

Summary of the Integrated Adaptive Wing Technology Maturation Wind Tunnel Tests of a High Aspect Ratio Commercial Transport Aircraft

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The Integrated Adaptive Wing Technology Maturation program is a cooperative effort between NASA and Boeing to demonstrate and validate control and sensing technologies necessary to enable higher aspect ratio flexible transport aircraft. The culmination of this program was a pair of wind-tunnel tests in the Transonic Dynamics Tunnel at NASA's Langley Research Center: the first occurring July through August 2024 and the second June through July 2025. Both tests utilized a semispan wind-tunnel model of a commercial transport aircraft based on an aspect ratio 13.5 version of the NASA Common Research Model. The first entry focused on open-loop aerodynamic and aeroservoelastic characterization of the wind-tunnel model and the second entry tested closed-loop control laws for drag optimization, maneuver load alleviation, gust load alleviation, and active flutter suppression. This paper summarizes the two test entries, open- and closed-loop testing results, and lessons learned that are applicable to the design and wind-tunnel testing of high aspect ratio, thin wing, commercial transport aircraft.

I. Introduction

THE Integrated Adaptive Wing Technology Maturation (IAWTM) research program is a NASA-Boeing collaborative effort that culminated in a set of two wind-tunnel tests in the Transonic Dynamics Tunnel (TDT) at NASA's Langley Research Center (LaRC). As part of the NASA Advanced Air Transport Technology (AATT) project under the Advanced Air Vehicles Program (AAVP), Boeing partnered with NASA in the IAWTM program to design, build, and test a semispan, aeroelastically scaled model to demonstrate technologies for active control of flexible, thin wings that will inform future aircraft designs. To improve aerodynamic efficiency and reduce operational costs, these next-generation transport configurations incorporate lightweight, high aspect ratio wings that are inherently more flexible than current aircraft designs, which leads to a variety of aeroservoelastic (ASE) challenges.

In the IAWTM effort, Boeing and NASA conducted research and testing to demonstrate the use of active controls to address these ASE challenges. Closed-loop controllers were developed by both Boeing and NASA teams to target several objectives deemed important for these configurations:

- 1) Drag optimization at off-nominal conditions

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- 2) Maneuver load alleviation (MLA)
- 3) Gust load alleviation (GLA)
- 4) Active flutter suppression (AFS)

Guided by the NASA Higher Aspect Ratio Optimal Wing technical challenge of enabling a 1.5 to 2 times increase in the aspect ratio of a thin, lightweight wing, the aspect ratio 13.5 variant of the NASA Common Research Model (CRM) [1, 2] was selected as the baseline configuration used for IAWTM. A test article was designed and built based on this configuration for testing in the TDT. A picture of the IAWTM wind-tunnel model installed in the TDT is shown in Fig. 1.



Fig. 1 IAWTM wind-tunnel model installed in the Transonic Dynamics Tunnel during Entry 1 [Source: NASA].

This paper provides an overview of the two wind-tunnel test entries conducted during the IAWTM research program, which are summarized in Table 1 below. A brief summary of test results is presented, with specific papers providing additional details referenced throughout.

Table 1 Summary of IAWTM wind-tunnel tests conducted in the NASA LaRC TDT.

Entry	Test	Year	Purpose
1	T679	2024	Acquire open-loop data to characterize aerodynamic and ASE models
2	T683	2025	Test closed-loop controllers targeting drag optimization, MLA, GLA, and AFS

II. NASA Transonic Dynamics Tunnel Overview

The IAWTM test article was specifically designed for testing in the NASA LaRC TDT, a world unique facility that was essential to the successful conduct of the IAWTM test campaign. The TDT is a continuous flow, single return wind tunnel with a 16×16 foot slotted test section with cropped corners, resulting in a cross-sectional area of approximately 248 square feet. This allows for the accommodation of larger-scale models, like the IAWTM. It is capable of testing at total pressures ranging from 0.01 to 1.0 atmospheres and at Mach numbers from near 0 to 1.20. Testing can be conducted in one of two test mediums: air or heavy gas (R-134a). Due to its higher density and lower speed of sound, heavy gas facilitates model aeroelastic scaling and enables testing at or near the representative Mach numbers and dynamic pressure values of nominal commercial transport aircraft at cruise altitudes and velocities. Figure 2a provides a top-down view of the TDT wind-tunnel facility and its major components. The control room identified in the figure provides direct visual access to the test section, enabling rapid identification of any model issues by a safety observer. The figure also shows two additional safety features critical for testing higher-risk aeroelastic test articles: (1) a set of four quick-actuating bypass valves that can be activated to rapidly reduce test section velocity and (2) safety screens

just upstream of the tunnel motor and fan blades to protect them from potential model debris should any liberation of components occur. These features, in addition to real-time monitoring of model instrumentation and wind tunnel system parameters, are essential for testing and risk mitigation of models like the IAWTM.

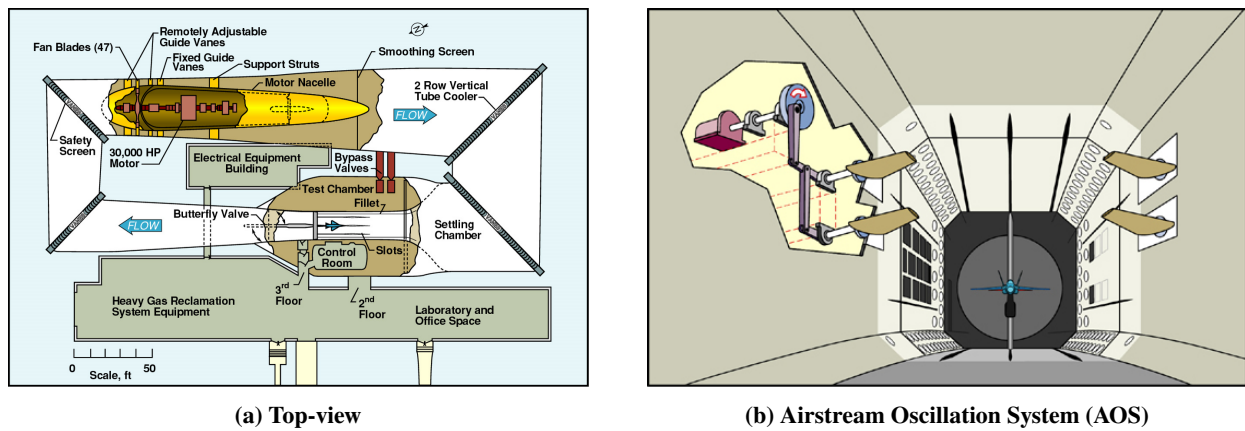


Fig. 2 NASA Langley Research Center Transonic Dynamics Tunnel [Source: NASA].

Two additional TDT capabilities utilized during the IAWTM tests were the hydraulic supply and the Airstream Oscillation System (AOS). The 3,000-psi hydraulic supply in conjunction with closed-loop control hardware allows the TDT to support active, closed-loop control of wind-tunnel test articles with multiple control effectors. The AOS, illustrated in Fig. 2b, generates gust perturbations on a model in the test section and was the single critical enabler for the open-loop gust characterization and closed-loop GLA testing for the IAWTM model. It consists of four vanes that are mounted in a biplane arrangement on the east and west sidewalls of the tunnel at the entrance to the test section. The two vanes in each pair are mechanically linked so they oscillate sinusoidally in-phase at the same frequency, with a flywheel holding that frequency constant, but the east and west vane pairs can be operated in- or out-of-phase (up to 180 degrees) relative to each other. The vanes can oscillate at variable frequencies (0 to 20 cycles per second) at a preset, manually-fixed amplitude (0 to 12 degrees); however, they are power limited, so maximum achievable tunnel dynamic pressure will be lower at the higher amplitudes and frequencies. During IAWTM testing, the AOS vane pairs were operated in-phase at a set amplitude of ± 3 degrees (6 degrees peak to peak), and were only utilized during Entry 2 due to time constraints.

The TDT test envelope in R134a gas is shown in Fig. 3. Note the blue dashed lines in the figure, which represent the Mach and dynamic pressure combinations that exist along nearly constant total pressure (H) lines. An H -line is traversed solely by changing the TDT main drive motor RPM, which is significantly more efficient time-wise than moving “vertically” within the test envelope via total pressure variations. To take advantage of this increased efficiency, the planned IAWTM test points were chosen to lie along two of those H -lines ($H = 500, 700$ psf, represented by the thick blue dashed lines).

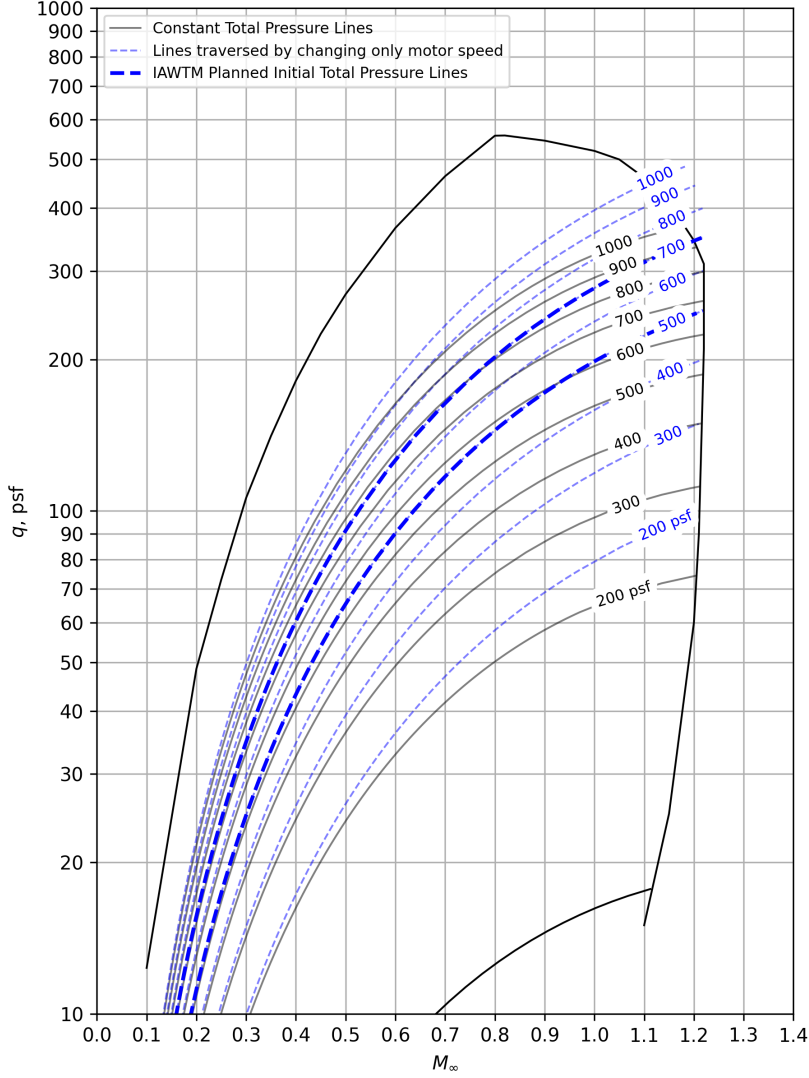


Fig. 3 TDT test envelope in R134a test medium.

III. Wind-Tunnel Test Article

The IAWTM wind-tunnel test article is a semispan, aeroelastically scaled model of the CRM13.5 airframe. An isometric CAD view of the model and its major components is shown in Fig. 4. The model consists of a metric wing, a non-metric fuselage, and a flow-through nacelle. The metric wing has been aeroelastically scaled for a nominal cruise condition: Mach 0.85, 33,000 ft altitude, and dynamic pressure of $\bar{q}_\infty = 277$ psf. The fuselage is non-metric and has been truncated to fit within the TDT test section and allow unencumbered pitching of the model. The flow-through nacelle is intended to model the impact of the nacelle weight and inertia upon the wing structural dynamics rather than real aerodynamic effects. The model also includes 10 control surfaces that span the trailing-edge of the wing.

A. Model Design

The IAWTM model was fabricated by NextGen Aeronautics using knowledge and experience from preceding aeroelastic wind-tunnel models. The primary structural component of the wing is an aluminum spar specifically designed to match the spanwise bending and torsional stiffness distributions of the nominal full-scale aircraft. Carbon fiber skin panels attach to the spar via aluminum ribs. The skin panels are broken into pods centered around each rib to minimize load transfer through the skin and allow for large aeroelastic bending and twisting. The fuselage consists of a

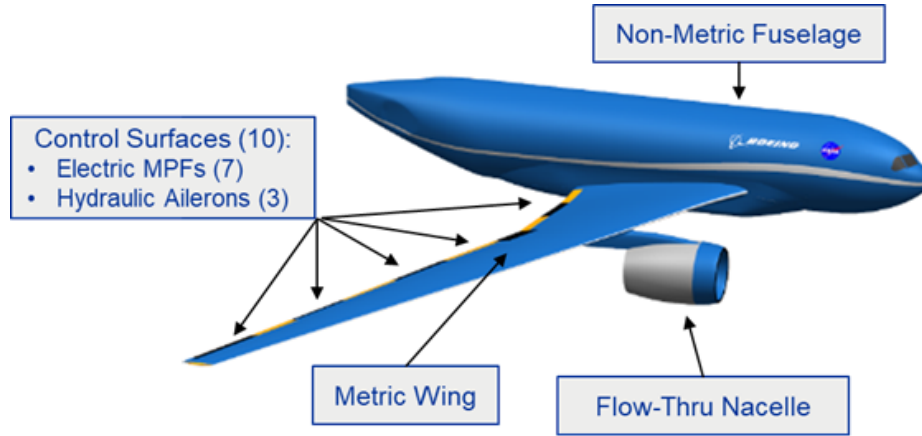


Fig. 4 IAWTM wind-tunnel model and major subcomponents.

centerline aluminum rib with additional bulkheads used to support and fasten the fiberglass skin. Additional details on the wind-tunnel model design and construction can be found in an earlier paper [3].

One design feature of particular relevance to the wind-tunnel testing is the use of three different model configurations. The desire to perform drag optimization, maneuver/gust load alleviation, and flutter suppression testing with the same model imposes different restrictions upon the test article. For open-loop aerodynamic characterization and closed-loop drag optimization, NASA Langley’s 1637S balance can be used to obtain aerodynamic force and moment measurements. However, during more risky dynamic test phases, such as gust load alleviation and flutter suppression testing, it is undesirable to keep the valuable and sensitive balance in the load path. To navigate these competing requirements, NextGen designed the IAWTM wind-tunnel model with the ability to switch between two different load paths with minimal effort.

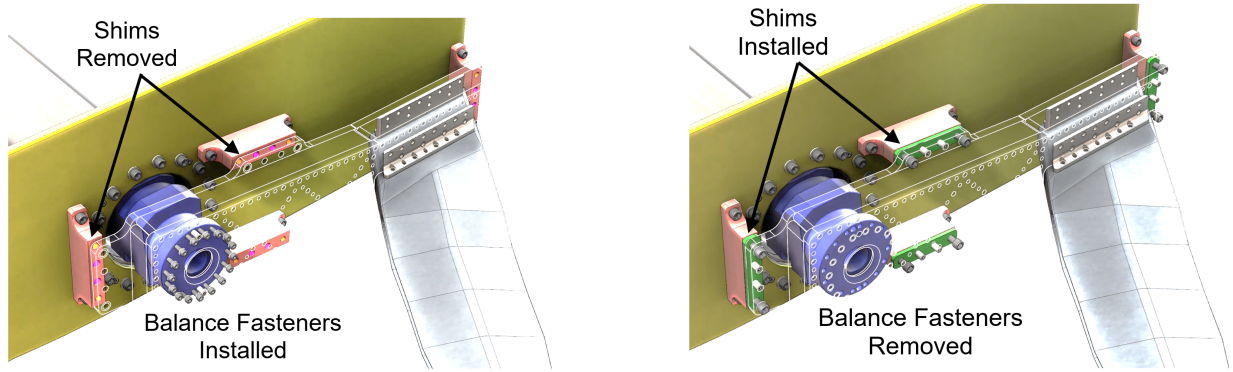
The two load paths are illustrated in Fig. 5. The first configuration is called the “shims-off” load path, where there are fasteners installed that connect the NASA 1637S balance to the mounting box assembly on the model. In this configuration, the aerodynamic loads travel through the mounting box assembly to the balance, enabling the measurement of aerodynamic forces and moments. In the second “shims-on” configuration, the balance fasteners are removed, and four shims are inserted between the mounting box assembly and the fuselage centerline rib. The installation of these shims also introduces a small lateral separation between the balance and the mounting box assembly. As a result, the loads no longer travel through the 1637S balance, and aerodynamic force and moment measurements are no longer available.

A third model configuration was designed for active flutter suppression testing. As will be shown in the flutter analysis results (Section IV.D), the predicted flutter dynamic pressure boundary for the model in the shims-on configuration is outside the IAWTM test envelope. While this is desirable from a test-safety perspective, it would make it impossible to test active flutter suppression controllers. As a result, NextGen Aeronautics designed a tip ballast with adjustable fore-aft installation locations that would change the dynamic aeroelastic stability of the model. Flutter analyses performed by NASA identified that the aft-most position would lower the flutter dynamic pressure boundary enough to allow for useful active flutter suppression testing. The flutter ballast was not intended to be used with the shims-off configuration due to the risk to the balance.

A summary of the three model configurations is given in Table 2. The nomenclature S0B0, S1B0, and S1B1, indicating shims (S) and ballast (B) off (0) or on (1) state, will be used throughout the paper to distinguish between the three different configurations.

B. Instrumentation

The IAWTM model is equipped with a variety of onboard sensors. There are 22 accelerometer channels: 13 single-axis and 3 triaxial accelerometers distributed across the wing. One of the triaxial accelerometers is located on the engine nacelle with another at the location where the engine pylon joins the wing. The third triaxial accelerometer is located on the leading-edge of the spar at the outboard wingtip. Of the remaining single-axis accelerometers, two are aligned with the x-axis to measure fore-aft bending while the rest are aligned with the vertical z-axis. Most of these



(a) Shims-off configuration

(b) Shims-on configuration

Fig. 5 Wind-tunnel model load path configurations.

Table 2 Summary of the three wind-tunnel model configurations.

Configuration	1	2	3
Objective	Drag, MLA	GLA	Flutter
Shims	NO	YES	YES
Balance on Load Path	YES	NO	NO
Tip Ballast Installed	NO	NO	YES (aft position)
Nomenclature	S0B0	S1B0	S1B1

z-axis accelerometers are located at the leading-edge of the spar, but 5 are installed at the trailing-edge for measurements of wing twist. The locations of the accelerometers are shown in Fig. 6a.

The IAWTM model also has 10 strain gauge pairs distributed equally along the span of the wing. The full bridge strain gauge pairs are installed on the upper and lower surface of the spar at the same X and Y location. NextGen machined flat areas into the spar for placement and installation of the strain gauges. The locations of the strain gauge pairs are shown in Fig. 6b.

Lastly, the IAWTM model is equipped with three fiber optic strain sensing (FOSS) strands that were placed along the leading-edge, middle, and trailing-edge of the spar. NextGen machined channels into the spar for placement and installation of the FOSS lines. FOSS data was only collected at selected test points during the two entries using an auxiliary interrogator and was not used for closed-loop control.

External to the IAWTM test article, the TDT acquired data from several additional sensors during IAWTM testing. First, when the gust vanes were operated, a “gust sniffer” probe was also installed upstream of the model, shown in Fig. 7. This gust sniffer probe measures the gust response in front of the test article and is used to simulate a forward-looking sensor, such as a light detection and ranging (LIDAR) atmospheric gust detection sensor. Many of the GLA control laws utilize the gust sniffer to demonstrate the potential benefits of emerging LIDAR sensors for predictive control. Data from a ceiling mounted laser displacement sensor that measures the deflection of the wing tip during aeroelastic excitations was also acquired, and was used to quantify the performance of gust load alleviation and active flutter suppression control laws. During the second wind-tunnel test entry, speckle pattern vinyl wrap was also applied to several upper outboard wing sections to enable digital image correlation (DIC) data acquisition for increased spatial resolution deflection data. Lastly, the TDT also provided real-time measurements of tunnel parameters such as Mach, dynamic pressure, and angle of attack that can be used for closed-loop control.

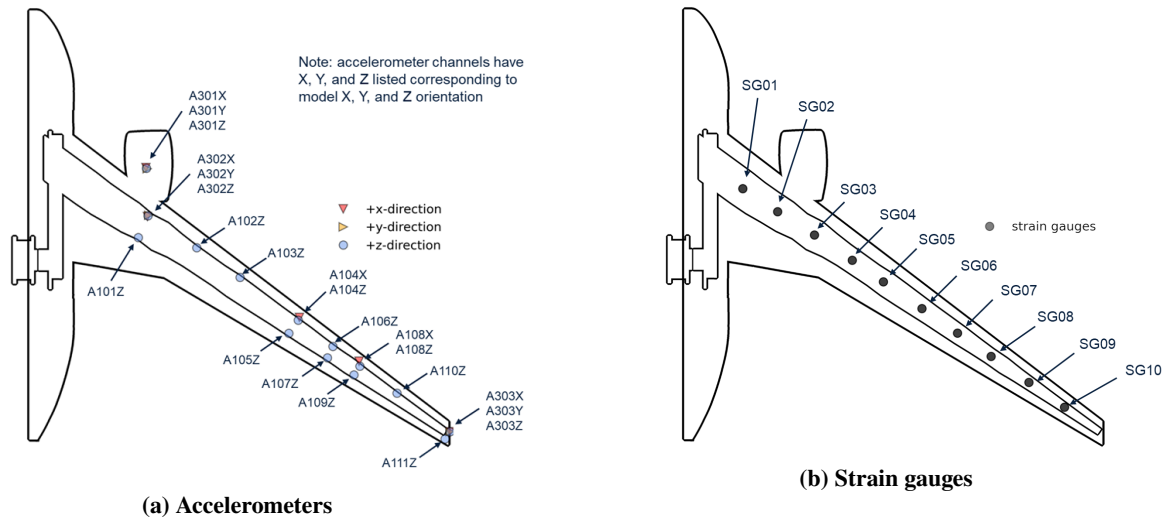


Fig. 6 Locations of accelerometers and strain gauge pairs on the wind-tunnel model.



Fig. 7 Gust sniffer probe located upstream of the IAWTM wind-tunnel model during Entry 2 [Source: NASA].

C. Control Surfaces

The wind-tunnel test article has 10 trailing-edge control surfaces: 7 electric servo-powered mini-plain flaps (MPFs) and 3 hydraulically-powered ailerons (AILs). Figure 8 depicts the layout of the surfaces and the nomenclature used to refer to each one. Bumps in the outer mold line (OML) of the wing were incorporated where necessary to accommodate these actuators and linkages. After model delivery to NASA, testing of the control subsystem revealed that freeplay, backlash, and compliance in the servos and linkages resulted in high surface freeplay; in particular, the large freeplay and linkage compliance identified in the third MPF (MPFC) necessitated a linkage redesign and remanufacture. The control subsystem design relied on accurate servo and actuator position signals, calibrated to the kinematics of the linkages, to calculate the control surface position. Servo and linkage freeplay, therefore, resulted in significant measurement uncertainty in the surface positions while under aerodynamic load. Sensors directly measuring the control surface deflections (e.g., rotary variable differential transformers, RVDTs) would have been an improved and more accurate method to measure the control surface positions during testing, but they were not included in the test article due to space constraints. While hydraulically-powered ailerons encountered significantly less freeplay and overall uncertainty than the servo-powered surfaces, it was not feasible to use hydraulics for all 10 surfaces.

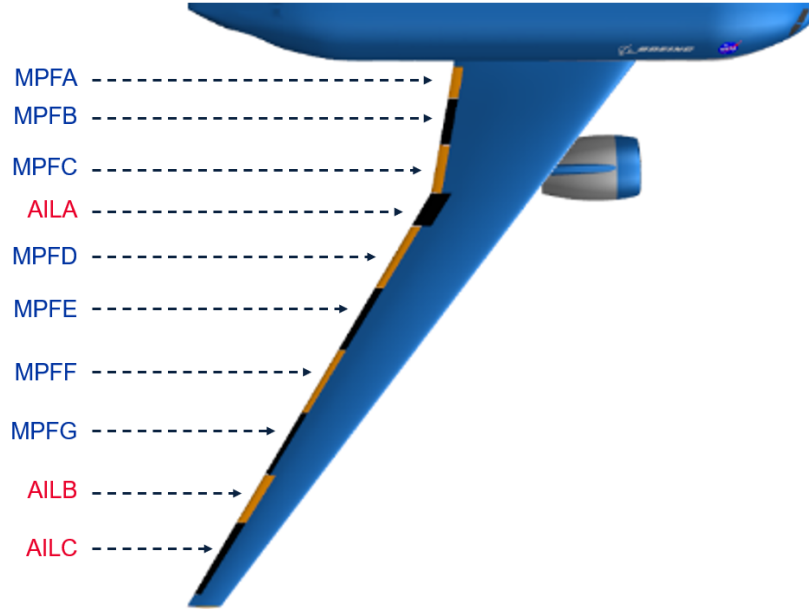


Fig. 8 Layout of the IAWTM wind-tunnel model control surfaces. Electric servo-actuated MPFs are labeled in blue and hydraulically-actuated ailerons are labeled in red.

D. Model Limits

Boeing, NASA, and NextGen Aeronautics performed extensive analyses of the IAWTM wind-tunnel model and determined the maximum allowable test conditions. These analyses included the structural limits of the wing spar, ribs, wing skins, fuselage, nacelle, and mounting box assembly. As a result, the wind-tunnel model and its components have been analyzed and approved by NASA to operate up to Mach 0.85 and $\bar{q}_\infty = 230$ psf.

Analyses conducted by Boeing and NextGen Aeronautics have also shown the control surface structure can operate with $\pm 10^\circ$ deflection limits up to Mach 0.85, $\bar{q}_\infty = 230$ psf or with deflection limits of $\pm 20^\circ$ up to $\bar{q}_\infty = 200$ psf. These control surface structural limits are further constrained by actuator torque limitations. In particular, the electric servos can be overloaded by sustained loads even if those loads fall well within the peak load limits. Boeing and NASA decided to use the servo continuous load limits to define the deflection limits for the servo-powered MPFs. Although the hydraulic actuators used for the three ailerons are sufficiently overpowered relative to the aerodynamic loads and do not restrict the aileron deflection limits, it was decided to use $\pm 10^\circ$ for the three ailerons as there was no need to command deflections larger than that during closed-loop control. A summary of the control surface deflection limits is shown in Table 3, where the restricted deflection range of most servo-powered MPFs, such as MPFC, can be seen.

Table 3 Maximum deflection range for the control surfaces.

	MPFA	MPFB	MPFC	AILA	MPFD	MPFE	MPFF	MPFG	AILB	AILC
Minimum (deg)	-7	-6	-5	-10	-7	-8	-9	-10	-10	-10
Maximum (deg)	6	5	3	10	4	5	7	10	10	10

In addition to the limits of the wind-tunnel model, there are several constraints related to test support hardware as well, namely the AOS and 1637S balance. The maximum amplitude of the AOS vanes varies with dynamic pressure, and for the planned test points of Mach 0.55, $\bar{q}_\infty = 110$ psf and Mach 0.7, $\bar{q}_\infty = 160$ psf, the AOS vanes were approved to operate at $\pm 3^\circ$ deg at frequencies up to 20 Hz.

Similarly, the 1637S balance limits only apply to the S0B0 configuration when the balance is in the load path. The 1637S balance limits are summarized below in Table 4, where the two columns correspond to the nominal design load and an approved overload limit. For IAWTM, due to the positioning of the spar ahead of the balance, the weight of the model and the large moment arm produces a significant pitching moment on the balance. Thus, under only gravity loading, the installed model exceeds the design load limits for pitching moment. NASA conducted additional testing

and analysis to approve the 1637S balance for the indicated overload values based on extensive static load predictions developed for the planned test conditions.

Table 4 NASA Balance 1637S load limits.

Component	Design Load	Approved Overload
Normal force	5,700 lbs	1,456 lbs
Axial force	500 lbs	160 lbs
Pitching moment	15,000 in-lbs	22,991 in-lbs
Rolling moment	215,000 in-lbs	252,098 in-lbs
Yawing moment	15,000 in-lbs	7,183 in-lbs

IV. Pre-Test Modeling Activities

Prior to the wind-tunnel entries, NASA and Boeing completed multiple characterization and modeling activities to better understand the dynamics of the wind-tunnel test article. This included developing models of the actuator dynamics, aerodynamics, and aeroservoelastic response. These models were leveraged for control law design and incorporated into simulation environments to evaluate controller performance.

A. Actuator Modeling

After delivery of the wind-tunnel test article, NASA performed bench tests to characterize the dynamics of both the servo-powered MPFs and hydraulic-powered ailerons. Once data was available from Entry 1 in the wind tunnel, the actuator models were further refined. The resulting second-order MPF transfer function dynamics from commanded, δ_{cmd} , to actual, δ , surface position are given in Eqn. 1 below.

$$\frac{\delta(s)}{\delta_{cmd}(s)} = K \left(\frac{a_1}{s + a_1} \right) \left(\frac{a_2}{s + a_2} \right) e^{-\tau s} \quad (1)$$

During the first entry, it was discovered that these parameters varied significantly with dynamic pressure and commanded surface amplitude. Table 5 lists the parameters for the nominal test condition of Mach 0.55, $\bar{q}_\infty = 110$ psf.

Table 5 MPF modeling parameters corresponding to Eqn. 1 for each surface at Mach 0.55, $\bar{q}_\infty = 110$ psf.

Surface	MPFA	MPFB	MPFC	MPFD	MPFE	MPFF	MPFG
K	0.9447	0.9184	0.7309	0.5376	0.7358	0.8651	0.9565
a_1	40.9708	38.6042	23.0201	46.0986	27.8485	31.0597	27.5089
a_2	124.1990	122.0204	75.5594	182.7350	199.0982	171.6039	63.4511
τ (sec)	0.0194	0.0192	0.0254	0.0199	0.0211	0.0195	0.0245

The second-order transfer function dynamics for the three ailerons are given below in Eqn. 2.

$$\frac{\delta(s)}{\delta_{cmd}(s)} = K \left(\frac{\omega_n^2}{s^2 + 2\zeta\omega_n s + \omega_n^2} \right) e^{-\tau s} \quad (2)$$

Unlike the MPFs, there was not significant aileron performance variation between test conditions, so the same parameters can be used. Table 6 lists the model parameters for the aileron surfaces.

B. Aerodynamic Modeling

In preparation for the wind-tunnel test entries, NASA conducted extensive aerodynamic analyses of the model [6–10]. NASA also conducted CFD studies of the AOS in the TDT to better quantify the aerodynamic behavior of the system [11]. Accompanying publications [12–14] provide additional details on the recent work related to aerodynamic analyses of the IAWTM wind-tunnel model.

Table 6 Aileron modeling parameters corresponding to Eqn. 2 for each surface.

Surface	K	ζ	ω_n (rad/sec)	τ (sec)
AILA	0.9792	0.5482	156	0.0027
AILB	1.0760	0.5857	59.055	0.0048
AILC	1.0280	0.3508	195.75	0.0040

C. Aeroservoelastic Modeling

Boeing and NASA performed multiple experimental characterization tests to improve the pretest structural models of the test article. These efforts included documenting a measured mass properties database, static load testing, and four ground vibration tests (GVTs). In particular, the GVTs were used to sequentially refine the finite element model (FEM) of the wind-tunnel test article. Table 7 provides a short summary of the GVT efforts.

Table 7 Summary of IAWTM wind-tunnel model GVTs.

GVT	Year	Location	Boundary	Resulting Correlated FEM
1	2022	Model Preparation Area	Backstop	AWT04
2	2023	TDT Test Section	Electric Turntable	AWT05
3	2024 (Entry 1)	TDT Test Section	Electric Turntable	AWT06
4	2025 (Entry 2)	TDT Test Section	Electric Turntable	

GVT1 was conducted in the TDT model preparation area with the model mounted to a backstop. This mounting configuration limited the effect of the boundary condition, facilitating the calibration of the test article FEM. During GVT2, the wind-tunnel model was mounted in the TDT test section to assess the structural models of the TDT support structure. Results from this test highlighted the substantial impact of the change in boundary condition from the backstop in GVT1 to the electric turntable (ETT) and oscillating turntable (OTT) support structure in GVT2, leading to an extensive FEM support structure remodeling and calibration effort by Boeing. A third GVT, conducted in the TDT test section during the first entry, identified a difference in the clocking angle of the ETT installation between GVT2 and GVT3, as well as the importance of modeling the wing pod tape and the foam between the fuselage and the tunnel wall. The incorporation of these changes led to a final release of the FEM in November 2024 (AWT06). The same ETT clocking angle was maintained for the second entry, with a reduced scope GVT4 conducted during that test entry for assessing any test-to-test differences.

With each updated release of the FEM, NASA generated ASE models of the three wind-tunnel model configurations across the test envelope [3, 15]. ZAERO [16] was used to generate 468 models for each of the three model configurations (S0B0, S1B0, and S1B1) covering Mach 0.3 – 0.95 and 50 – 300 psf in a nonuniform lattice. The ASE state space models each consist of 290 states, 45 inputs, and 152 outputs. The 45 inputs include position, rate, and acceleration for the 10 control surfaces, 4 gust vanes, and model pitch angle. The 152 outputs consist of the 22 accelerometer signals, 10 strain gauge readings, 117 FOSS channels, gust sniffer position, and lift force and pitching moment from the doublet lattice model. Data from the open-loop characterization testing acquired during the first and second entries provided experimental comparisons to these ASE model predictions.

D. Flutter Predictions

The open-loop stability of the wind-tunnel model was analyzed with Nastran [17], ZAERO, and FUN3D. Given the relatively higher computational cost for CFD flutter analyses, FUN3D was used to assess linear tools at particular tunnel conditions and model configurations of interest, while ZAERO and Nastran were used to predict the flutter boundary throughout the planned test conditions [13]. Figure 9 depicts the predicted flutter boundary for the three model configurations using the latest AWT06-based ASE models and GVT3 experimentally-estimated modal damping. The S0B0 and S1B0 flutter boundaries are sufficiently outside the maximum approved Mach and dynamic pressure test conditions, shown on the plot as “Test Article Limit”, that they were not expected to pose any risk to testing. For the S1B1 configuration, the effect of the tip ballast on the flutter boundary is clear, and flutter is predicted to occur around

Mach 0.75 for points along the $H = 700$ psf wind-off pressure line.

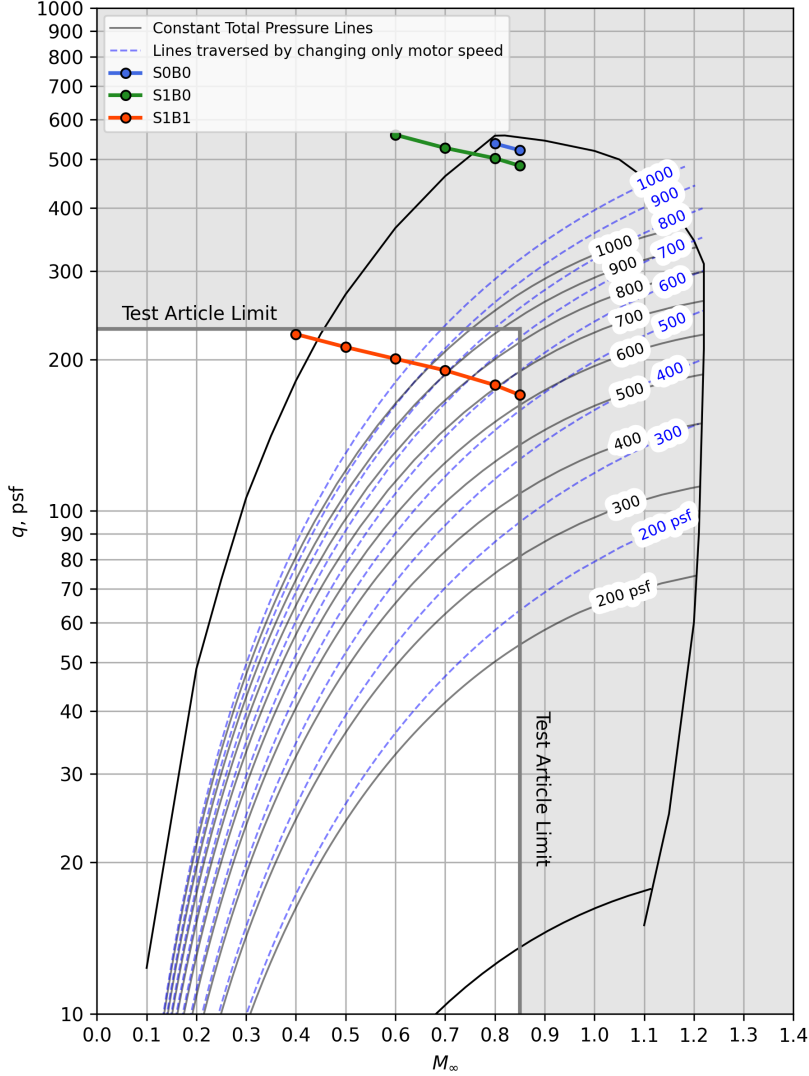


Fig. 9 Predicted flutter boundary for the three IAWTM configurations.

V. Wind Tunnel Entry 1, Test 679

The first wind-tunnel entry ran from July to August 2024. This entry focused on initial checkouts and characterization testing of the test article that were used to refine finite element and analytical models and plans for the closed-loop testing in the second entry. Figure 10 shows the planned test points for Entry 1 prior to the start of wind-on testing. For the S0B0 and S1B0 configurations, the plan was to advance up the $H = 700$ psf pressure line towards the nominal, full-scale aircraft cruise point of Mach 0.85, $\bar{q}_\infty = 230$ psf. Once these model envelope expansions were complete, aerodynamic and aeroservoelastic characterization tests would be performed at the primary test condition of Mach 0.85, $\bar{q}_\infty = 230$ psf and a secondary test condition of Mach 0.7, $\bar{q}_\infty = 161$ psf. The sequence of testing for the S1B1 flutter configuration would advance up the same H -line until the test points neared the approximate flutter boundary.

A. Initial Checkouts and Test Expansion

Initial wind-on testing was conducted using the S1B0 configuration (balance not in the load path) to reduce risk to the balance hardware as the model was first being checked out. Prior to heavy-gas testing in R134a, the model was

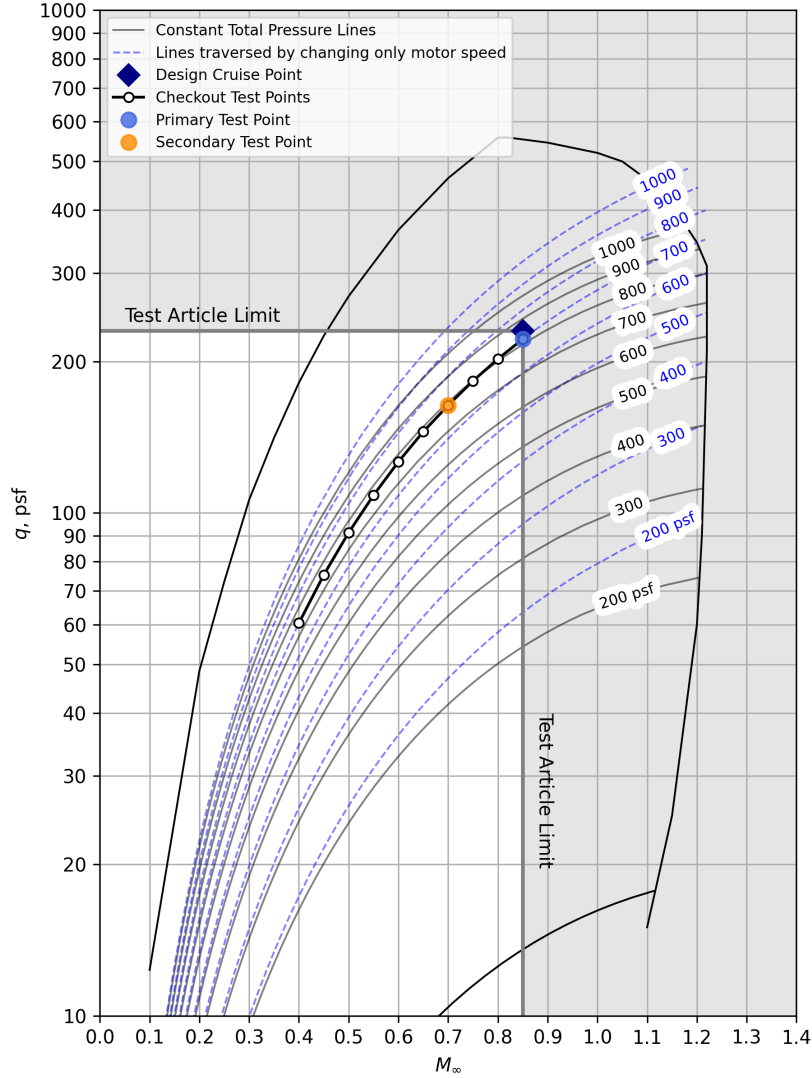


Fig. 10 Pretest planned test conditions for Entry 1.

tested up to Mach 0.3, $\bar{q}_\infty = 47$ psf in air. Limited air testing facilitates test article checkout under aerodynamic loading to confirm subsystem operation before transitioning to heavy gas, and it permits faster test section entries to make physical inspections and to address any potential model or instrumentation issues. After the successful completion of air testing, the tunnel was filled with R134a heavy gas for subsequent testing.

The heavy-gas test conditions for the initial checkout and envelope expansion are shown in Fig. 11. Following the air testing, it was decided to perform the first set of test points along the $H = 500$ psf pressure line, targeting Mach 0.8 at a lower dynamic pressure to reduce aerodynamic loading on the wing. After successful completion of the $H = 500$ psf envelope expansion, testing would proceed to the desired $H = 700$ psf pressure line.

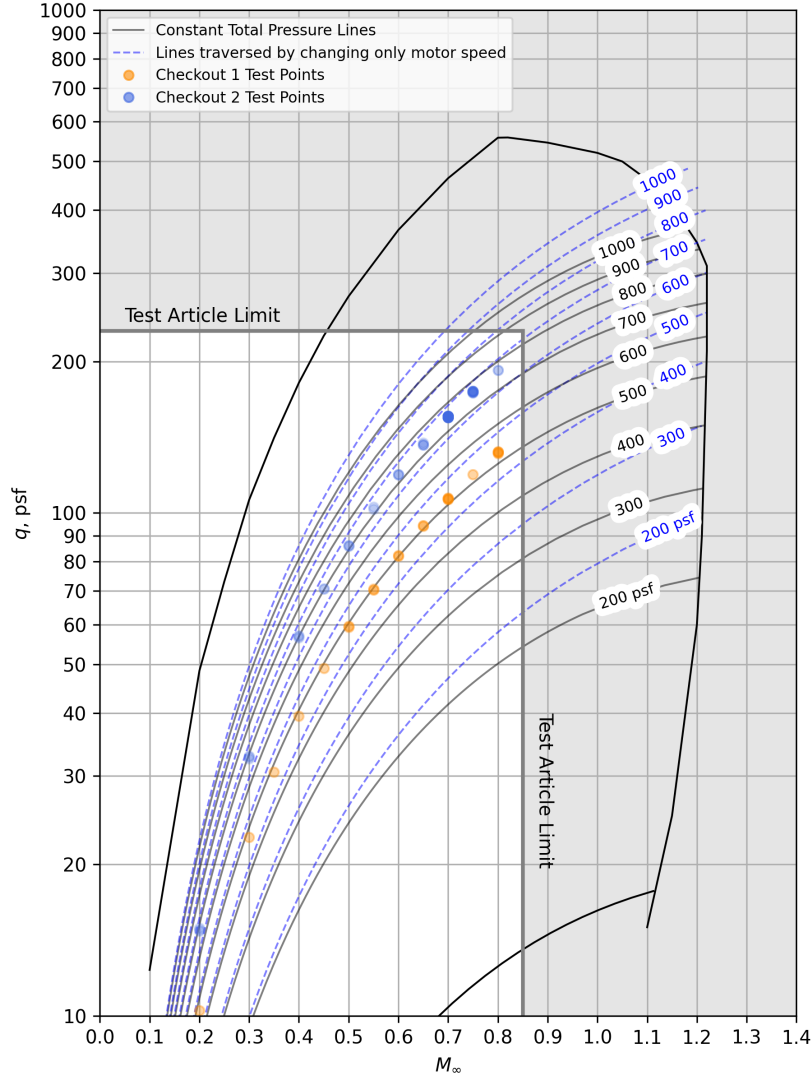


Fig. 11 Initial checkout and envelope expansion points with the S1B0 configuration in Entry 1.

1. Wing Skin Pod Damage

As the wind-tunnel test article transitioned through Mach 0.75, $\bar{q}_\infty = 120$ psf along the $H = 500$ psf pressure line, unconstrained vibration of the trailing edges of two inboard wing pods, Pods 2 and 3, occurred. The TDT bypass valves were activated, and subsequent physical inspection of the test article revealed that six fasteners along the trailing edges of Pods 2 and 3 had backed out. The resulting free-edge bending vibration caused the wing skin of Pod 3 to hit the pushrod for MPFA, which sheared and damaged the trailing-edge of the wing skin. Figure 12 shows the locations of the affected skin pods and the damage to the upper surface of Pod 3.

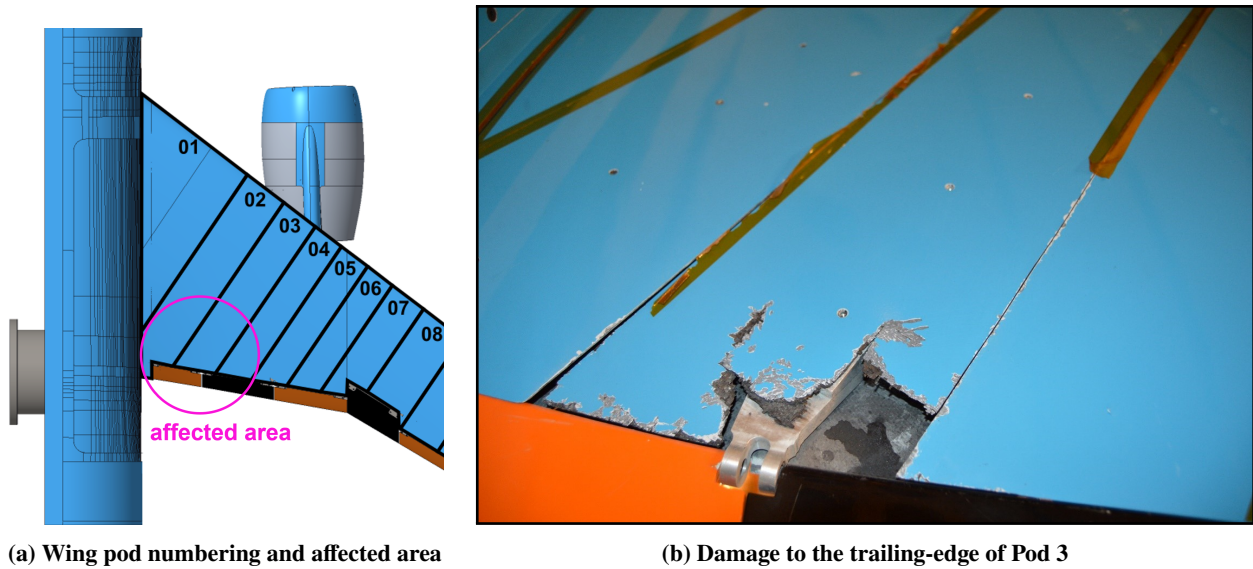


Fig. 12 Locations of Wing Pods 2 and 3 affected by the skin pod vibration event [Source: NASA].

NASA and Boeing conducted an analysis of the incident and determined that a secondary method of retention would be required for the wing skin fasteners to prevent vibration-induced backing out. The model was repaired onsite by NASA, with thread lock glue used as the secondary method of retention, and cleared for operation within a week. Once all repairs were complete, testing was restarted, and no further issues with the wing skin pods were encountered.

2. ETT Operational Limits

During the checkout and test expansion runs, the model was pitched using the TDT ETT between -2° and $+2^\circ$ at several conditions to assess model response and system performance. During testing at Mach 0.75, $\bar{q}_\infty = 175$ psf along the $H = 700$ psf pressure line, the ETT tripped offline while pitching the model at $\alpha = 1.7^\circ$.

After bringing the tunnel down and allowing the system to cool, normal operation was restored. Based on these observations, it was determined that the ETT motor had likely been overdriven during the preceding test run when the model pitch angle was adjusted in fine 0.25° increments from -2° to $+1.7^\circ$ by cycling the motor in short, rapid bursts. During that run, a combination of the test article inertial loading (in particular the high pitching moment previously discussed) and aerodynamic loading (particularly in pitch down conditions) led to a high loading on the ETT motor. The same behavior was not able to be replicated at wind-off conditions while operating the ETT motor in a similar manner.

To reduce the likelihood of overdriving the ETT motor during future runs, the operation of the ETT was modified to more smoothly transition between pitch breakpoints to reduce transient starting loads on the motor. For closed-loop drag testing in Entry 2, it was decided to change the target test points for drag testing to a lower aerodynamic loading condition, which will be discussed further in upcoming sections.

3. Maximum Test Condition

The maximum condition reached during the envelope expansion was Mach 0.8, $\bar{q}_\infty = 192$ psf with $\alpha = 0^\circ$. This is the last point along the $H = 700$ psf line shown in Fig. 11. At this point, the test article dynamic response was relatively high, and it was determined that increasing the test condition higher to the originally-planned Mach 0.85, $\bar{q}_\infty = 230$ psf target condition would not be necessary to meet the test objectives given the high model response to ambient excitation. This Mach 0.8, $\bar{q}_\infty = 192$ psf test point was the maximum condition reached during both test entries.

B. Aeroservoelastic Characterization Testing

Immediately following the initial checkouts and envelope expansion, numerous aeroservoelastic characterization tests were performed. The goal of these tests was to validate the ASE models that were developed for control system design and to investigate potential nonlinearities that would impact ASE modeling assumptions. If necessary, the ASE models could be further updated to better match experimental results in preparation for Entry 2. Note that the gust vanes

were not installed during Entry 1, so this aeroservoelastic characterization testing did not include gust effects.

A summary of the ASE characterization test points is shown below in Fig. 13. The tests were conducted at two points each along the $H = 700$ psf and $H = 500$ psf pressure lines, with the primary points of interest on the 700 psf H -line. For the Mach 0.55, $\bar{q}_\infty = 105$ psf primary test point, the ASE characterization tests were conducted at $\alpha = \{-2^\circ, 0^\circ, +2^\circ\}$ to quantify angle of attack variations. For Mach 0.7, $\bar{q}_\infty = 158$ psf, the same tests were performed at $\alpha = \{-2^\circ, 0^\circ, +1^\circ\}$, with the lower $\alpha = +1^\circ$ upper limit intended to reduce the chance that the ETT motor would be overdriven. The secondary test points at Mach 0.7, $\bar{q}_\infty = 108$ psf and Mach 0.8, $\bar{q}_\infty = 135$ psf had a reduced angle of attack schedule with $\alpha = [-2^\circ, +2^\circ]$.

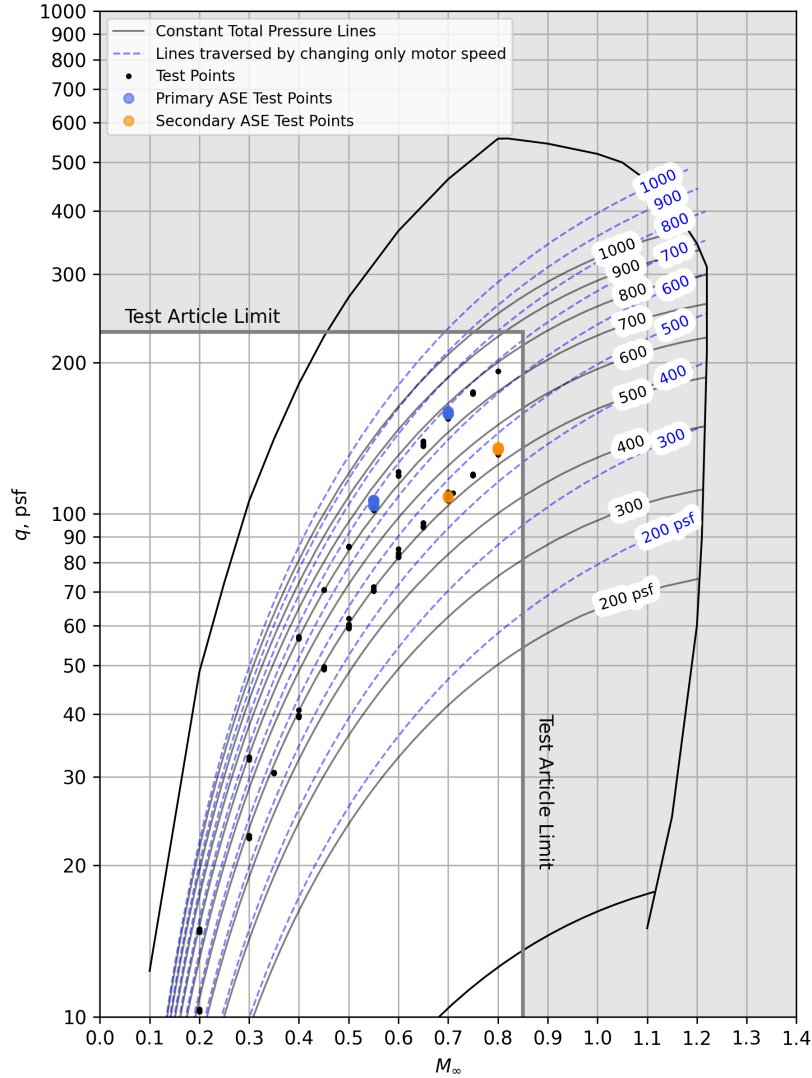


Fig. 13 Primary and secondary ASE characterization test points with the S1B0 configuration in Entry 1.

The ASE characterization testing utilized two excitation methods: logarithmic frequency sweeps and fast multisines. The logarithmic frequency sweeps used chirps from 0.1 to 20 Hz that were applied to individual MPF and aileron surfaces. Figure 14 shows the frequency sweep excitation sent through AILA during Test Point 391 at Mach 0.55, $\bar{q}_\infty = 105$ psf, $\alpha = 0^\circ$. Figure 15 shows time- and frequency-domain comparisons of the experimental test data against the ASE predictions for strain gauge SG01 (planform location shown in Fig. 6b) due to the AILA frequency sweep seen in Fig. 14. Similar comparisons were made with other priority sensors. In general, these comparisons confirmed that the ASE predictions accurately captured the experimental response, and only small updates were needed to the models based on the experimental data. A limited number of test points investigated potential nonlinear effects by

increasing frequency sweep magnitudes and exciting two surfaces at once with the in- and out-of-phase versions of the same signals.

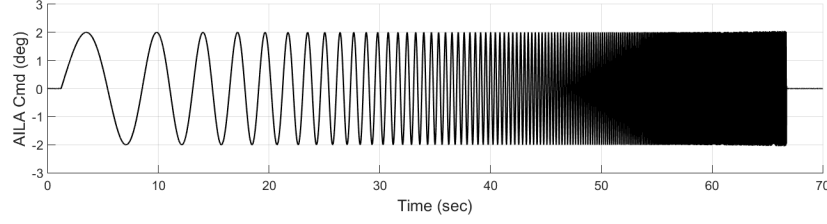
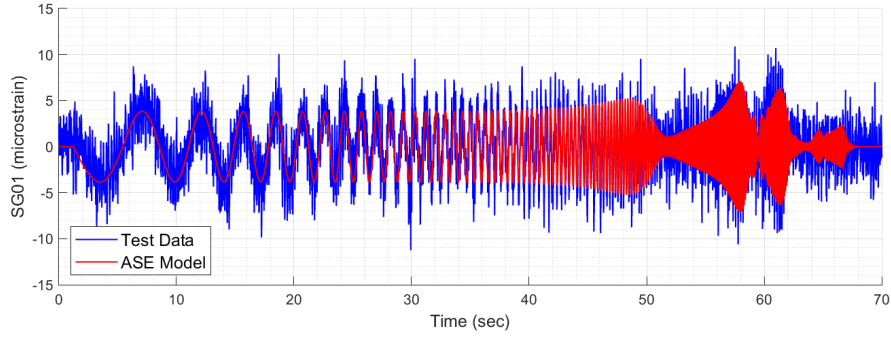
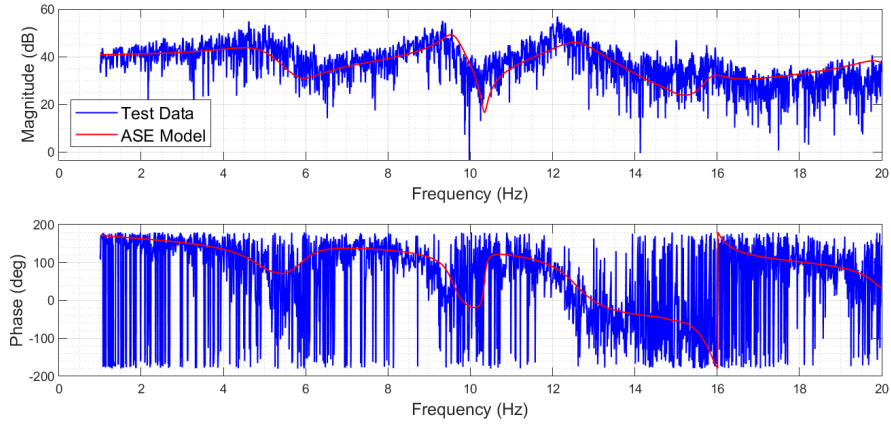


Fig. 14 Single surface logarithmic sweep on AILA for Entry 1, Test Point 391 (Mach 0.55, $\bar{q}_\infty = 105$ psf, $\alpha = 0^\circ$).



(a) Time domain comparison



(b) Frequency domain comparison

Fig. 15 Time- and frequency-domain comparison of test data and ASE model predictions for strain gauge SG01 response to the AILA sweep seen in Fig. 14.

Alongside the frequency sweeps, fast multisine excitations were used to excite all 10 surfaces at once, with each surface perturbed by the sum of multiple sinusoidal inputs spanning the desired range of frequencies [18, 19]. The sinusoidal inputs for each surface were also specifically designed to be orthogonal to the inputs applied to the remaining 9 surfaces. The intent was to provide a more efficient alternative to the single-surface frequency sweeps described above, since all the surfaces could be simultaneously excited. A subset of a multisine input is shown below in Fig. 16, where the commands sent to the 3 ailerons during Test Point 363 at Mach 0.7, $\bar{q}_\infty = 159$ psf, $\alpha = -2^\circ$ are plotted. Note that the multisine excitations were applied to all 10 surfaces for a 70 second duration. However, for clarity in this paper, Fig. 16 focuses on the first 10 seconds with only the 3 ailerons. The multisine inputs in Fig. 16 were restricted to a 3° magnitude. Additional test points with 2° and 4° magnitudes explored potential nonlinear effects.

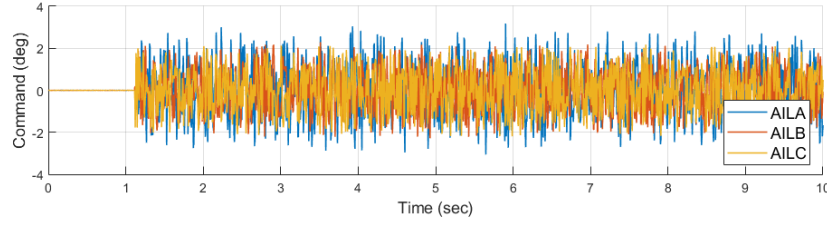


Fig. 16 Fast multisine excitation of the ailerons in Entry 1, Test Point 363 (Mach 0.7, $\bar{q}_\infty = 159$ psf, $\alpha = -2^\circ$).

During some of the multisine characterization test points, it was observed that the accelerometer A111Z signal was intermittently dropping out due to what is suspected to be contact between the sensor and the interior surface of the wing skin. Figure 17 shows an example of this intermittency during the multisine excitation applied in Test Point 363 from Fig. 16, with two peaks at 26 and 44 seconds in the data record corresponding to suspected dropouts. This issue did not appear to occur during single surface frequency sweeps and was only encountered during multisine testing. Although A111Z is one of the primary sensors for observing the wing modal response, due to this observed behavior, it was removed from the closed-loop controllers to avoid feeding back a potentially suspect signal.

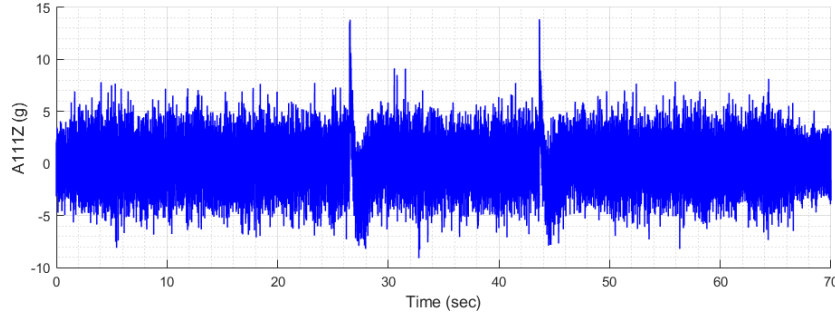


Fig. 17 Response of accelerometer A111Z to the fast multisine excitation in Entry 1, Test Point 363. The suspected intermittent dropouts can be seen around $t = 26$ and $t = 44$ seconds.

C. Flutter Expansion and Characterization Testing

Following the completion of S1B0 ASE characterization testing, the flutter ballast was installed for S1B1 configuration testing. As discussed, the flutter ballast lowered the predicted flutter boundary to test conditions within the IAWTM test envelope. Testing during Entry 1 focused on validation of the ASE models to reduce risk for the closed-loop flutter suppression testing planned for Entry 2. Test execution consisted of small, incremental increases in tunnel motor RPM to progress towards the predicted flutter boundary. The S1B1 test points are shown in Fig. 18.

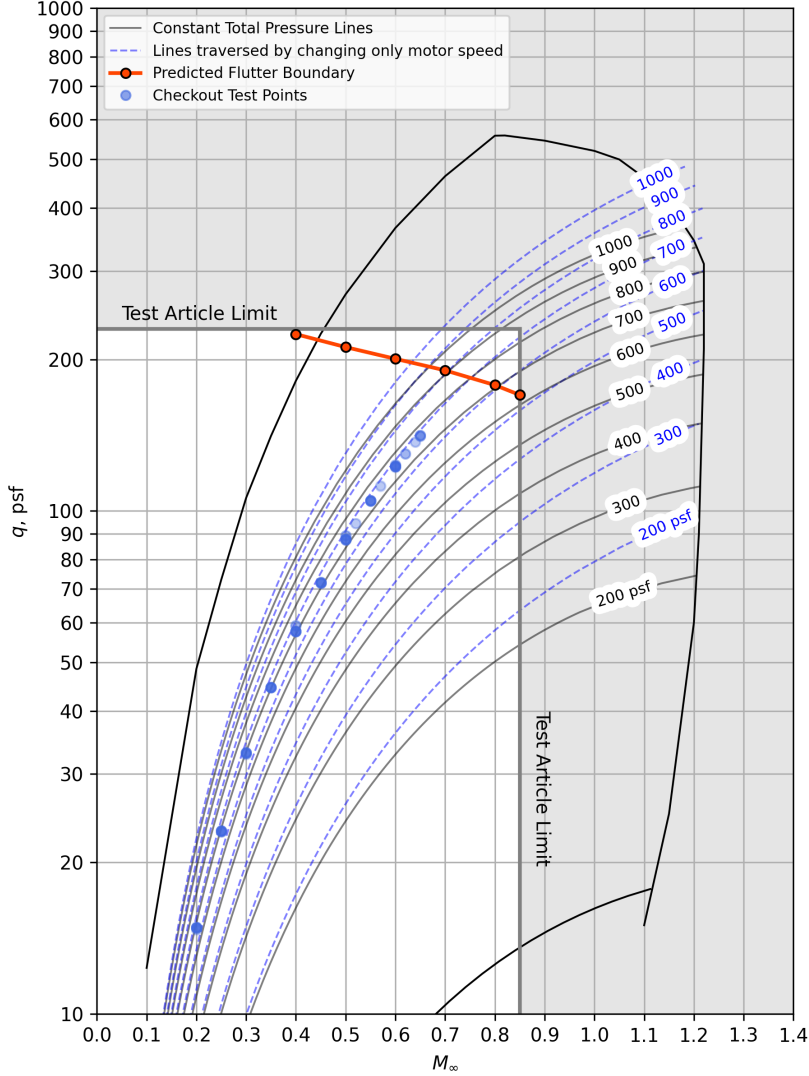


Fig. 18 Flutter checkout and ASE characterization test points with the S1B1 configuration in Entry 1.

A limited amount of ASE characterization tests were conducted at each test condition. Frequency sweeps were applied to each of the three ailerons until Mach 0.62, where only AILA was used to reduce the risk of exciting the model with higher modal energy as the predicted flutter boundary was approached. No aileron frequency sweeps were performed at Mach 0.65 for the same reason. Multisines of all 10 surfaces were conducted at each test condition up to Mach 0.65. These results were used to validate the AWT06 ASE models. Note that the angle of attack was held fixed at 0° for all test conditions.

Testing in the S1B1 configuration reached Mach 0.65, $\bar{q}_\infty = 140$ psf, which was determined to be close enough to the predicted flutter boundary for this open-loop characterization phase of testing without risking the test article.

D. Aerodynamic Characterization Testing

The last phase of testing during Entry 1 utilized the S0B0 configuration with the NASA 1637S balance in the load path to acquire aerodynamic performance data. The goal of testing in this configuration was to validate the aerodynamic models developed by NASA and collect data that would assist the planned drag optimization and maneuver load alleviation testing.

A summary of the aerodynamic characterization test points is shown below in Fig. 19. These conditions were chosen to overlap with the S1B0 ASE characterization testing, only removing the Mach 0.8, $\bar{q}_\infty = 130$ psf target condition on the $H = 500$ psf pressure line and adding a reduced set of points at Mach 0.55, $\bar{q}_\infty = 73$ psf. The majority of the test

points were conducted at Mach 0.55, $\bar{q}_\infty = 107$ psf. At this condition, the unsteadiness due to turbulence was lower relative to Mach 0.8, and the likelihood of overdriving the ETT motor with varying pitch commands during planned Entry 2 closed-loop testing was lower. These considerations were deemed more important than reaching the originally planned target condition with higher aerodynamic loading and Mach number.

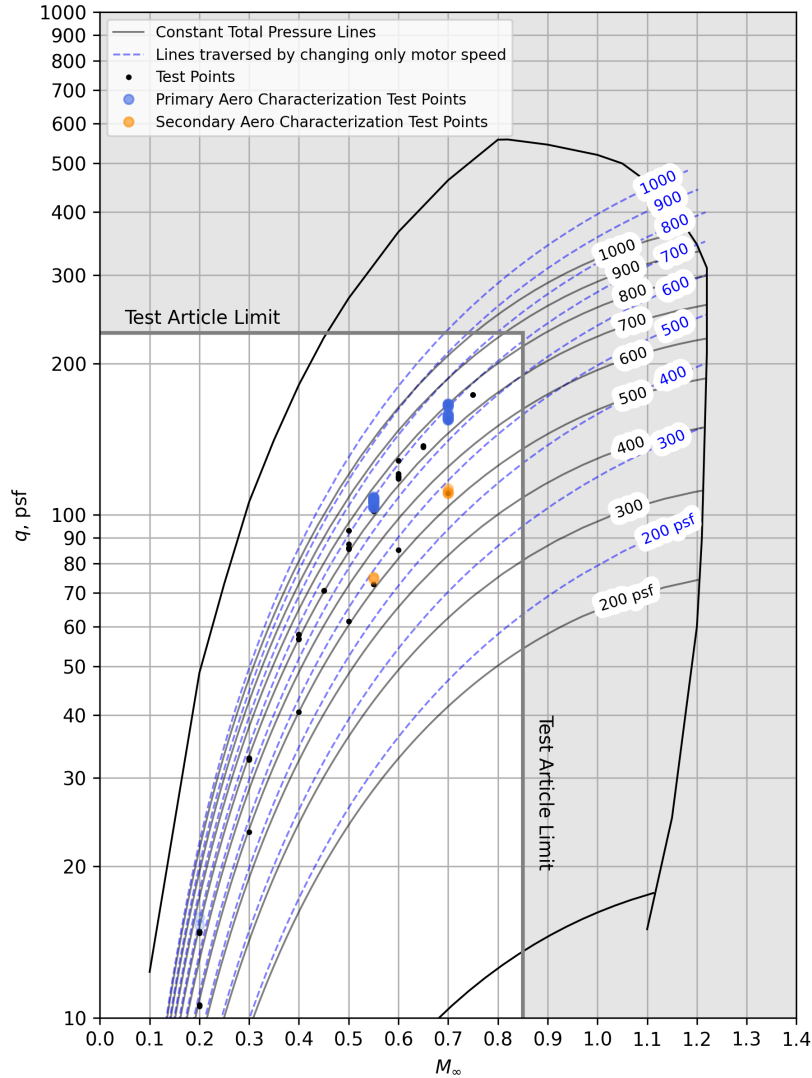


Fig. 19 Primary and secondary aerodynamic characterization test points with the S0B0 configuration in Entry 1.

The aerodynamic characterization testing consisted of three test sequences, each occurring over a long duration with low rate commands to the surfaces to avoid exciting aeroelastic modes. First, individual surfaces were moved in discrete “pitch-pause” sweeps that ramped surfaces to setpoints in the specified deflection schedule, holding each deflection constant for 7 seconds to allow any transients to dampen out. This is illustrated in Fig. 20 with the pitch-pause sweep for AILA during Test Point 611 at Mach 0.7, $\bar{q}_\infty = 156$ psf, $\alpha = 0^\circ$. In this pitch-pause sweep, AILA moved between 0° , $\pm 3^\circ$, $\pm 6^\circ$, and $\pm 10^\circ$. Sweeps were tailored for each surface to fall within the deflection limits specified in Table 3. For example, the MPFC sweep only reached 0° , $\pm 3^\circ$, and -5° due to the restrictive surface limits.

The second test sequence was a slower version of the fast multisines used for ASE characterization testing. In the “slow” multisines, all 10 surfaces were simultaneously excited at a different orthogonal frequency. Unlike the fast multisines, each surface was only excited at one frequency. This can be seen in Fig. 23, where MPFA oscillated at roughly 0.0022 Hz while MPFG oscillated at roughly 0.012 Hz. These low frequency sinusoidal responses were intended to minimize unsteady aeroelastic response.

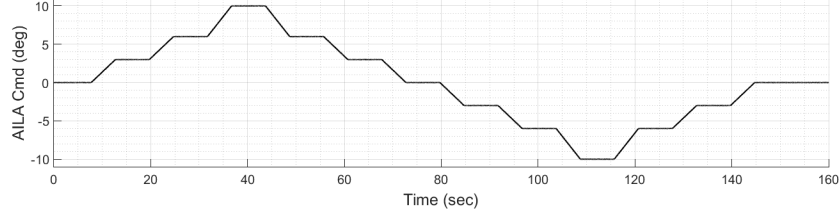


Fig. 20 Single surface pitch-pause test on AILA during Entry 1, Test Point 611 (Mach 0.7, $\bar{q}_\infty = 156$ psf, $\alpha = 0^\circ$).

The third test sequence was a “design of experiments” (DOE) sweep that was similar in execution to the pitch-pause sweeps. Each surface was ramped to a set point, held constant for 8 seconds, and then moved to a different set point; however, all 10 surfaces moved simultaneously. The set points for each surface were also not the same structured sequence as was used in the pitch-pause tests, but they were optimized to have the same number of breakpoints for comparison with the pitch-pause data. Figure 21 shows the commands for the three ailerons during the DOE sequence at Test Point 624 (Mach 0.7, $\bar{q}_\infty = 156$ psf, $\alpha = 0^\circ$). Detailed comparisons of the three methods can be found in Ref. [12].

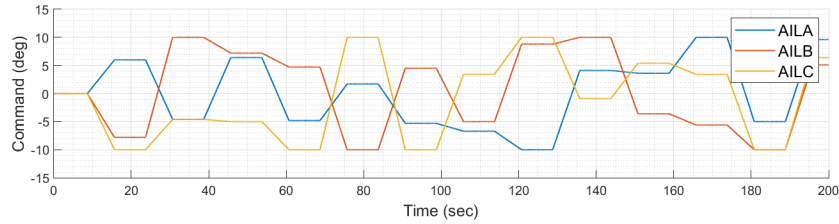


Fig. 21 DOE sweep commands for the three ailerons during Entry 1, Test Point 624 (Mach 0.7, $\bar{q}_\infty = 156$ psf, $\alpha = 0^\circ$).

The data collected from these aerodynamic characterization test sequences were used to construct an empirical aerodynamic model. One important observation from the data was that the signal-to-noise ratio (SNR) in the balance measurements varied by component and could adversely impact the aerodynamic model. For example, lift coefficient SNR was relatively high since the lift magnitude was relatively high, but the drag coefficient SNR was quite low due to the low signal and relatively high unsteadiness. Figure 22 shows the lift and drag coefficient histograms for control surface neutral measurements ($\delta = 0^\circ$) during Test Point 662 (Mach 0.55, $\bar{q}_\infty = 107$ psf, $\alpha = 0^\circ$). The mean of the lift coefficient signal was $C_L = 0.3066$, while the standard deviation was $\sigma_{C_L} = 0.0058$. Conversely, the mean of the drag coefficient signal was $C_D = 0.0161$, but the standard deviation was $\sigma_{C_D} = 0.0039$. It was therefore difficult to observe trends in the drag measurements in real time during test execution before processing the test data.

1. MPF Behavior

During wind-on testing, there were several issues identified with the servo-powered MPFs. Most relevant for subsequent testing, several servos experienced overloading/overheating during longer duration test runs. Figure 23 illustrates this behavior during the slow multisines conducted at Test Point 676 (Mach 0.7, $\bar{q}_\infty = 163$ psf, $\alpha = -2^\circ$). Figure 23a shows a normal position signal time history for MPFA, while Fig. 23b shows a position signal dropout for MPFG at approximately $t = 672$ seconds. To minimize the chance of servo overheating, subsequent testing and Entry 2 plans were adjusted to reduce long duration MPF loading.

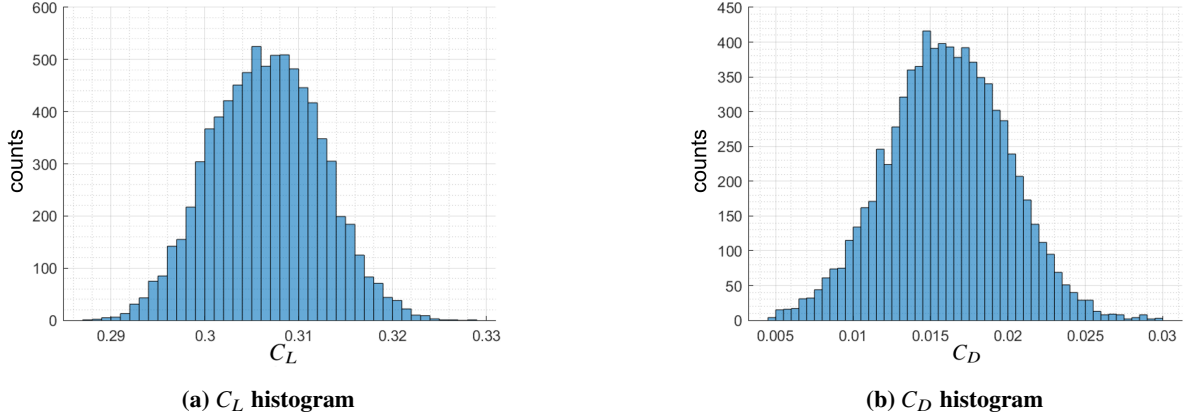


Fig. 22 Lift and drag measurement distributions for $\delta = 0^\circ$ during Entry 1, Test Point 662 (Mach 0.55, $\bar{q}_\infty = 107$ psf, $\alpha = 0^\circ$).

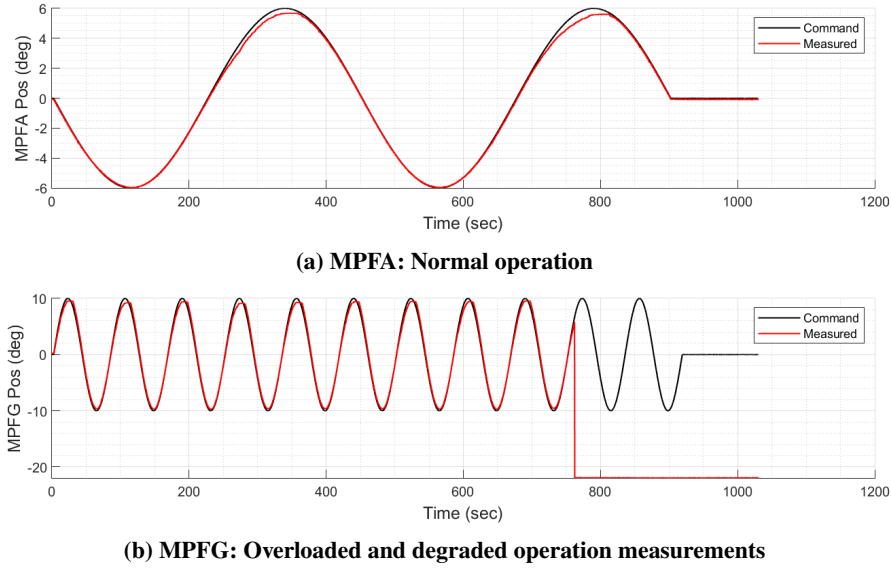


Fig. 23 Inconsistent MPF behavior during slow multisines during Entry 1, Test Point 676 (Mach 0.7, $\bar{q}_\infty = 163$ psf, $\alpha = -2^\circ$).

2. Aerodynamic Model Fitting

With the aerodynamic characterization test data, NASA performed a least-squares fit to the time-averaged lift and drag data to develop an aerodynamic model. The model was only developed for the Mach 0.55, $\bar{q}_\infty \approx 106$ psf test condition since that condition had the most complete data set. Only control surface pitch-pause sweep data were used for the control surface aerodynamic model fitting. The lift and drag equations are given below:

$$C_L = C_{L_0} + C_{L_\alpha} \alpha + C_{L_{\alpha^2}} \alpha^2 + \sum_{i=1}^9 \left(C_{L_{\delta_i}}(\alpha) \delta_i + C_{L_{\delta_i^2}}(\alpha) \delta_i^2 \right) \quad (3)$$

$$C_D = C_{D_0} + C_{D_\alpha} \alpha + C_{D_{\alpha^2}} \alpha^2 + \sum_{i=1}^9 \left(C_{D_{\delta_i}}(\alpha) \delta_i + C_{D_{\delta_i^2}}(\alpha) \delta_i^2 \right) \quad (4)$$

where $i = \{1, \dots, 9\}$ corresponds to the 9 control surfaces except for MPFC. MPFC was dropped from the modeling

effort because, as noted in Sec. ??, it was decided to avoid further use of this surface due to its poor performance in Entry 1 aerodynamic characterization testing.

A comparison of the aerodynamic model versus the experimental data is shown below in Fig. 24. The test data was collected during Test Point 662 (Mach 0.55, $\bar{q}_\infty = 107$ psf, $\alpha = 0^\circ$), which is an AILA pitch pause sweep – the same sweep shown earlier in Fig. 20. The plots compare the distribution of test data at each commanded deflection against the predicted value from the aerodynamic model. It should be noted that for the drag coefficient plot in Fig. 24b, the error bar upper and lower limits correspond to $0.05\text{-}\sigma$ rather than $1\text{-}\sigma$ because the standard deviation was so large, making it difficult to see the trend in the mean drag coefficient against commanded position. Figure 22b previously illustrated the large distribution tails on the drag measurements. Despite the higher noise, the aerodynamic model accurately predicts the mean lift and drag coefficient values when compared against the experimental data. Additional details on the aerodynamic modeling and comparison to the experimental test data can be found in accompanying papers [12, 20].

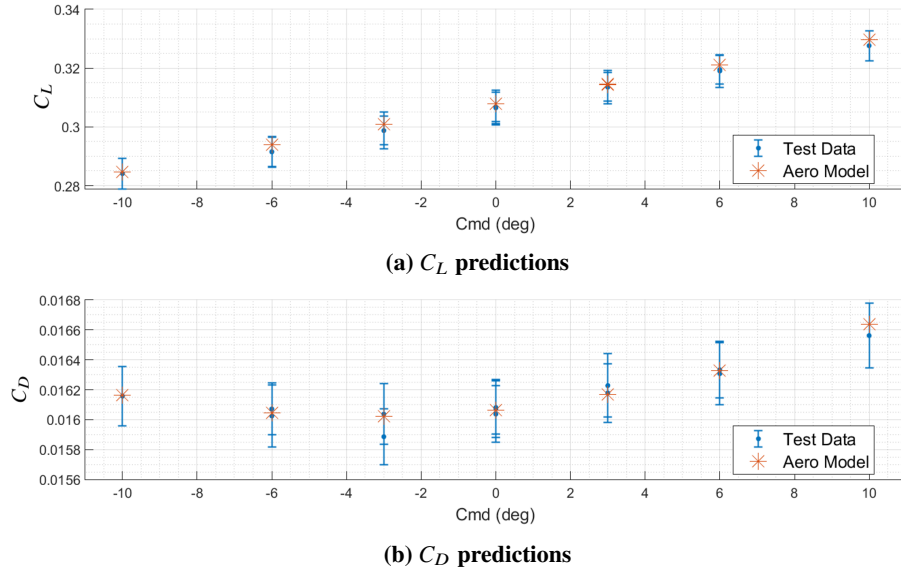


Fig. 24 Comparison of test and aerodynamic model (Eq. 3) for the AILA pitch-pause sweep during Entry 1, Test Point 662 (Mach 0.55, $\bar{q}_\infty = 107$ psf, $\alpha = 0^\circ$). Experimental standard deviations shown with error bars (drag coefficient standard deviation is shown as $0.05\text{-}\sigma$ rather than the $1\text{-}\sigma$ to make the plots visible due to the large noise).

E. Observations from Test Entry 1

The aerodynamic characterization testing in the S0B0 configuration was the final test phase during Entry 1. A number of observations drawn from the test conduct were used to adjust the Entry 2 test plan:

- The originally planned Mach 0.85, $\bar{q}_\infty = 230$ psf test point from Fig. 10 was not necessary to achieve the test objectives, and high unsteadiness and test article response would have likely led to lower quality data. Instead, the primary test point was switched to Mach 0.55, $\bar{q}_\infty = 107$ psf, and Mach 0.7, $\bar{q}_\infty = 161$ psf was kept as the secondary test point.
- MPF performance was relatively poor. Servo overloading and signal offsets could be problematic during higher-risk testing, where high bandwidth, precision control is necessary. The MPFs were removed from GLA and AFS test plans, but were still employed in testing with the S0B0 configuration. Due to its inconsistent performance, MPFC was disabled for all future closed-loop tests.
- AILB performance demonstrated inconsistent response. AILB was removed from active flutter suppression controllers, and alternate control laws were developed for gust load alleviation to avoid use of AILB.
- Wingtip accelerometer A11Z demonstrated intermittency during higher-bandwidth control surface commands. Control laws were redesigned to remove that accelerometer from feedback control.
- Low drag SNR made it difficult to observe real-time trends. Closed-loop control will require a low-pass filter on the signal.

- The S1B1 flutter configuration approached the predicted flutter boundary and confirmed the proximity of the actual boundary to the predicted one.

VI. Wind Tunnel Entry 2, Test 683

The second IAWTM wind-tunnel entry was conducted from June to July 2025. This entry focused on open-loop gust characterization and closed-loop controller testing. There were also some limited repeats of Entry 1 test points for comparison and validation purposes. Figure 25 shows the planned test points for Entry 2 prior to the start of wind-on testing.

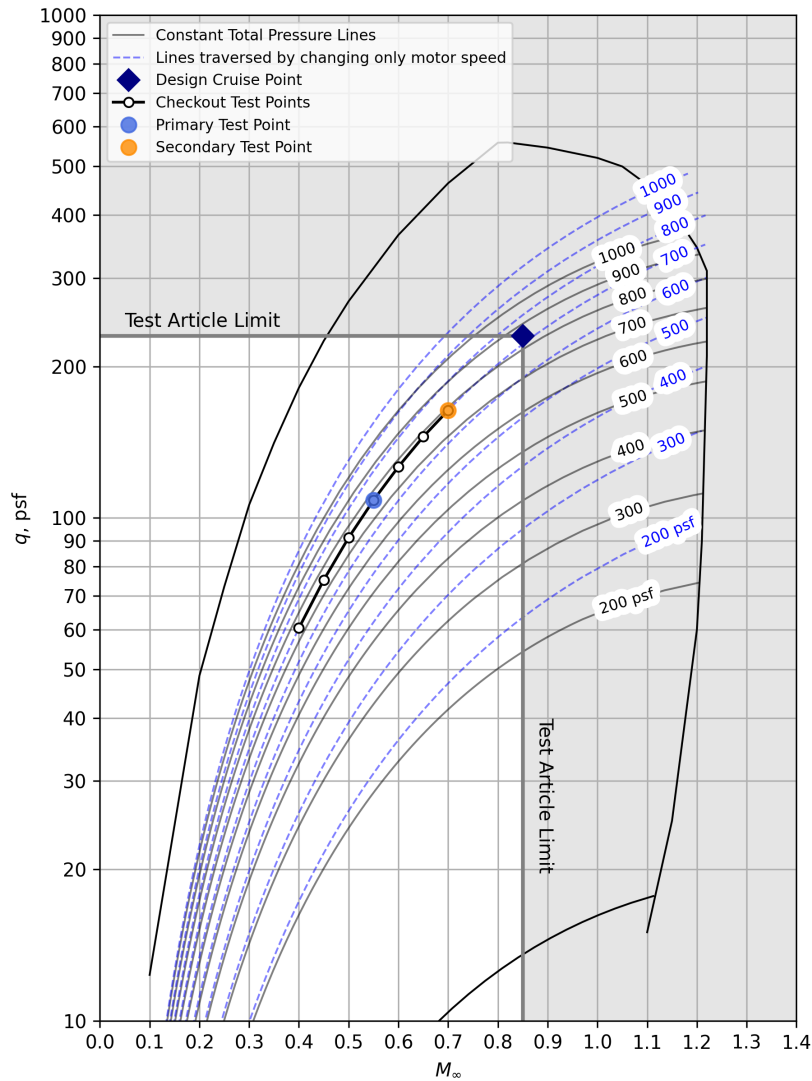


Fig. 25 Pretest planned test conditions for Entry 2.

A. Aerodynamic Characterization Testing

The first phase of testing in Entry 2 focused on limited repeats of the aerodynamic characterization testing conducted during the first wind-tunnel entry. One test-to-test difference was an adjustment of the hydraulic line routing to reduce its secondary load path effect and minimize any potential impact on fouling the balance load path. Figure 26 depicts the aerodynamic characterization test points in the S0B0 configuration during Entry 2.

The aerodynamic testing targeted the primary test point of Mach 0.55, $\bar{q}_\infty = 107$ psf and the secondary test point of

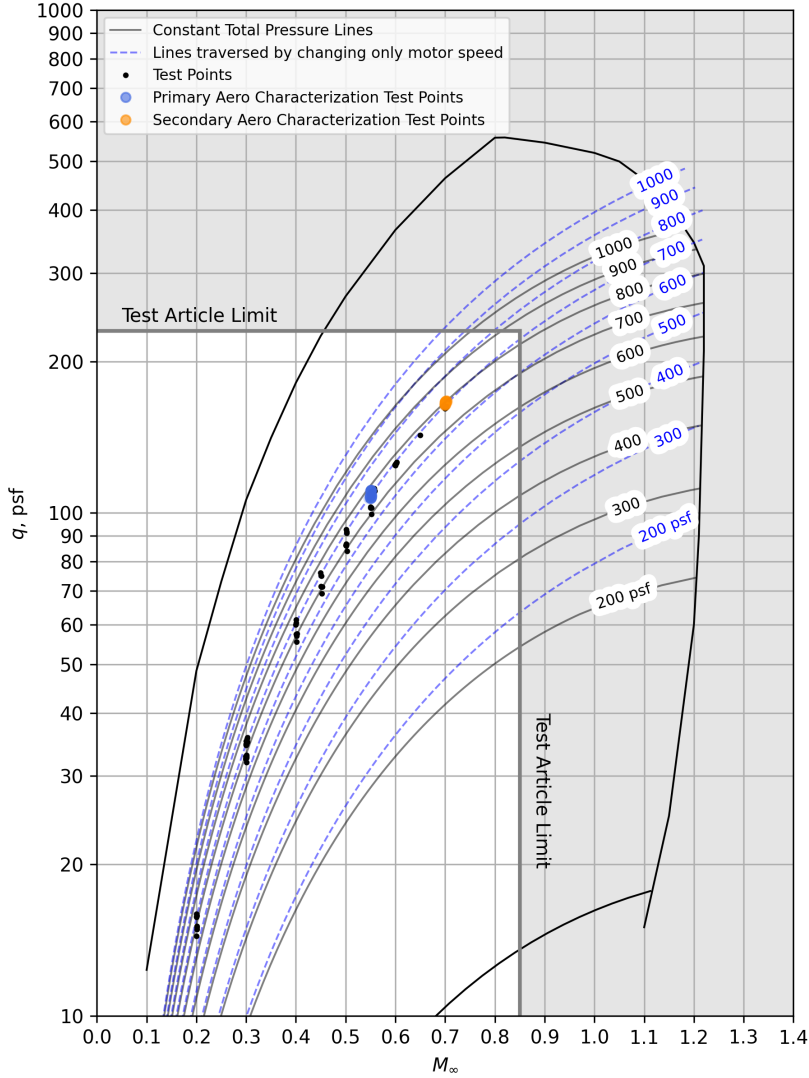


Fig. 26 Aerodynamic characterization test points with the S0B0 configuration in Entry 2.

Mach 0.7, $\bar{q}_\infty = 163$ psf. For the primary test point, pitch-pause sweep tests were conducted at $\alpha = \{-2^\circ, 0^\circ, +2^\circ, +3^\circ\}$ for all surfaces except MPFC. The $\alpha = +3^\circ$ data point was new and intended to explore any nonlinearities at higher angle of attack. Slow multisines were conducted with the 3 hydraulic ailerons, but they did not include the servo-powered MPFs. Design of experiment tests were omitted from Entry 2. There was also a small amount of aerodynamic interaction testing with AILA and MPFA to neighboring surfaces. The testing at the secondary test point covers the same $\alpha = \{-2^\circ, 0^\circ, +1^\circ\}$ from Entry 1. At both the primary and secondary test points, there was a limited amount of ASE logarithmic sweeps and fast multisines conducted with the three ailerons. This was new for Entry 2 and had not been conducted during Entry 1.

Figure 27 compares the lift and drag force measurements between the two entries. These test-to-test comparisons of the balance data indicate the relatively large differences in drag, which were likely the result of balance fouling by the hydraulic lines during Entry 1, which was improved during the Entry 2 installation. As a result of this visible offset, NASA fit a new aerodynamic model to the Entry 2 data for use in drag optimization control law design and evaluations.

B. Drag Optimization Testing

After completing the aerodynamic characterization testing, both NASA and Boeing drag optimization control laws were tested at the primary test point (Mach 0.55, $\bar{q}_\infty = 107$ psf) shown in Fig. 26. These closed-loop tests required

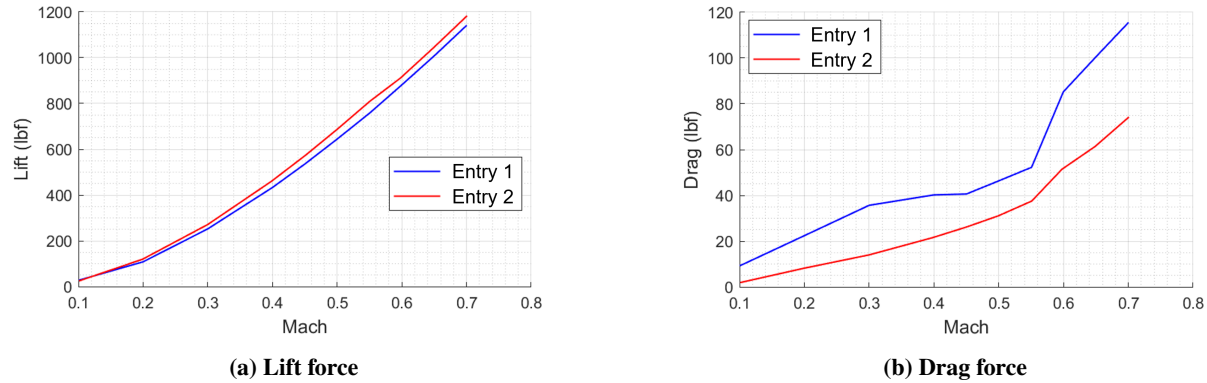


Fig. 27 Comparison of Entry 1 and Entry 2 aerodynamic data at $\alpha = 0^\circ$.

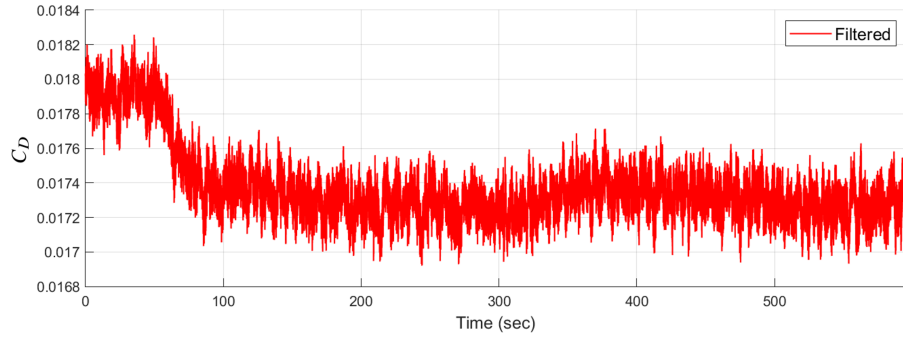
the use of the balance in the S0B0 configuration to obtain lift and drag measurements. Just like with the open-loop aerodynamic characterization testing in the two entries, the drag optimization control laws were specifically designed to minimize the risk of exciting aeroelastic modes. The control surface commands were rate limited to $0.5^\circ/\text{sec}$ to slow down the response. This also allowed NASA and Boeing test personnel to monitor the balance loads and give adequate time to terminate the tests if balance loads approached the limits discussed in Table 4. The drag optimization tests required long durations (≥ 10 minutes), so a significant amount of test time was needed to acquire each point. An accompanying paper provides additional details on the control law descriptions and their corresponding results [20]. Just like with the aerodynamic characterization testing, the level of noise in the drag measurements complicated design and analysis of the controllers. A high-level summary of the controllers and the results is described below.

The Boeing drag optimization control law utilized a peak-seeking control architecture to perform model-free optimization on drag measurements. In order to reduce the complexity of the problem and improve tractability, the MPFs were paired into pseudo-surface groupings of MPFA+MPFB, MPFD+MPFE, and MPFF+MPFG, with AILA operating by itself. AILB and AILC were skipped due to low control effectiveness and reversals at the test conditions of interest. The control surfaces were limited to either $\pm 5^\circ$ or the limits in Table 3, whichever was the more restrictive. The peak-seeking drag controller did not explicitly attempt to maintain lift, so a second parallel lift controller was created in order to maintain constant lift to simulate longitudinal control laws for elevator and throttle on a full-scale aircraft. This lift controller produced angle of attack commands that were displayed for the human TDT operator, who then pitched the wind-tunnel model to the desired angle.

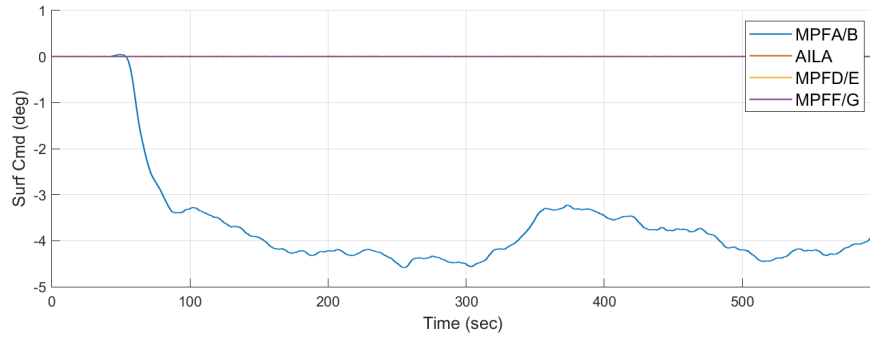
The Boeing drag optimization controller showed mixed performance. When the peak-seeking drag optimization controller was limited to a subset of the pseudo-surface groupings and the lift controller was deactivated, the controller demonstrated a clear reduction in drag. An example of this performance at the primary test condition is shown in Fig. 28. During this test point, only the MPFA+MPFB grouping was activated while the lift controller was deactivated. As a result, the controller moved the MPFs in the direction that would reduce drag by dumping lift. This produced a roughly 3% reduction in drag, but at the cost of lift (not shown). Other test points activated more of the surfaces and demonstrated a reduction in drag, but the lift controller was still deactivated. Once the lift controller was activated, the system did not converge towards a steady-state solution. This could have been due to an overly aggressive set of lift controller gains, but there was insufficient time remaining to modify and perform repeat tests.

The NASA drag optimization control law performed real-time system identification and optimization to reduce drag while maintaining constant lift [21, 22]. The controller utilized small control surface perturbations to excite the model and identify the sensitivities of the lift and drag measurements. This system identification process constructed an onboard surrogate aerodynamic model, which was then used for real-time optimization. The controller produced control surface commands sent to the actuators and angle of attack commands that were sent to the human TDT operator controlling the wind-tunnel model angle of attack.

The NASA drag optimization control law was able to successfully demonstrate a reduction in drag while maintaining constant lift. The results for Test Points 513-525 are shown in Figures 29 and 30. The control surface positions in Fig. 30b are shown inboard (MPFA) to outboard (AILC), with MPFC disabled (Flap Index = 3). The NASA drag optimization controller operated over multiple test points in an iterative manner to converge towards a solution. Over the course of the optimization process, the mean drag coefficient demonstrated a 12% reduction, while the lift coefficient



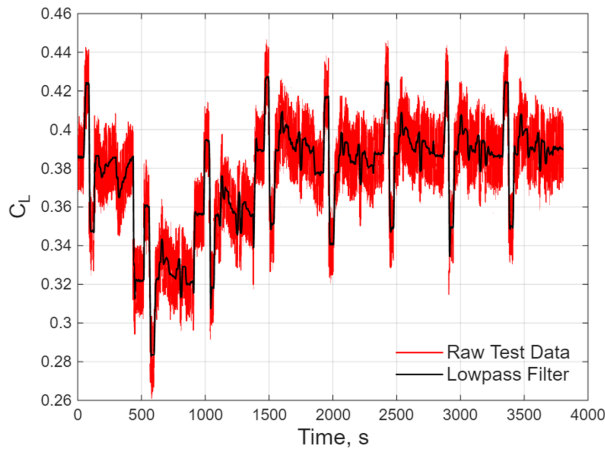
(a) Filtered drag coefficient measurements



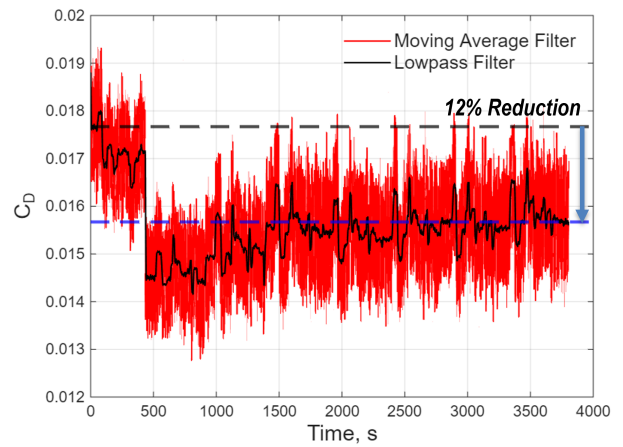
(b) Control surface commands

Fig. 28 Drag coefficient and control surface commands for the Boeing drag controller during Entry 2, Test Point 662 (Mach 0.55, $\bar{q}_\infty = 107$ psf, $\alpha = 0^\circ$).

remained steady.



(a) Lift coefficient



(b) Drag coefficient

Fig. 29 Lift and drag measurements for the NASA drag controller during Entry 2, Test Points 513-525 (Mach 0.55, $\bar{q}_\infty = 108$ psf).

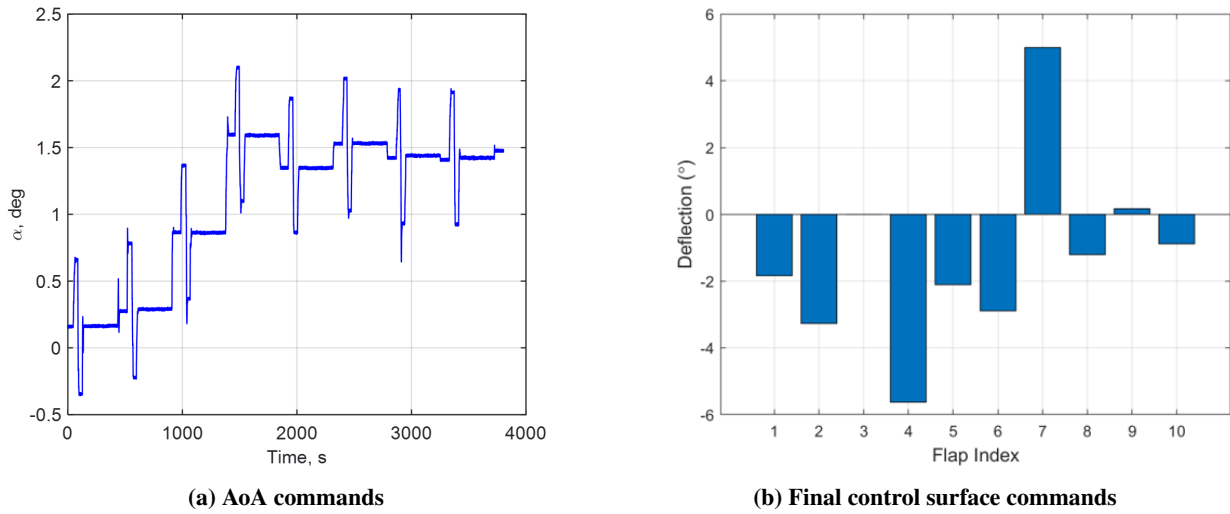


Fig. 30 Angle of attack commands and final surface deflections for the NASA drag controller during Entry 2, Test Points 513-525 (Mach 0.55, $\bar{q}_\infty = 108$ psf).

C. Maneuver Load Alleviation (MLA) Testing

The Boeing and NASA MLA control laws were tested at the same Mach 0.55, $\bar{q}_\infty = 107$ psf primary test point in the S0B0 configuration. For both controllers, the objective was to minimize the wing root bending moment, as that is a primary driver of wing structural sizing. In the IAWTM wind-tunnel testing, inboard-most strain gauge SG01 (planform location shown in Fig. 6b) was used as a surrogate for the wing root bending moment, so the goal of the IAWTM MLA control laws was to minimize SG01 measurements. Additional details on the controllers and results are found in an accompanying paper [20].

The NASA MLA control law performed similar real-time system identification and optimization as the NASA drag optimization control law. Angle of attack and control surface perturbations were used to identify lift and SG01 sensitivities to construct a surrogate aerodynamic model. The real-time optimization attempted to minimize SG01 readings while maintaining the target lift. As before, the controller displayed angle of attack commands for the human TDT operators and sent control surface commands directly to the actuators.

The test results of the MLA control law demonstrated a successful convergence to a lower strain gauge measurement for the same lift coefficient. NASA performed two versions of the same test, one with the deflection limits set to $\pm 5^\circ$ and another where the surface limits were increased to $\pm 10^\circ$. In both cases, the controller converged towards a new angle of attack and a set of control surface commands to keep lift constant. Figure 31 shows the final control surface positions for the two versions, while Fig. 32 illustrates the SG01 measurements. The two NASA MLA controllers demonstrated a 7% and 13% reduction, respectively, in SG01 while tracking the lift coefficient of $C_L = 0.5$.

The Boeing MLA control law operated differently from the NASA controller. Boeing designed a feedforward MLA control law that was based around load factor. For a wind-tunnel test, there is no equivalent load factor, so the control law utilized estimated load factor derived from trim C_L . The feedforward control allocation groups the surfaces into two groups, an inboard and outboard pseudo-surface, with the intent of shifting the bending load inboard in order to minimize wing root bending moment. The Boeing MLA controller would only command nonzero surface deflections once the load factor exceeded the 0.5g to 1.5g range.

For the Boeing MLA testing, the wind-tunnel model was pitched through a simulated pull-up maneuver. The wind-tunnel model started at $\alpha = -2^\circ$ and was pitched up to $\alpha = +2^\circ$. As the wind-tunnel model pitched up, the lift coefficient increased from $C_L = 0.15$ and triggered the MLA control surface commands once $C_L > 0.25$. The resulting controller performance is illustrated in Fig. 33, where the closed-loop controller is shown to reduce root strain by 10-13% compared to the open-loop maneuver.

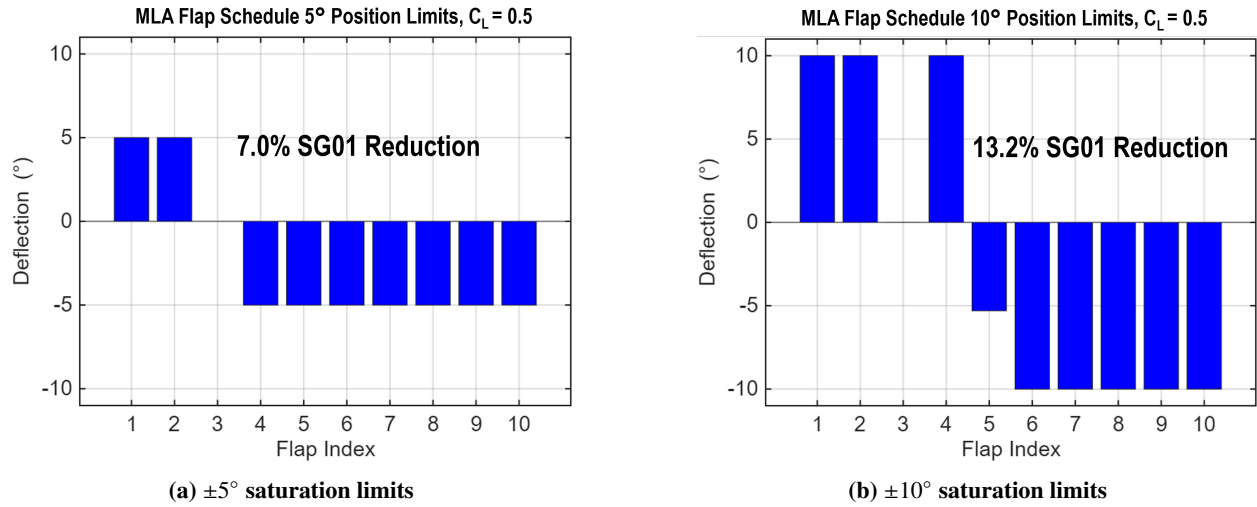


Fig. 31 Final control surface commands for the NASA MLA controller during Entry 2, Test Points 497-503 (Mach 0.55, $\bar{q}_\infty = 108$ psf).

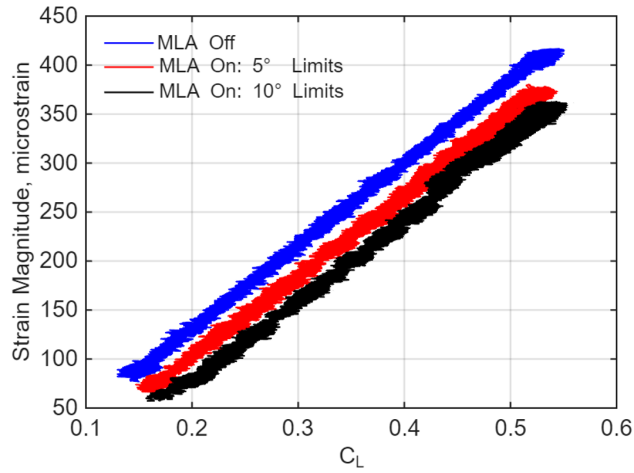


Fig. 32 NASA MLA controller performance for Entry 2, Test Points 497-503 (Mach 0.55, $\bar{q}_\infty = 108$ psf).

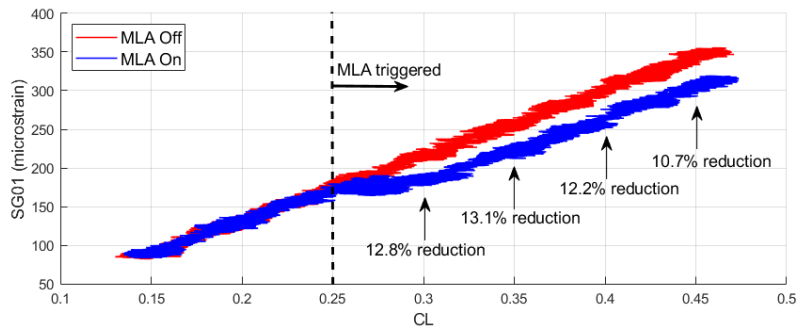


Fig. 33 Boeing MLA controller performance during a simulated pull-up maneuver at Mach 0.55, $\bar{q}_\infty = 108$ psf.

D. Gust Characterization Testing

After the conclusion of drag optimization and maneuver load alleviation testing, the balance shims were installed to transition to the S1B0 configuration for gust characterization testing. Since the gust vanes were not installed during Entry 1, there was no experimental data to confirm the accuracy of the ASE model predictions associated with the gust response. As a result, the Entry 2 gust characterization testing was critical to validate the gust vane dynamics and measurements prior to engaging closed-loop gust load alleviation.

For the Entry 2 gust testing, a gust sniffer probe was installed upstream of the model, shown in Fig. 7. This sniffer measured flow angularity in front of the aircraft and was treated as a surrogate for emerging atmospheric sensing technologies, such as light detection and ranging (LIDAR) sensors. There was particular interest in validation of the ASE model output for the gust sniffer position.

The gust characterization testing was centered around the same primary and secondary test points shown in Fig. 26, with the majority of the test runs conducted at Mach 0.55, $\bar{q}_\infty = 107$ psf. As noted earlier in the paper, all testing was conducted with the AOS vane pairs operated in-phase at a set amplitude of ± 3 degrees (6 degrees peak to peak). There were two gust characterization tests of interest. The first test sequence was a gust vane dwell that oscillated the gust vanes at a fixed frequency for a set period of time, usually about 30 seconds. Gust vane dwells were performed in 0.5 Hz increments between 0.5 and 4 Hz. The gust vane frequency was adjusted between data records, and transients were allowed to damp out before a new data record was started. The second test sequence was a gust vane frequency sweep, where the gust vane frequency was linearly increased from 0.5 to either 3 or 4 Hz.

These tests confirmed that the ASE model response matched the experimental data quite well. Figure 34 compares ASE model predictions against experimental data taken during a 0.5 Hz gust vane dwell at Test Point 678 (Mach 0.55, $\bar{q}_\infty = 107$ psf), where gust vane position measurements from the wind-tunnel test were fed into the ASE model to generate the ASE model predictions. The resulting ASE model prediction for the gust sniffer position compared well with the filtered experimental signal. Ultimately, the NASA-developed ASE models accurately modeled the gust response and confirmed the readiness for gust load alleviation testing.

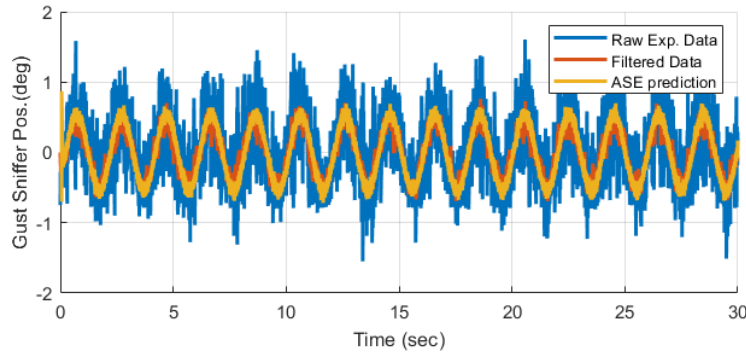


Fig. 34 Comparison of predicted ASE model output for gust sniffer position against the experimental data for a 0.5 Hz gust vane dwell during Entry 2, Test Point 678 (Mach 0.55, $\bar{q}_\infty = 107$ psf). Measured gust vane positions were used as input to the ASE model.

E. Gust Load Alleviation (GLA) Testing

The closed-loop gust load alleviation testing immediately followed the completion of open-loop gust characterization testing. Both NASA- and Boeing-developed gust load alleviation control laws were tested at the Mach 0.55, $\bar{q}_\infty = 107$ psf primary test point. The controllers were tested at various gust vane dwells between 0.5 and 3 Hz. Similar to the MLA objective, the GLA controllers were designed to minimize wing root bending moment. For the IAWTM wind-tunnel testing, this corresponded to a reduction in the response of the inboard strain gauge SG01 due to gust excitations. The controllers were also designed to demonstrate predictive control using the gust sniffer acting as a surrogate forward-looking atmospheric sensing LIDAR. There are additional details on the control laws and results found in accompanying papers [23–25].

The Boeing GLA control law coupled an extended Kalman filter (EKF) with a feedforward controller to demonstrate the potential benefits of a LIDAR-equipped aircraft. The EKF is used to estimate the gust vane states using the gust sniffer measurement. These estimated gust vane states were then fed into the feedforward controller, which utilized the

hydraulic ailerons to minimize strain gauge response. The controller gains were designed using the output-weighting linear quadratic regulator (LQR) process to incorporate the strain gauges into the LQR cost function. Two versions of the controller were developed, one with and one without AILB, to address concerns regarding AILB performance.

NASA developed two parallel GLA feedback control laws: one multi-objective linear quadratic Gaussian (LQG) design [24, 26] and one Hamiltonian-based control law [25, 27, 28]. The multi-objective LQG control law explicitly incorporated the wing root strain response into the LQR cost function. The multi-objective LQR controller was paired with two distinct state estimation techniques: one with an EKF that only utilized onboard sensors and a second one that added the gust sniffer as a surrogate LIDAR. The NASA Hamiltonian-based control law used the same state estimation techniques, but it had a different control law design process. Rather than an LQR-based design, the Hamiltonian controller used Hamilton's principle for distributed structural dynamics to design a distributed optimal controller [29].

The various versions of the NASA and Boeing controllers all successfully demonstrated GLA for different gust vane frequencies at the primary test point of Mach 0.55, $\bar{q}_\infty = 108$ psf. Figure 35 illustrates the performance improvement by the Boeing GLA feedforward controller during a 0.5 Hz gust vane dwell at Test Point 973. Figure 35a shows a 27% reduction in the root-mean-square (RMS) of SG01 measurements. Although not an explicit objective, Fig. 35b demonstrates that the GLA controller also reduced oscillation in the wingtip displacement by 30%, as measured with a laser displacement sensor added by NASA for Entry 2 testing. Figure 36 illustrates the performance of two of the NASA GLA feedback controllers during a 0.5 Hz gust vane dwell. The LQG-based controller produced a 53% reduction in the SG01 RMS, while the Hamiltonian-based controller produced a 66% reduction in the SG01 RMS. Tests of the Boeing and NASA controllers at different frequencies demonstrated similar improvements in performance.

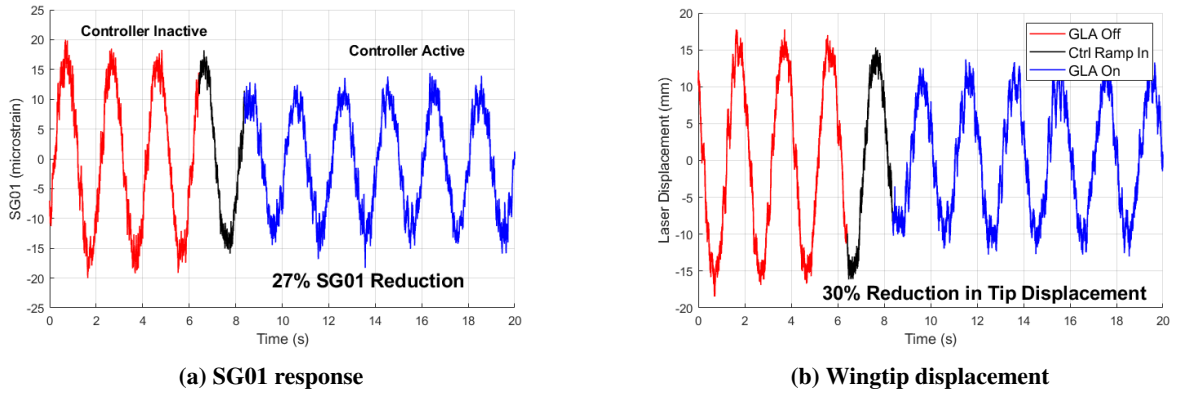


Fig. 35 Boeing GLA controller performance for a 0.5 Hz gust excitation at Mach 0.55, $\bar{q}_\infty = 108$ psf.

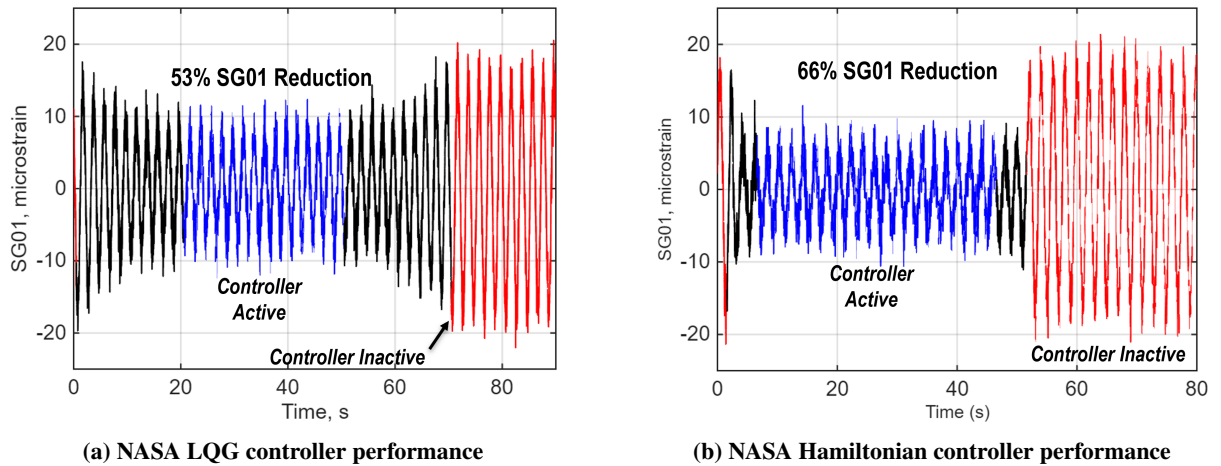


Fig. 36 NASA GLA controller performance for a 0.5 Hz gust excitation at Mach 0.55, $\bar{q}_\infty = 108$ psf.

F. Active Flutter Suppression (AFS) Testing

At the conclusion of the closed-loop GLA testing, the model was changed to the S1B1 flutter testing configuration by installing the flutter ballast at the wingtip. Prior to testing the control laws, a limited number of repeat open-loop test points were acquired on the same $H = 700$ psf pressure line as Entry 1 (shown in Fig. 18) up to Mach 0.65 in order to assess repeatability. During this portion of testing, a small number of ASE logarithmic sweeps and fast multisines were also performed on ailerons AILA and AILC to collect repeat characterization data. Note that AILB was not used due to its less effective performance encountered in Entry 1. Following these characterization tests, NASA tested multiple active flutter suppression controllers. Additional details on these controllers are found in accompanying papers [25, 30], and a brief overview is provided below.

NASA Ames developed a version of their Hamiltonian-based feedback control law to suppress the aeroelastic modes and stabilize the model as it approached the flutter boundary. This controller used outboard AILC for closed-loop control. AILB was not used for closed-loop control, but it was employed to generate external doublets to excite the aeroelastic dynamics of the wing. Originally, accelerometer A303Z was planned to be used for feedback control; however, early closed-loop tests in Entry 2 revealed that A303Z was also affected by the suspected case pressure issue observed for accelerometer A111Z during Entry 1. This was new to Entry 2, as no poor A303Z performance was noticed in Entry 1. The controller was subsequently redesigned to remove A303Z.

NASA Langley developed a collocation-based feedback control law for AFS called identically located accelerometer and force (ILAF) control. The ILAF controller used the same general layout as the Hamiltonian controller, with AILC used for closed-loop control, AILB generating external excitations, and A303Z dropped from use due to the suspected case pressure issue.

The NASA control laws were first tested in the previously-cleared open-loop stable test points at and below Mach 0.65. With closed-loop control engaged, a fast multisine excitation was sent through AILC to estimate real-time stability. This process was repeated at the previous checkout points as the tunnel was brought back up to Mach 0.65. Once the controllers were shown to be stable at the open-loop checkout points, the tunnel conditions were increased slowly beyond Mach 0.65. During separate test runs, the closed-loop system with the NASA Langley ILAF controller reached Mach 0.68, and the NASA Ames Hamiltonian controller reached Mach 0.72 before the team decided to come back down to assess model responses. The real-time stability estimation from the multisine excitation predicted that the wind-tunnel model was at or very close to the flutter boundary at the Mach 0.72 condition. It should be noted that for model safety reasons, open-loop testing did not reach a flutter point, so an experimental flutter boundary was not determined. Figure 37 illustrates the test points reached during the AFS testing. Figure 38 compares the aeroelastic mode response with and without the NASA Ames Hamiltonian controller at Mach 0.65, $\bar{q}_\infty = 137$ psf. There is a noticeable decrease in the first bending mode (Mode 1) achieved by the Hamiltonian controller as the predicted flutter boundary is approached.

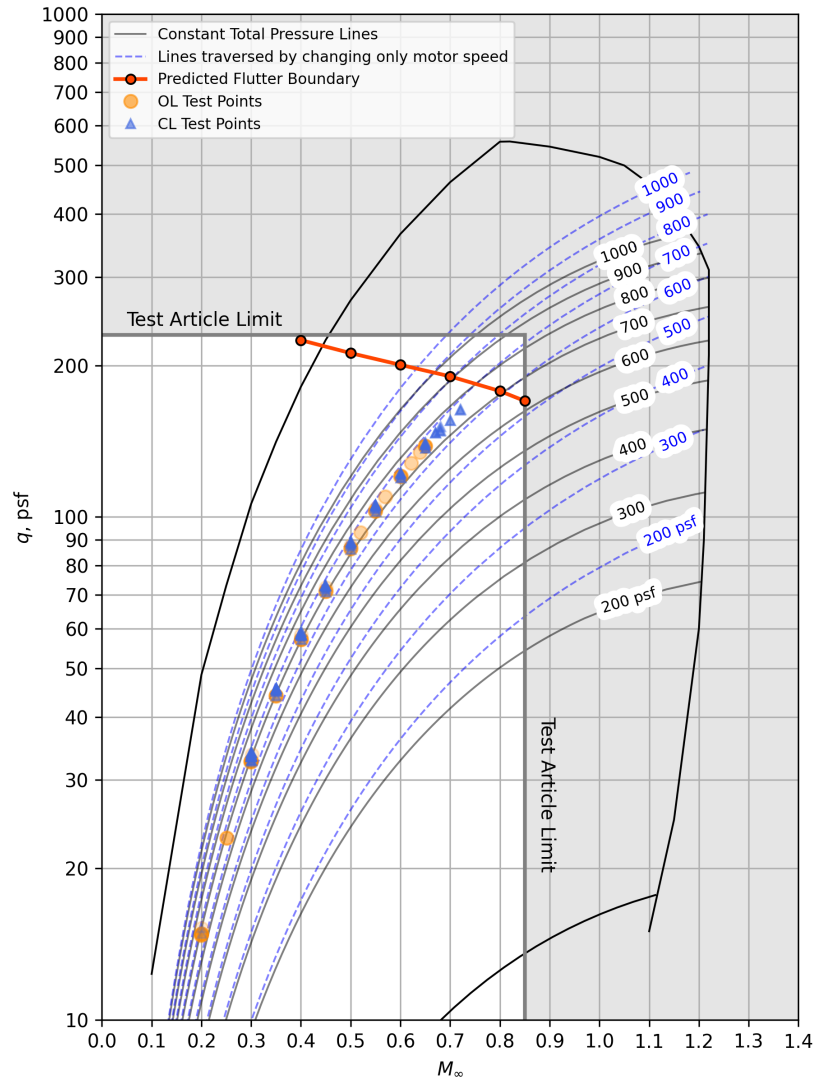


Fig. 37 Flutter suppression test points with the S1B1 configuration in Entry 2.

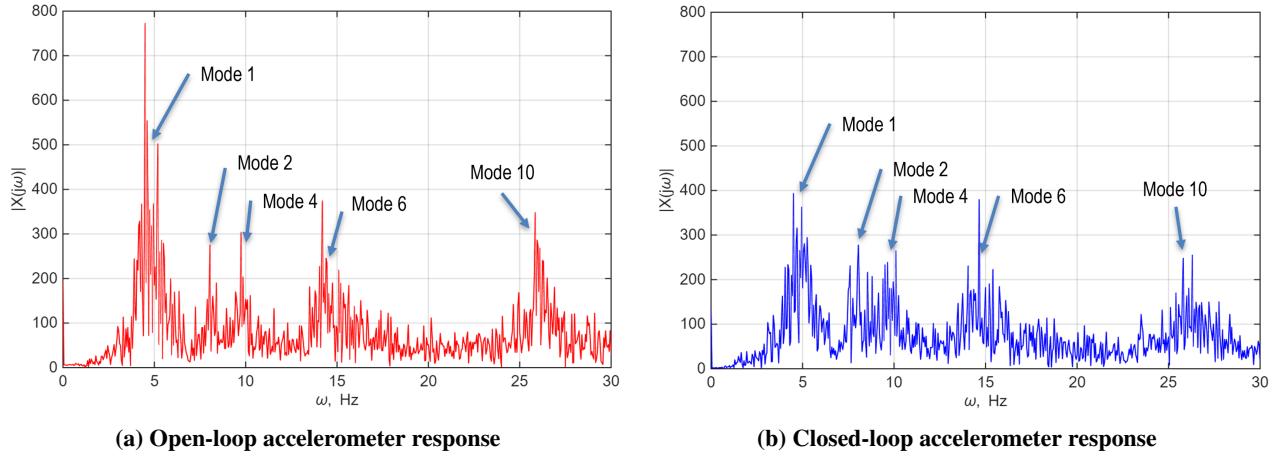


Fig. 38 Comparison of aeroelastic modal response measured by A110Z for the open- and closed-loop system with the NASA Ames Hamiltonian controller at Mach 0.65, $\bar{q}_\infty = 137$ psf. For each test point the model is excited with an AILB doublet.

G. Observations from Test Entry 2

The conclusion of the AFS controller testing marked the end of wind-tunnel testing for the NASA-Boeing IAWTM collaborative research program. Additional observations drawn from the open- and closed-loop testing conducted during Entry 2 include:

- All four control objectives were successfully demonstrated: drag optimization, maneuver load alleviation, gust load alleviation, and active flutter suppression.
- Measuring aerodynamic loads on a complex aeroservoelastic test article using a balance is challenging due to the high cabling and hydraulic line requirements that make it difficult to eliminate all load paths that are routed external to the balance. Fouling by the externally-routed hydraulic lines during Entry 1 appears to be the cause of poor aerodynamic load repeatability during Entry 2, when the hydraulic line supports were adjusted during extensive weight-alpha tares conducted during installation. Entry 2 closed-loop drag optimization tests were successfully accomplished, but aerodynamic load data acquired during Entry 1 was not as high quality as that acquired during Entry 2.
- Low signal-to-noise on some balance measurements complicated drag optimization control law performance and analysis. A possible mitigation may have been to initialize the test article in a high-drag configuration, i.e. all surfaces deflected to max, before engaging the controller during at least some of the drag optimization test points to improve observability of drag reduction controllers.
- Increased time margin for closed-loop test schedule planning would have been valuable, as available time allocations restricted the ability to incorporate model updates and correct issues encountered during testing.
- Open-loop gust characterization test results exceeded expectations. NASA Langley's ASE model predictions matched the test data much more closely than originally anticipated. The significant amount of FEM and ASE modeling effort demonstrated a good return on investment and enabled successful completion of gust load alleviation testing by minimizing last-minute rework.
- Fiber optic strain sensor (FOSS) data collected at various points along the Entry 2 test runs should be compared against the other on- and off-board sensor measurements. This includes comparisons of FOSS measurements against other measurements during closed-loop testing to investigate its use as a potential sensor for feedback control.
- Removing sensors from closed-loop controllers during Entry 2 due to suspected accelerometer case pressure issues was overcome by swapping for alternative sensors on the highly-instrumented model. The degraded sensors, however, would have been more optimal and yielded valuable data. Although in general the interior volume of the IAWTM test article available for sensor installation is relatively large, at the wingtip there is significantly less space.

VII. Conclusion

This paper summarized the planning and execution of the wind-tunnel test campaign completed as part of the joint NASA-Boeing IAWTM program. The IAWTM program aims to develop and test new technologies, sensors, and techniques that would enable higher aspect ratio commercial transport aircraft. The wind-tunnel model used during testing is a higher aspect ratio version of the NASA Common Research Model that represents a notional evolution of the trend towards higher aspect ratio wings with lighter structure. The dynamically-scaled halfspan wind-tunnel model is equipped with onboard strain gauges and accelerometers coupled with ten actuated control surfaces distributed along the trailing edge of the wing.

The wind-tunnel test campaign was broken into two tests. Entry 1 (Test 679) was conducted during the summer of 2024. This entry focused on aerodynamic, aeroservoelastic, and flutter characterization testing to validate and improve aerodynamic and aeroservoelastic models. NASA and Boeing employed these models during the design of drag optimization, maneuver load alleviation, gust load alleviation, and flutter suppression control laws. Entry 2 (Test 683) was conducted during the summer of 2025. This entry performed limited repeats of test points from the first entry, as well as characterization testing of the model response to gust vane excitations. The majority of testing in the second entry centered around closed-loop testing of the aforementioned drag optimization, MLA, GLA, and flutter suppression control laws. These controllers were successfully demonstrated over the course of this testing. A subset of lessons learned during the two entries are summarized in Sections V and VI, which may be beneficial to future aeroelastic test programs.

Acknowledgments

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