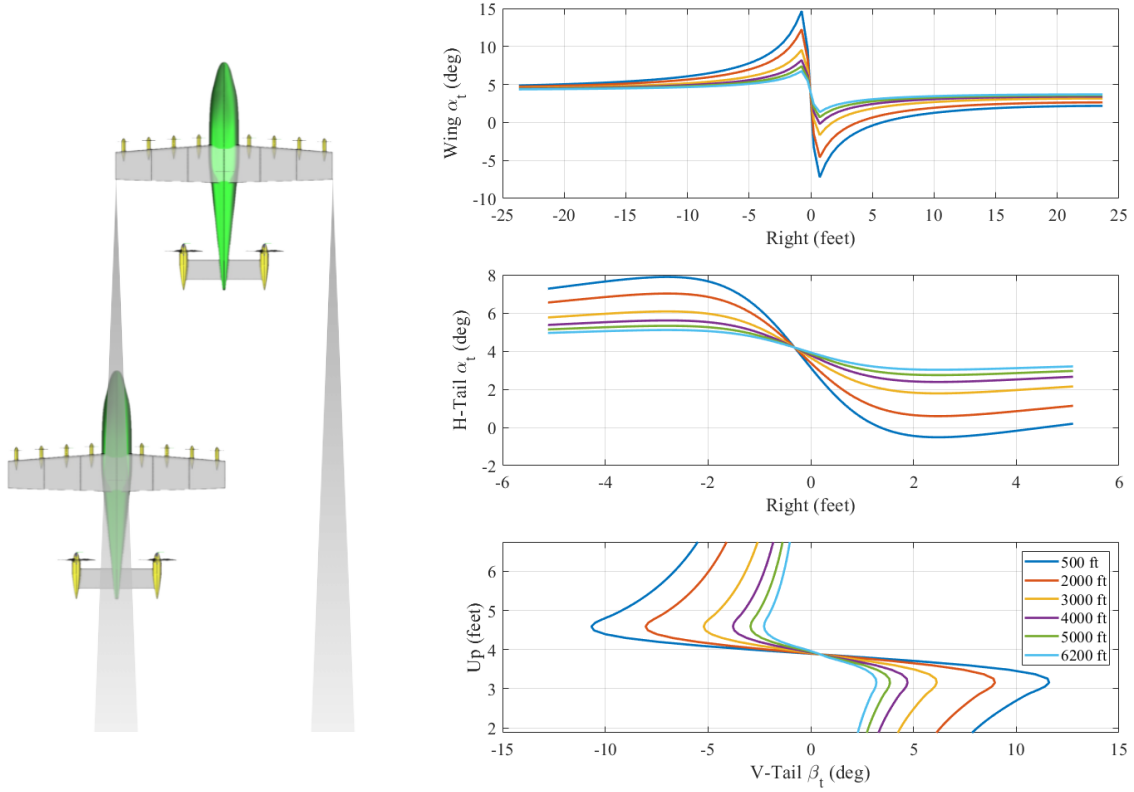


Figure 8: Total angle of attack along the wing and horizontal tail and total sideslip on the vertical tail due to an RVLTL Lift+Cruise following another RVLTL Lift+Cruise in the half wingspan lateral offset.

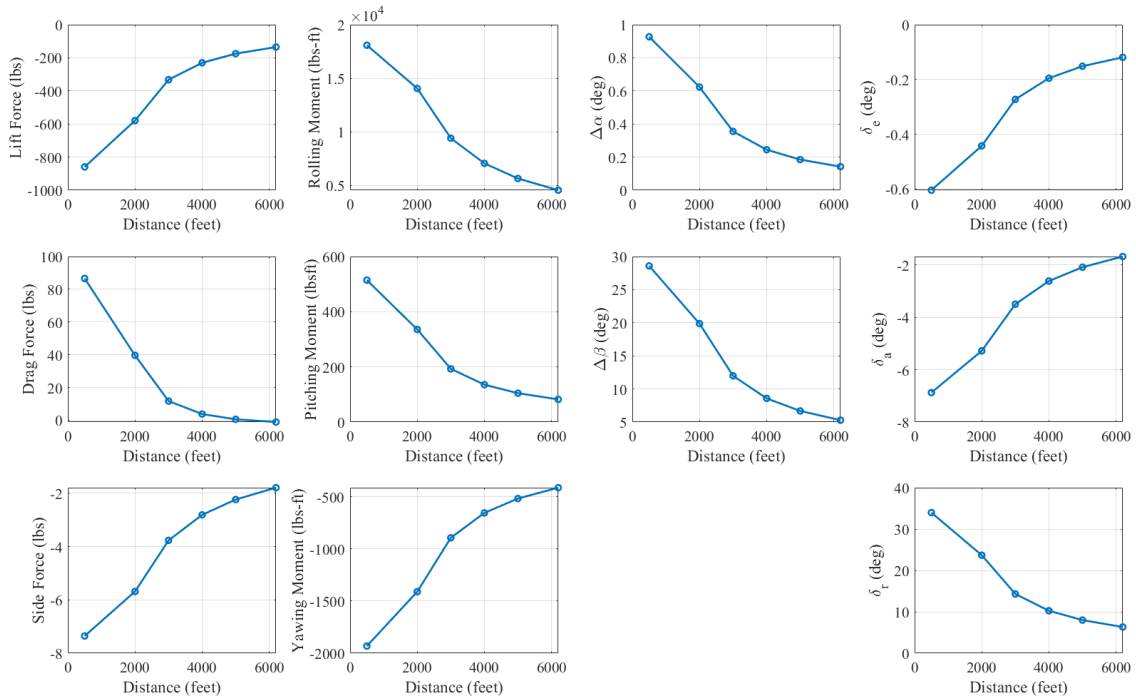


Figure 9: Wake-induced forces and moments and the subsequent trim conditions due to an RVLTL Lift+Cruise following another RVLTL Lift+Cruise in the half wingspan lateral offset.

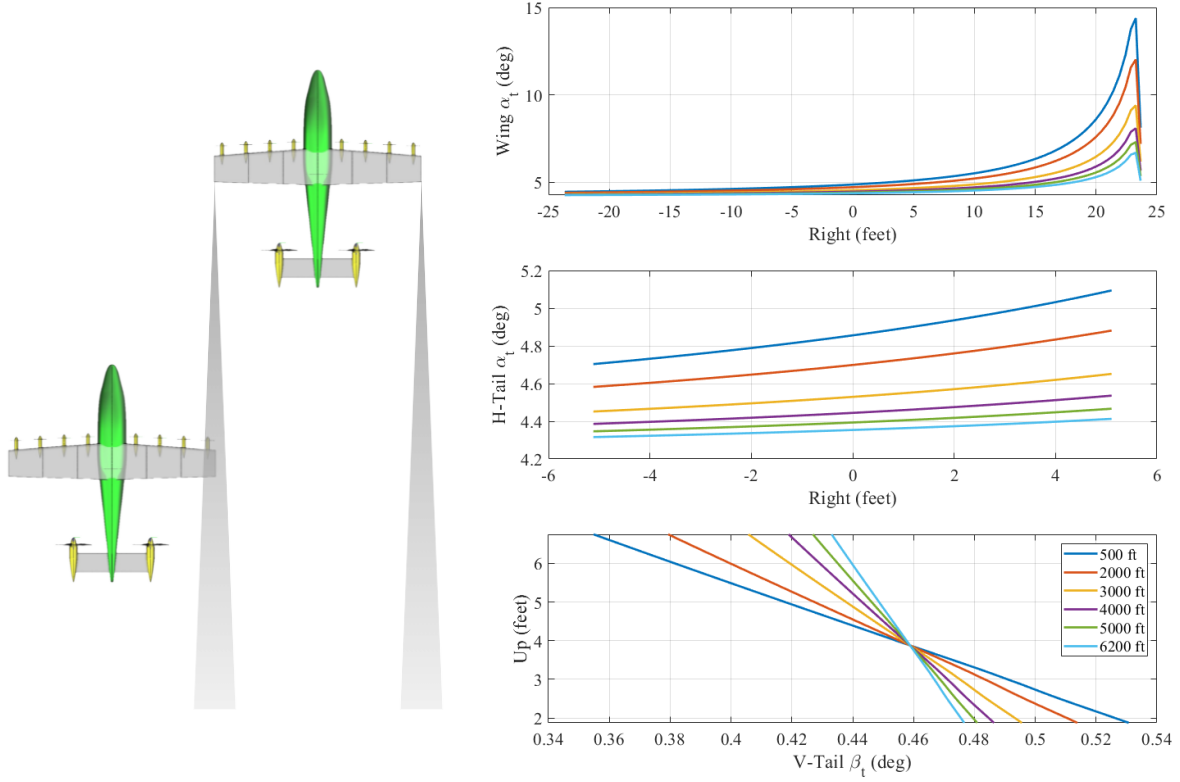


Figure 10: Total angle of attack along the wing and horizontal tail and total sideslip on the vertical tail due to an RVLTLift+Cruise following another RVLTLift+Cruise with a full wingspan lateral offset.

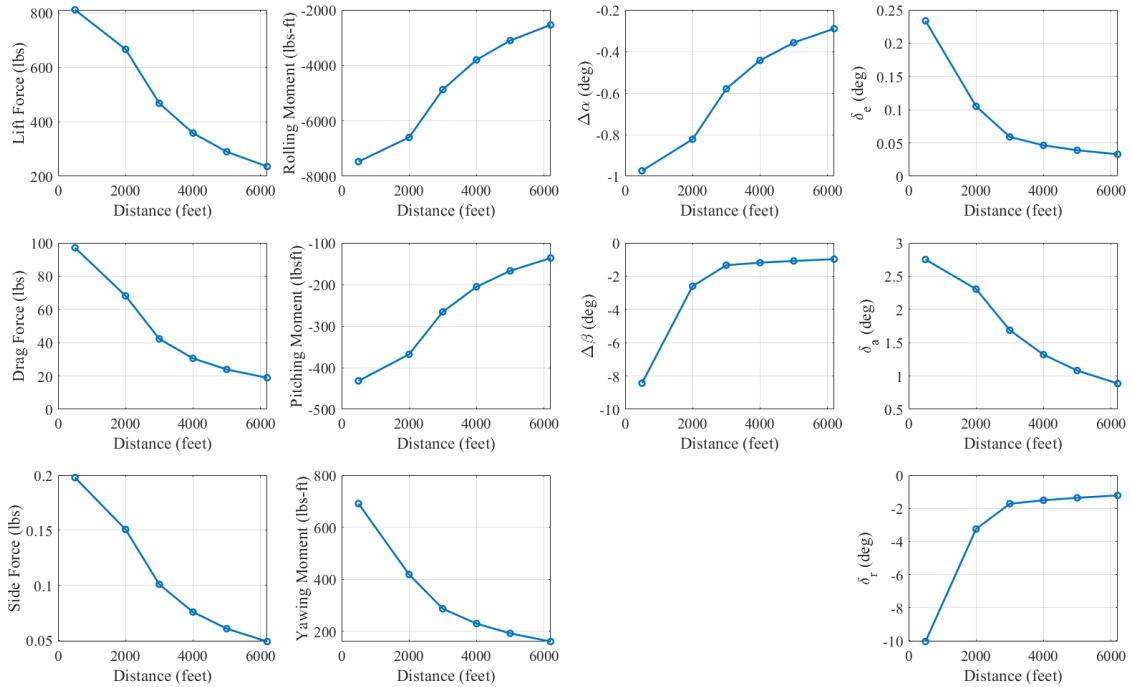


Figure 11: Wake-induced forces and moments and the subsequent trim conditions due to an RVLTLift+Cruise following another RVLTLift+Cruise with a full wingspan lateral separation.

C. SWEARINGEN Merlin Lead Aircraft

Figure 12 shows the total effective angle of attack and sideslip angles and the wake-induced forces and moments of an RVLTLift+Cruise trailing a Swearingen Merlin lead aircraft in the half wingspan lateral offset. The Swearingen Merlin is simulated to fly at the same airspeed as the trailing aircraft, $V = 200$ ft/s, to maintain a fixed separation. Due to the Swearingen Merlin being much heavier than the RVLTLift+Cruise with a similar wingspan (see Tables 3 and 5), we see a larger effective angle of attack and sideslip in Figure 12 compared to that generated by the RVLTLift+Cruise in Figure 8 at the same lateral and longitudinal separation. $\alpha_t > \alpha_{\text{stall}}$ as far out as 2000 ft. Figure 13 shows the large angle of attack results in greater wake-induced forces and moments and infeasible trim conditions. At 6200 ft separation, the trimmed incremental sideslip and rudder deflection are at similar values shown at 4000 ft separation in Figure 9 when trailing another RVLTLift+Cruise.

Figures 14 and 15 show the same setup as Figures 12 and 13, except the Swearingen Merlin is flying at its cruise airspeed of 438 ft/s as listed in Table 3. This changes the separation distance and allows to observe the wake effects of the Merlin operating within other phases of its typical flight range. Flying at 438 ft/s more halves the induced angle of attack and sideslip, which is consistent with the initial circulation strength defined in Equation (1). This brings the effective angle of attack well below stall, allowing the trailing RVLTLift+Cruise to achieve feasible trim conditions up to 3000 ft separation. The stark difference in wake effects due to the change in leading aircraft airspeed indicate separation criteria can be more tightly set in a RECAT Phase III dynamic pairwise manner that considers parameters such as altitude, airspeed, and atmospheric conditions.

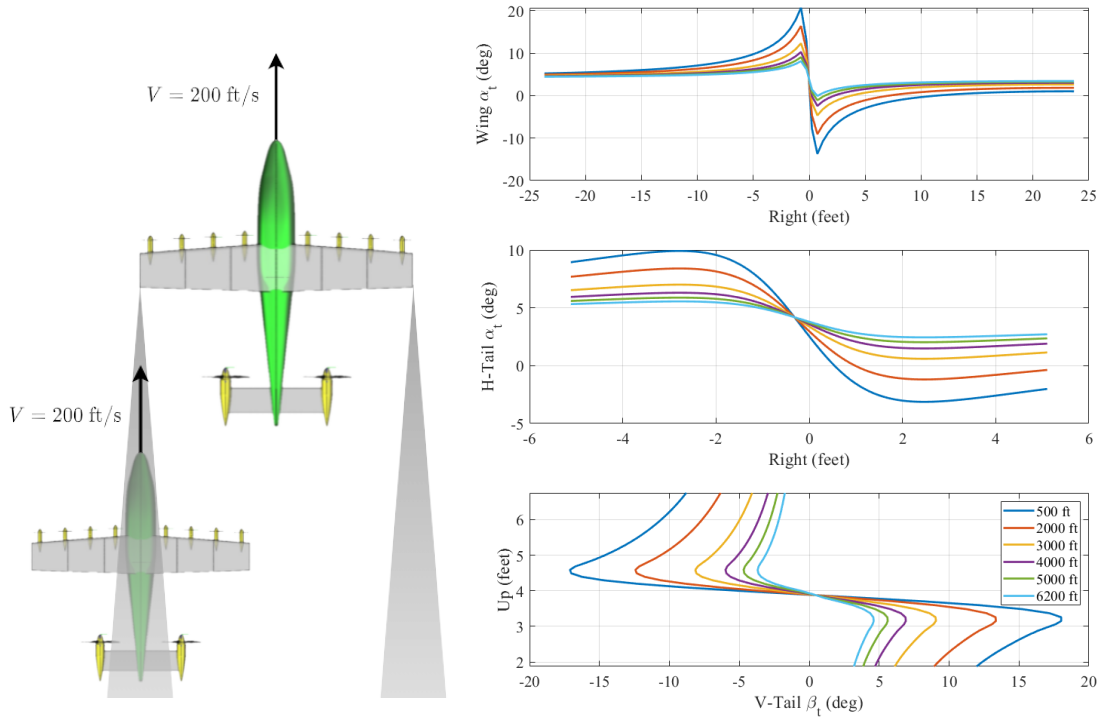


Figure 12: Total angle of attack along the wing and horizontal tail and total sideslip on the vertical tail due to an RVLTLift+Cruise following the larger Swearingen Merlin in the half wingspan lateral offset. The Swearingen Merlin is flying at 200 ft/s to match the trailing aircraft.

D. Closed-Loop Response

For these closed-loop simulations, we do not apply the nonlinear time-varying aerodynamic wake effects on the state-space system as outlined in Section III.B. Furthermore, we assume the wake effects are static and applied 10 seconds into the simulation. These assumptions are implemented to reduce the variability that

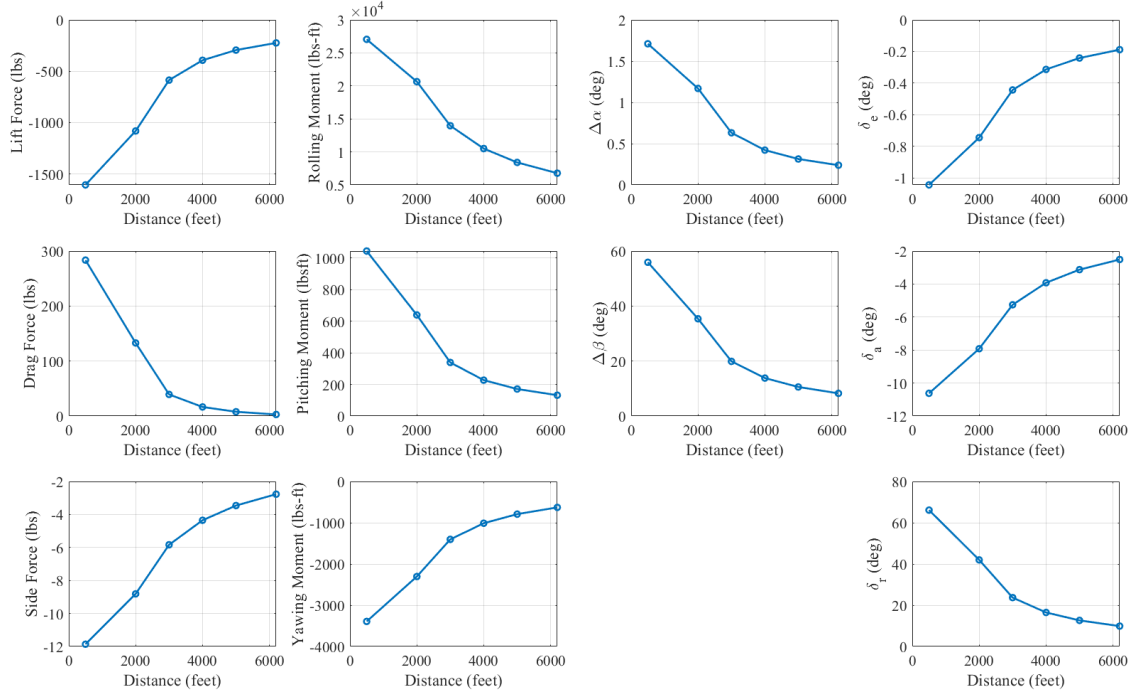


Figure 13: Wake-induced forces and moments and the subsequent trim conditions due to an RVLTLift+Cruise following a Swearingen Merlin in the half wingspan lateral offset configuration. The Swearingen Merlin is flying at 200 ft/s to match the trailing aircraft.

comes with real-time wake effects calculations. If the wake effects were implemented as functions of real-time position, the wake-induced forces and moments may push the trailing aircraft outside of the wake trail, thus reducing the effects. This would place heavy emphasis on the type of inner and outer loop controller implemented on the trailing aircraft to track the wake position, which is outside the scope of this study.

Figure 16 shows the RVLTLift+Cruise closed-loop system described in Section III.C responding to the 500 ft one wingspan wake inputs detailed in Section IV.B. The states and control inputs are shown to reach steady-state at or close to the trim values shown in Figure 11 at 500 ft. The steady-state rudder deflection settles just above the -10° lower saturation limit. As the deflection of control surfaces such as the rudder increases, their effectiveness also decreases. Our implementation is optimistic in assuming the control surface effectiveness remains constant throughout their operated ranges.

Figure 17 shows the closed-loop response due to the wake at 4000 ft separation and a half wingspan lateral offset. Although this separation distance is almost 10 times the separation in Figure 16, the control inputs settle at similar absolute values. While 500 ft is fairly close, 4000 ft separation is not conducive to a dense AAM airspace. Therefore lateral separation should be considered in addition to longitudinal separation when determining safe separation criteria.

While both Figures 16 and 17 show the controller successfully tracking the reference commands, the rudder deflection takes almost 5 minutes to converge to steady-state values. This is too long for an aircraft to hazardously trail another aircraft's wake. Furthermore, while 10° sideslip may be acceptable for short periods such as landing with crosswind, it is not desirable for prolonged flight such as the cruise phase. In this simulated scenario, the trailing aircraft has ample time to simply move out of the incoming wake vortex via a roll maneuver or an altitude change. This study focuses on the observing the trends of wake effects at different separation configurations and so we do not heavily focus on tuning the controller for performance. It is important to note that the RVLTLift+Cruise is a conceptual vehicle design and the stability and control derivatives are in preliminary stages of development. Further investigation is needed to determine why the wake-induced yawing moments are trimmed by relatively large rudder deflections.

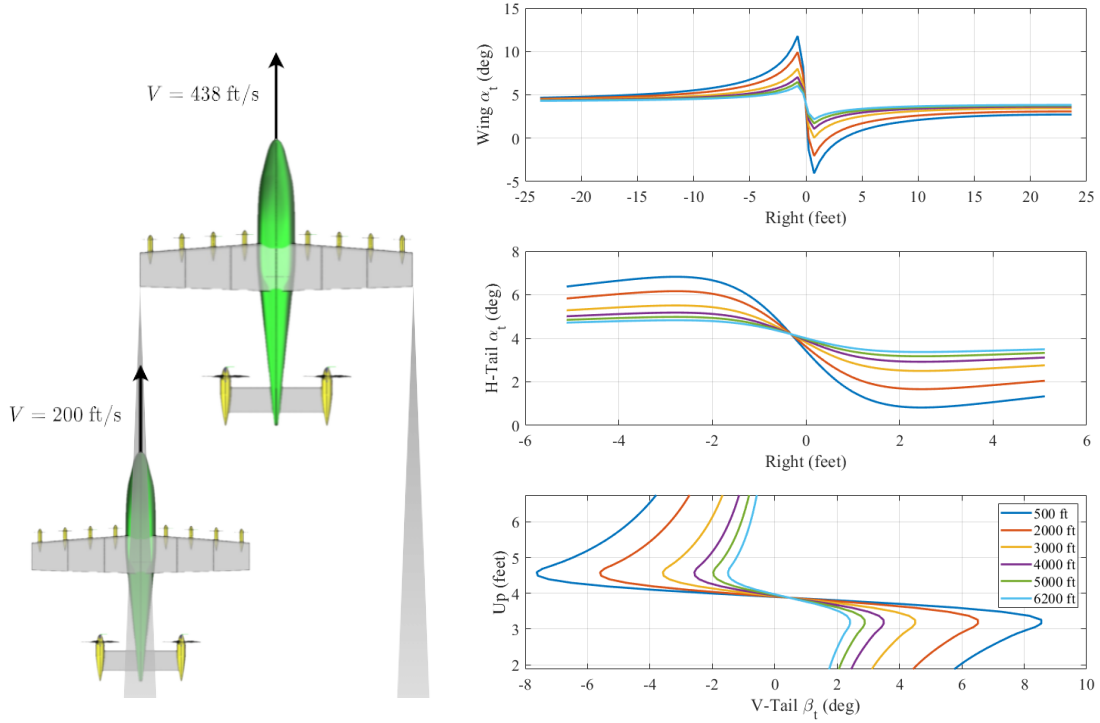


Figure 14: Total angle of attack along the wing and horizontal tail and total sideslip on the vertical tail due to an RVLTLift+Cruise following the larger Swearingen Merlin in the half wingspan lateral offset. The Swearingen Merlin is flying at a cruise airspeed of 438 ft/s.

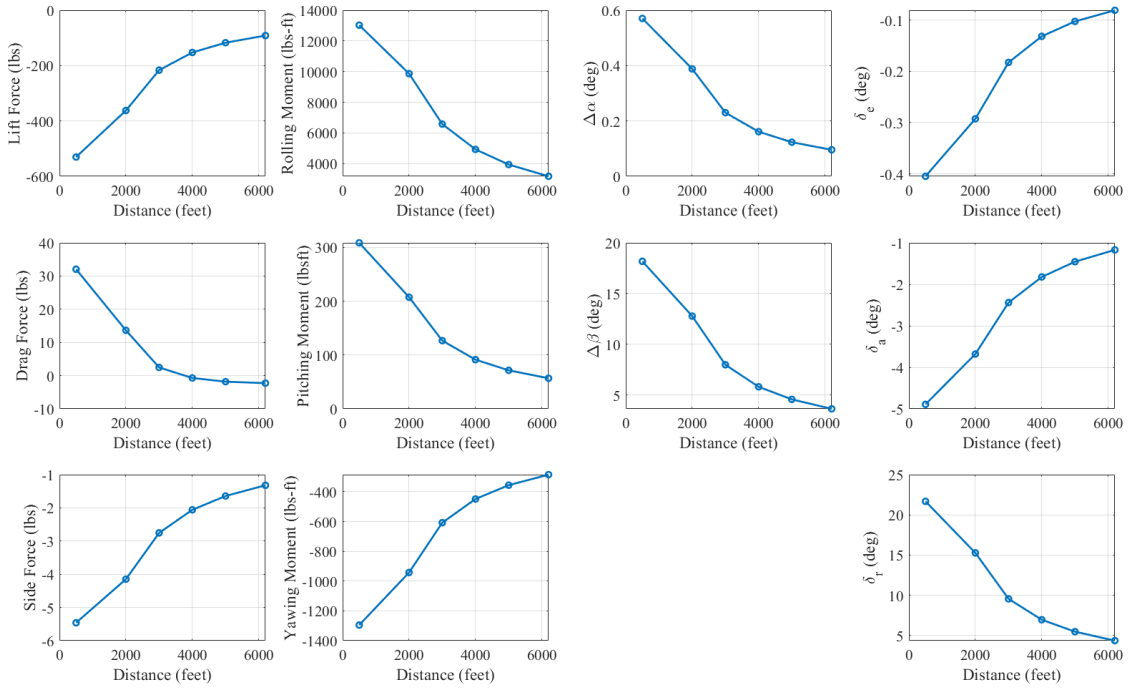


Figure 15: Wake-induced forces and moments and the subsequent trim conditions due to an RVLTLift+Cruise following a Swearingen Merlin in the half wingspan lateral offset configuration. The Swearingen Merlin is flying at a cruise airspeed of 438 ft/s.

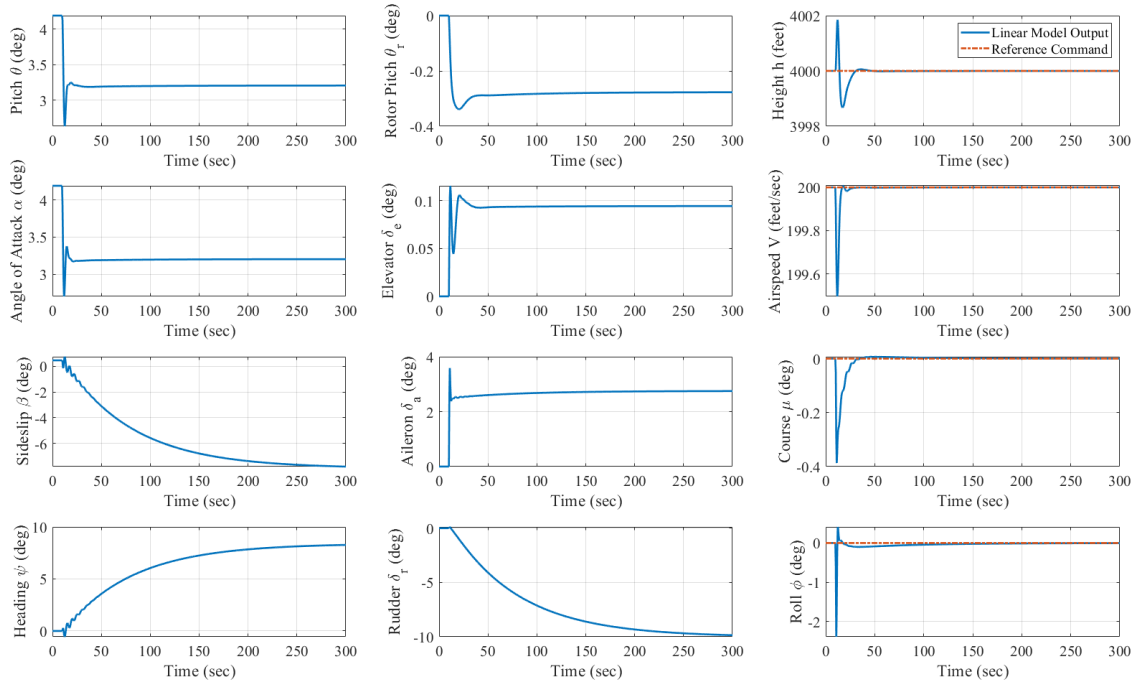


Figure 16: States, control inputs, and outputs of the RVLTLift+Cruise trailing another RVLTLift+Cruise at 500 ft separation and one wingspan lateral offset.

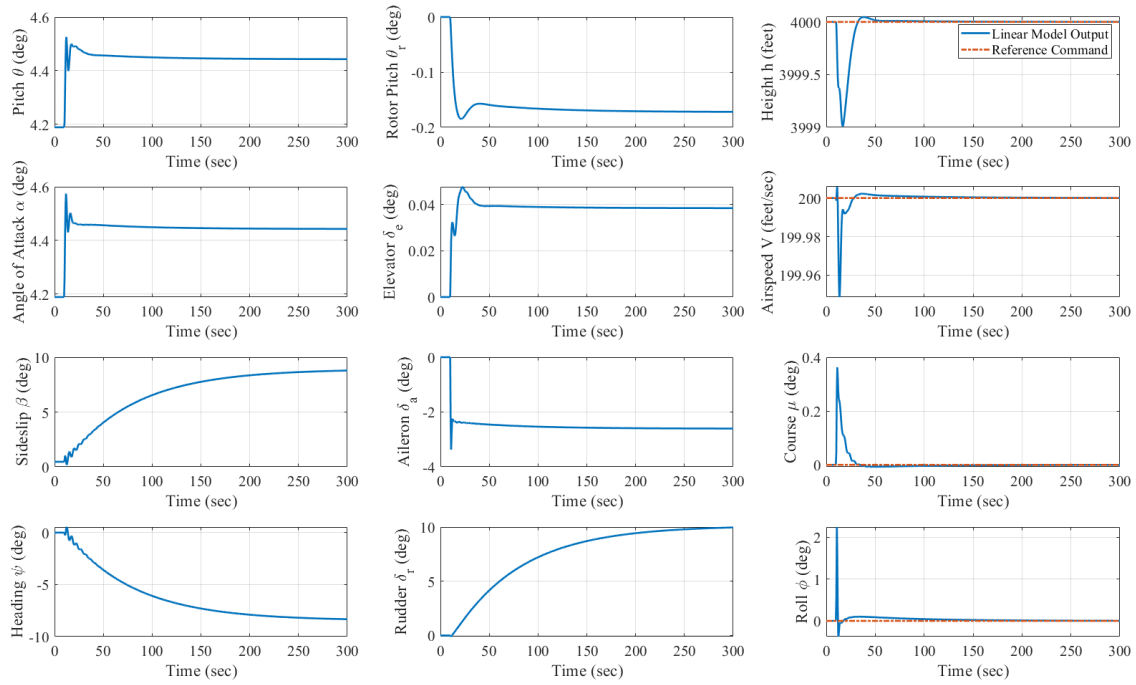


Figure 17: States, control inputs, and outputs of the RVLTLift+Cruise trailing another RVLTLift+Cruise at 4000 ft separation and a half wingspan lateral offset.

V. Conclusion

In this paper, we simulated the wake effects on an RVLTLift+Cruise UAM aircraft to evaluate separation criteria. Trim condition calculation and closed-loop simulations indicate a half wingspan lateral offset where the trailing aircraft flies directly into the wake vortex of the leading aircraft is the most hazardous lateral configuration. The separation minimum for this configuration is significantly larger compared to a full wingspan offset. Furthermore, the circulation strength is heavily influenced by the weight, wingspan, and airspeed of the wake-generating aircraft, therefore aircraft within the same wake RECAT category may produce drastically different wake downwash. To enable dense AAM airspace, it is recommended that separation criteria be determined in a dynamic pairwise manner that considers the type of aircraft involved as well as situational parameters such as atmospheric conditions, altitude, and airspeed.

Future work includes implementing the nonlinear time-varying wake effects on the RVLTLift+Cruise stability and control derivatives to achieve higher fidelity in evaluating separation criteria. In addition to fixed-wing wake effects, rotorcraft-generated wake vortices and urban canyon wake effects from buildings will be studied to better simulate a complex UAM airspace. Incorporating a diverse variety of factors into the wake interaction will allow for more comprehensive tools in determining safety metrics for separation between aircraft.

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