Design, Manufacturing, and Testing of a Data Acquisition System for Use in the Development of Control Systems for Fixed-Wing Unmanned Aerial Systems

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The development of accurate flight control systems for fixed-wing unmanned aerial systems (UAS) depends critically on high-quality experimental data suitable for system identification. System identification, the process of developing mathematical models of a system based on measured input-output data, requires precise, high-rate, and time-synchronized measurements of aircraft states and control inputs. Although modern autopilot platforms provide basic telemetry and data logging, their limited sampling rates, constrained sensor configurations, and timing inconsistencies often compromise the fidelity of the collected data, reducing its suitability for rigorous system identification. To address these challenges, a dedicated data acquisition system (DAS) was designed, manufactured, and tested specifically for fixed-wing UAS applications. The DAS enables high-rate logging of inertial, navigation, and actuator signals with precise timing alignment, ensuring that input-output relationships can be accurately captured for model estimation. This paper details the system architecture, hardware implementation, and validation methodology, highlighting how the DAS enhances the reliability and precision of system identification. Its deployment facilitates the development and tuning of flight control systems by providing the high-fidelity data necessary for creating accurate dynamic models of UAS behavior.

Nomenclature

a_x, a_y, a_z	=	body-axis translational accelerometer measurements, ft/s^2 or g
b	=	wing Span, ft
C_l, C_m, C_n	=	body-axis moment coefficients
C_X, C_Y, C_Z	=	body-axis force coefficients
$\bar{c}$	=	mean aerodynamic chord, ft
g	=	gravitational acceleration
I_x, I_y, I_z, I_{xz}	=	aircraft moments of inertia, slug · ft^2
L, M, N	=	applied body-axis moments, ft·lbf
m	=	aircraft mass, slug
p, q, r	=	body-axis angular velocity, rad/s or deg/s
$\bar{q} = \frac{1}{2}\rho V^2$	=	dynamic pressure, lbf/ft^2
S	=	wing reference area, ft^2
T	=	thrust, lbf
u, v, w	=	body-axis translational velocity, ft/s
V	=	true airspeed, ft/s
X, Y, Z	=	applied body-axis forces
α	=	angle of attack, rad or deg
β	=	angle of sideslip, rad or deg
η_e, η_a, η_r	=	control surface commands, normalized
ϕ , θ, ψ	=	Euler angles , rad or deg

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I. Introduction

RECENT advancements have significantly expanded the capabilities and accessibility of fixed-wing unmanned aerial systems (UAS), making them essential tools for commercial, military, and research applications. Small UAS, in particular, provide a low-cost and low-risk environment to enhance sensor, estimation, and control technologies before moving to larger aircraft. As mission demands grow, expectations for performance, robustness, and predictive accuracy continue to amplify. Meeting these expectations requires reliable dynamic models for controller design, simulation, and flight validation. For small UAS, traditional approaches, such as wind-tunnel testing or high-fidelity computational fluid dynamics, are often impractical due to cost and logistics. As a result, vehicle characteristics are derived from flight experiments. This dependence places a strong emphasis on the quality and consistency of the onboard measurements. System identification and control law development are, therefore, limited by the capability of their respective data acquisition infrastructure. Synchronization errors, insufficient sampling rates, or data loss can directly translate into modeling inaccuracies and poor controller performance. Thus, a dedicated data acquisition system (DAS) is needed to capture aircraft dynamics while respecting strict size, weight, and power limits.

This paper focuses on the design and implementation of a high-performance DAS tailored for the development of fixed-wing UAS control laws. The flight test ecosystem is built around a commercial autopilot and companion computing architecture running the PX4 firmware* [1]. This system supports real-time, multi-channel sensor interfacing to ensure synchronized data capture at appropriate sampling rates while remaining lightweight and reliable to avoid adverse effects on aircraft performance. For model development and analysis, collected measurements are processed using the System Identification Programs for AirCraft (SIDPAC) software toolbox [2]. The combined infrastructure links flight data collection directly to established system identification workflows and control design activities. The DAS developed provides the foundation for characterizing aircraft dynamics, evaluating controller performance, and comparing predicted and observed flight behavior. By enabling data-driven analysis and iterative refinement of control strategies, the system supports more accurate modeling and reliable autonomous flight development. In addition, the paper establishes a framework for future enhancements in UAS data acquisition and control system research.

II. Flight Test Architecture and Operation

A. Research Aircraft and Instrumentation

The aircraft selected for the research platform is the commercially available, radio-controlled (RC), foam structure, fixed-wing aircraft under the name 3000mm FOX Aerobatic EP Glider PNP by FMS, as shown in Figure 1. The aircraft is electronically powered by a 5000 mAh, 22.2 V Li-Poly battery, with a single motor mounted on the front of the fuselage that drives a two-bladed, 15-in diameter, 7.5-in pitch propeller. The aircraft is instrumented with a Pixhawk flight computer for data acquisition, namely a CubePilot Cube Orange running PX4 firmware Version 1.16.1. This system is responsible for attitude and rate control, sensor function, actuator outputs, flight modes, and flight data logging. In addition, a companion computer was powered through Universal Serial Bus Type C (USB-C) and connected using an FTDI UART-to-USB converter for automated excitation maneuvers for system identification and the future implementation of custom or adaptive control laws. Specifically, the co-computer was a Raspberry Pi 5 running the Ubuntu 24.04 Linux distribution and using the Robot Operating System (ROS) software libraries †[1]. The sensor suite includes triple-redundant inertial measurement units, including accelerometers and gyroscopes, a Here4 multi-band Real-Time Kinematic (RTK) Global Navigation Satellite System (GNSS) module, and a Thunderfly TFRPM01 RPM sensor. The mass and geometric properties of the aircraft, tabulated in Table 1, capture key parameters required for aircraft system identification. The values labeled "unknown" in Table 1 are still in progress. The mass of the aircraft requires the final assembly of all avionics components before it can be placed on a scale. Inertia properties are to be determined from flight test data; the methodology used can be found in Ref.[3].

B. Flight-Test Operations

The ground-based pilot will operate the aircraft using a RadioMaster TX16S Mark II Max shown in Figure 2. The corresponding switch configuration is detailed in Table 2.

Pilot-flown system identification maneuvers shall be conducted in PX4 Manual mode, which disables feedback stabilization and allows measurement of the aircraft's open-loop dynamic response. The first flight will focus on testing

*Information available online at https://docs.px4.io/main/en/

†Information available online at https://docs.ros.org/

Fig. 1 FMS 3000mm FOX Aerobatic EP Glider PNP

Table 1 FOX Geometric and Mass Properties

Symbol	Value	Units
m	In-Progress	slug
$\bar{c}$	8.3564	in
b	9.85	ft
S	8.0083	ft^2
I_{xx}	In-Progress	slug·ft^2
I_{yy}	In-Progress	slug·ft^2
I_{zz}	In-Progress	slug·ft^2
I_{xz}	In-Progress	slug·ft^2
I_{xy}, I_{yz}	≈ 0	slug·ft^2

Table 2 Radio Controller Transmitter Channels

Channel	Use	Selection	Switch Type
1	Actuator Input	Aileron	Right Control Stick (L/R)
2	Actuator Input	Elevator	Right Control Stick (Fwd/Aft)
3	Actuator Input	Throttle	Left Control Stick (Fwd/Aft)
4	Actuator Input	Rudder	Left Control Stick (L/R)
5	Actuator Input	Flap	Three-place switch
6	PX4 Mode	Manual, Stabilized, or Offboard	Three-place switch
7	Arming	Unarmed-Armed	Two-place switch
8	PTI Activation	Off-On	Two-place switch
9	PTI Amplitude	0% to 100%	Rotary Knob
10	PTI Throttle Amplitude	0% to 100%	Rotary Knob

and validating three flight modes: Manual, Acrobatic, and Stabilized. The second flight will evaluate updated parameter settings derived from the first flight, such as the maximum speed determined during a level acceleration maneuver, and will include testing of Altitude mode to ensure safe automated operation in the event of a lost RC signal. The third flight will consist of automated system identification maneuvers executed in PX4 Offboard mode, enabling the companion computer to inject programmable test inputs (PTIs).

Fig. 2 RadioMaster TX16S Mark II Max RC Transmitter

For these PTIs, multi-axis multi-sine inputs were chosen because they provide wide-band, simultaneous excitation of all control axes, covering the expected frequency range of the aircraft's dynamic modes. Each multi-sine input is composed of multiple sinusoids with unique harmonic frequencies, optimized phase angles, and specified power distributions, designed to maximize information content while minimizing peak-to-peak amplitude. Harmonic frequencies are selected to cover a wide range, providing robustness to uncertainty in the aircraft's modal frequencies, while optimized phase and amplitude distributions ensure safe excitation without saturating actuators or excessively disturbing the aircraft. Multiple inputs are mutually orthogonal in both time and frequency domains, allowing simultaneous application of all axes, which reduces the time required to inject sufficient input energy and increases test efficiency. This combination produces high-quality uncorrelated data, allowing for accurate estimation of aerodynamic coefficients and dynamic response without requiring excessively long flights or single-frequency tests.[4][5]

Additional flights will be conducted to validate results using maneuvers not used for system identification, such as manually piloted rudder and elevator doublets and aileron 1–2–1 maneuvers. Automated inputs will be initiated by setting the PTI activation switch to the "On" position while the vehicle is in PX4 Offboard mode. For each PTI listed in Table 3, eight 30-second maneuvers are to be recorded. The most complex input, consisting of full independent actuation of all control surfaces, is shown in Figure 3.

Table 3 Programmed Test Inputs

PTI	Axes
Three-Axis Glide Multi-sine	Aileron, Elevator, Rudder
Four-Axis Aeropropulsive Multi-sine	Aileron, Elevator, Rudder, Throttle
Seven-Axis Aeropropulsive Multi-sine	Left and Right Aileron, Left and Right Flap, Elevator, Rudder, Throttle

Flight testing shall be conducted at the North Farm facility at Mississippi State University, using a semi-prepared airfield. The test site consists of a flat, open field environment with minimal surrounding obstructions, providing adequate lateral and longitudinal clearance for safe small unmanned aircraft operations. All system identification maneuvers will be conducted under controlled environmental and flight conditions to ensure data quality, repeatability, and minimal atmospheric disturbance. Operations require clear weather and good visibility to maintain safe visual line-of-sight control. Wind conditions are to be strictly limited to reduce exogenous aerodynamic excitation, with an ideal steady wind gust magnitude of less than 1 knot and an acceptable upper limit of less than 5 knots. Before

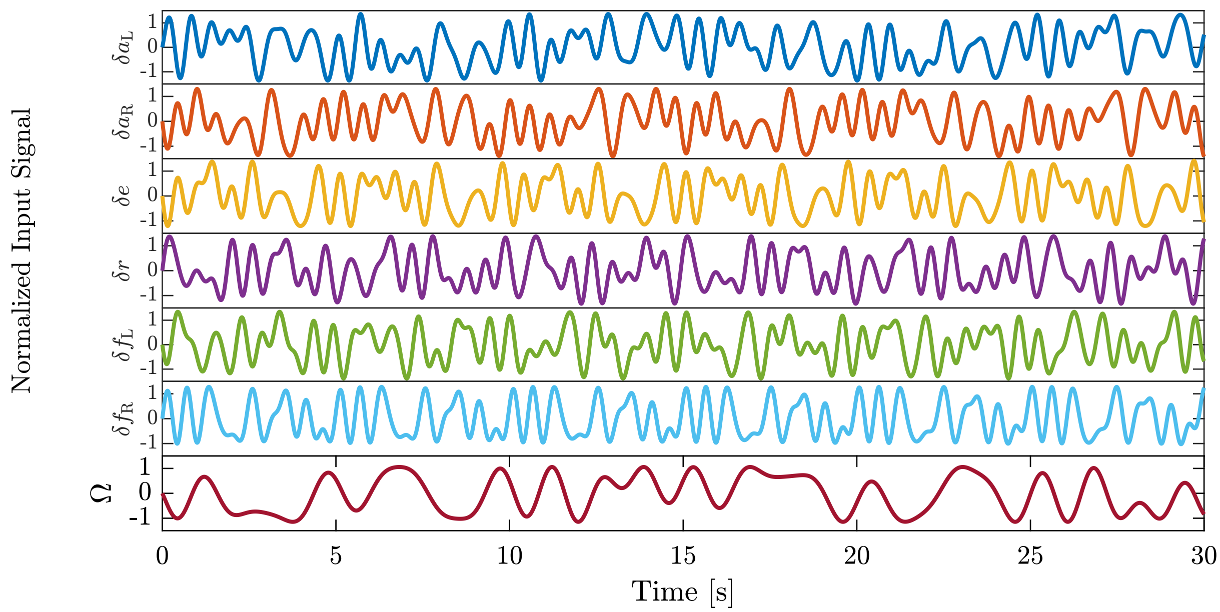

Fig. 3 Example of 7-axis aeropropulsive multi-sine excitation signal.

initiating each maneuver, the aircraft is to be established in steady, trimmed, wings-level flight at a constant airspeed and approximately constant altitude. Control inputs are to be applied only after steady-state conditions are achieved, and sufficient recovery time is allowed between maneuvers to restore trim conditions and eliminate transient effects.

III. System Identification Methodology

System identification describes the approach used to obtain mathematical representations of aircraft vehicle dynamics from experimental flight data. Specifically, it refers to the process of constructing useful dynamic models of key flight parameters based on measured or calculated inputs and outputs of a system. This process is done using imperfect knowledge of the inputs and outputs due to noise and bias, which requires it to be taken into account during analysis. Control excitations, together with the recorded aircraft response, provide the information required to estimate aerodynamic and control parameters suitable for analysis, simulation, and the development of control laws.

To create a model that is consistent with conventional flight-dynamics formulations, a standard aircraft motion model incorporated with several simplifying assumptions is adopted. The aircraft is modeled as a rigid six-degree-of-freedom body acted upon by external forces and moments due to gravity, thrust, and aerodynamics. The Earth is treated as a flat and non-rotating body, which establishes an inertial reference frame for the model. Atmospheric disturbances are neglected, with zero wind assumed, and gravitational acceleration is taken as constant. Aircraft mass is assumed to be non-varying over the time scale of each flight. Symmetry is assumed about the X-Z plane, allowing the products of inertia I_{xy} and I_{yz} to be approximated as zero. Propulsive modeling assumes that thrust acts along the longitudinal body axis, while rotational effects are not represented [6, 7]. From these assumptions, the governing rigid-body equations of the system can be derived. The translational dynamics equations can be calculated as

$$\dot{u} = rv - qw - g\sin\theta + \frac{X}{m} \tag{1}$$

$$\dot{v} = pw - ru + g\cos\theta\sin\phi + \frac{Y}{m} \tag{2}$$

$$\dot{w} = qu - pv + g\cos\theta\cos\phi + \frac{Z}{m} \tag{3}$$

The rotational dynamics equations can be calculated as

$$I_x\dot{p} - I_{xz}\dot{r} = L + (I_y - I_z)qr + I_{xz}pq \tag{4}$$

$$I_y\dot{q} = M + (I_z - I_x)pr + I_{xz}(r^2 - p^2) \tag{5}$$

$$I_z\dot{r} - I_{xz}\dot{p} = N + (I_x - I_y)pq - I_{xz}qr \tag{6}$$

It is important to note that since the inertia properties are unknown due to being derived from flight testing, the rotational dynamic equations will be oriented to create the inertia properties as unknown variables in the system. Specifically, the values will be derived from the equation error method. The methodology used to calculate the inertia properties is followed from Ref.[3]. For fixed-wing aircraft, the aerodynamic forces and moments are typically expressed in non-dimensional form to remove their dependency on airspeed, atmospheric density, and aircraft scale. Due to the foundation of the dynamics equations being known from physical principles, it is necessary in aircraft system identification to determine explicit expressions for the non-dimensional aerodynamic forces and moments in terms of the state and input variables. Although the coefficients are classified as response variables to the system, the individual quantities cannot be directly observed or measured, which requires them to be derived from other measurements or properties. From this, the body-axis force coefficients can be calculated as

$$C_X = \frac{ma_x}{\bar{q}S}, \quad C_Y = \frac{ma_y}{\bar{q}S}, \quad C_Z = \frac{ma_z}{\bar{q}S} \tag{7}$$

The body-axis moment coefficients can be calculated as

$$C_l = \frac{1}{\bar{q}Sb}[I_x\dot{p} - I_{xz}\dot{r} + (I_z - I_y)qr - I_{xz}pq], \tag{8}$$

$$C_m = \frac{1}{\bar{q}S\bar{c}}[I_y\dot{q} + (I_x - Iz)pr + I_{xz}(p^2 - r^2)] \tag{9}$$

$$C_n = \frac{1}{\bar{q}Sb}[I_z\dot{r} - I_{xz}\dot{p} + (I_y - I_x)pq + I_{xz}qr] \tag{10}$$

The explanatory variables used to develop a representation of these force and moment coefficients of the body-axis include the angle of attack α in radians, angle of sideslip β in radians, the dimensionless angular rates, and finally, the normalized control surface signals η_e, η_a, η_r in μs. The normalized control surface signals are used due to the absence of a control surface model; as a result, the method used to complete this process can be found in [8]. As we do not know the true nonlinear aerodynamic function, the models for each response variable are approximated using a polynomial expansion of the explanatory variables, namely the Taylor series expansion. The coefficients of the polynomial correspond to physical sensitivities that are estimated from flight-test data.

$$\alpha = \tan^{-1}(w/u), \quad \beta = \sin^{-1}(v/V), \quad \hat{p} = \frac{pb}{2V}, \quad \hat{q} = \frac{q\bar{c}}{2V}, \quad \hat{r} = \frac{rb}{2V} \tag{11}$$

where V is the true airspeed of the vehicle:

$$V = \sqrt{u^2 + v^2 + w^2} \tag{12}$$

The system identification process began with a clear motivation to develop an accurate aerodynamic flight model of the research aircraft for future use in the development and implementation of control laws. To support this, an appropriate avionics package with equipment found in Sec. II.A was installed. Before flight, ground tests will be conducted to calibrate sensors, verify actuator response and direction, and confirm the correct data relay and integrity. A structured flight-test shall follow, using excitation maneuvers to stimulate the dynamics of the aircraft. The collected data will then be processed, including resampling, filtering, and smoothing for further analysis. The model structure will then be determined from the measured data, and the model parameters will be estimated. The resulting model shall be validated against aerodynamic theory and rigid-body dynamics, evaluating the physical consistency of the estimated parameters. If the model is inadequate, further analysis will be required to determine the data needed to identify an accurate model. If the model is accepted, it is deemed a reliable representation of the aircraft's dynamics covering the tested flight envelope.

IV. Data Processing

Processing flight data is a critical component of the system identification process, as it transforms raw flight measurements into properly conditioned signals that can be used in model analysis and parameter estimation. This procedure includes, but is not limited to, smoothing noisy data, correcting time delays, calculating unmeasured signals, and performing data compatibility and necessary corrections. Although this process can be time-consuming and is sometimes overlooked, it is essential to provide accurate and reliable modeling results. The data processing methodology used in this project will closely follow Ref.[7].

A. Data Importing, Resampling, and Formatting

From the raw PX4 message format, the data will be imported into MATLAB, where the raw signals will be converted to have a constant sampling rate of $f_s = 50\,\mathrm{Hz}$. A detailed flow chart of the subsequent resampling process can be found in Ref. [7]. All measured or estimated signals will be converted to the 50 Hz sampling rate, but with varying techniques. Control surface actuator command signals will be converted using a shape-preserving piecewise cubic interpolation. This was chosen because it minimizes the overshoot and oscillatory behavior that can be seen in standard cubic spline interpolation, especially in sharp or square waves. In contrast, all other measured and estimated signals will be converted using cubic spline interpolation. For this technique, signals are either up-sampled or down-sampled to the uniform 50 Hz sampling rate. If the signal is originally less than 50 Hz, it is directly up-sampled to 50 Hz. If the signal is originally greater than 50 Hz, it is resampled at the median rate of the original signal, smoothed, and then directly down-sampled to 50 Hz. A low-pass filter will be applied to smooth the data to remove higher-frequency signals that could potentially cause aliasing when down-sampling. Once the data are resampled, they will be stored in a matrix where the timing and quality of the performed flight maneuvers are assessed.

B. Data Compatibility Analysis and Correction

Data compatibility is the process that ensures that the measured flight data is physically consistent with the rigid-body equations seen in Sec. III. Any discrepancies are identified and corrected to produce a dataset that accurately represents the aircraft's dynamics. Performing this step is essential because it prevents sensor errors or inconsistencies in aerodynamic or stability parameters, thus improving the accuracy and reliability of the estimated aircraft model. Kinematic consistency analysis will be conducted using a modified form of the translational equations of motion.

$$\dot{u} = rv - qw - g\sin\theta + a_x \tag{13}$$

$$\dot{v} = pw - ru + g\cos\theta\sin\phi + a_y \tag{14}$$

$$\dot{w} = qu - pv + g\cos\theta\cos\phi + a_z \tag{15}$$

Specifically, where the non-gravitational external forces per unit mass are substituted with the measured translational acceleration components determined in the body-fixed reference frame, and the rotational kinematics are incorporated appropriately.

$$\dot{\phi} = p + (q\sin\phi + r\cos\phi)\tan\theta \tag{16}$$

$$\dot{\theta} = q\cos\phi - r\sin\phi \tag{17}$$

$$\dot{\psi} = (q\sin\phi + r\cos\phi)\sec\theta \tag{18}$$

If the reconstructed and measured signals are comparable to each other, the data can be considered to be kinematically consistent. However, if the analysis indicates that the data are kinematically inconsistent, corrective actions must be taken before proceeding. A detailed outline of these corrective actions can be found in Ref.[7].

C. Data Smoothing

Once the previous steps are completed, the flight data (excluding the control surface signals) are smoothed using another low-pass filter with a cutoff frequency of 6 Hz, which is roughly 2–3 times the fastest mode of the system (i.e., the short period mode). Although true sensor noise was filtered in the initial data processing, this second processing step filters out high-frequency aeromechanical effects (such as aeroelastic effects and unsteady aerodynamics) that are assumed to be negligible for the identified model. After this last step of data processing, the data are stored in an updated matrix to perform model identification.

V. Model Identification

Model identification begins after flight data have been processed and requires model structure determination, parameter estimation, and validation. Model determination refers to explicitly defining the mathematical structure of the model by choosing appropriate states, inputs, and terms that adequately capture the dynamics of the system. For this paper, multivariate orthogonal function (MOF) modeling will be used, following a similar methodology found in Ref. [9]. MOFs are used because they provide an efficient and numerically stable path to model complex dynamic systems. Typical polynomial models face problems from highly correlated parameters, noise, and inaccurate parameter estimation. The MOF confronts these issues by ensuring that the functions are independent of each other, modeling without redundant terms, and avoiding unnecessary complexity. After determining the terms needed in the model structure, parameter estimation is performed to find the final parameters. Specifically, the equation error method is used for the determination of the parameter estimation. In this formulation, the measured force and moment coefficients are directly equated to the modeled force and moment coefficients, where the difference between them is the error term. This method is implemented using least-squares regression. This is done to minimize the difference between the measured and modeled force and moment coefficients. The anticipated results of this paper are expected to align closely with the results of Ref.[7], and will be made available in March at the 2026 AIAA Student Conference.

VI. Conclusion

System identification from flight data is challenging due to flight test environments and programming difficulties, but it has the ability to provide accurate flight dynamics for model development. This paper has provided an overview of the fixed-wing aircraft system identification methodology using flight data. The collection of flight data is performed by a pilot on the ground executing manual maneuvers and coordinating excitation signals using custom-developed flight software. Once flight data are collected, several steps in the processing of the data are used before model identification can be performed. The model development is then performed using several determination and estimation methods used in nonlinear dynamic modeling. After a model is developed, it is then validated using the equations governing the flight and can be used in the future to predict flight data. Data acquisition systems are critical in providing high-fidelity and accurate flight data used in system identification, and the methodology used in this paper provides a DAS methodology that can be replicated to support other fixed-wing UAS platforms.

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References

[1] "PACE-Laboratory", “GitHub Source Code Repository,” https://github.com/PACE-Laboratory, 2026.

[2] Morelli, E. A., “System IDentification Programs for AirCraft (SIDPAC), Version 4.1,” NASA Langley Research Center, 2026.

[3] Morelli, E. A., “Determining Aircraft Moments of Inertia from Flight Test Data,” *Journal of Guidance, Control, and Dynamics*, Vol. 45, No. 1, 2022, pp. 4, 13. https://doi.org/10.2514/1.G006072.

[4] Morelli, E. A., “Multiple input design for real-time parameter estimation in the frequency domain,” *IFAC Proceedings Volumes*, Vol. 36, No. 16, 2003, pp. 639–644. https://doi.org/10.1016/S1474-6670(17)34833-4.

[5] Morelli, E. A., “Practical aspects of multiple-input design for aircraft system identification flight tests,” *AIAA AVIATION 2021 Forum*, American Institute of Aeronautics and Astronautics, VIRTUAL EVENT, 2021, pp. 1–25. https://doi.org/10.2514/6.2021-2795.

[6] Morelli, E. A., and Klein, V., *Aircraft System Identification: Theory and Practice*, 2nd ed., Sunflyte Enterprises, Williamsburg, Virginia, 2016, Chap. 3.

[7] Simmons, B. M., Gresham, J. L., and Woolsey, C. A., "Flight-Test System Identification Techniques and Applications for Small, Low-Cost, Fixed-Wing Aircraft," *Journal of Aircraft*, Vol. 60, No. 5, 2023, pp. 1503, 1521. https://doi.org/10.2514/1.C037260.

[8] Marquis, D. J., and Farhood, M., "Development and application of a dynamic obstacle avoidance algorithm for small fixed-wing aircraft with safety guarantees," *Control Engineering Practice*, Vol. 168, 2026, p. 106719. https://doi.org/https://doi.org/10.1016/j.conengprac.2025.106719.

[9] Morelli, E. A., "Global nonlinear aerodynamic modeling using multivariate orthogonal functions," *Journal of Aircraft*, Vol. 32, No. 2, 1995, pp. 270–277. https://doi.org/10.2514/3.46712.