

THE X-59 JOURNEY TO AEROELASTIC AIRWORTHINESS CLEARANCE

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Abstract: The X-59 aircraft was conceived with a partnership between Lockheed Martin (LM) and the National Aeronautics and Space Administration (NASA) to demonstrate sonic boom reductions enabled by aerodynamic shaping. In order to safely fly the X-59 aircraft and get approval for expanding the flight envelope, LM and NASA structural dynamics engineers developed aircraft finite element models, completed numerous flutter analyses and aeroservoelasticity analyses, and conducted several structural ground tests. This paper will describe the journey towards attaining aeroelastic airworthiness clearance of the X-59 aircraft.

1 INTRODUCTION

The X-59 aircraft, or the Low-Boom Flight Demonstrator (LBFD), is a collaboration effort between Lockheed Martin (LM) (Lockheed Martin Aeronautics, Fort Worth, Texas) and the National Aeronautics and Space Administration (NASA) to tackle the challenge of the sonic boom [1]. Lockheed Martin designed and built the X-59 aircraft under a contract from NASA as part of the Quiet Supersonic Technology (QueSST) mission. One of the main objectives is to demonstrate that a sonic boom turns into a quiet “thump” that is achieved through aerodynamic shaping of its unique design of a long-nosed shape that manipulates shockwaves [2]. Reducing the sonic boom signature to acceptable levels would encourage federal and international regulators to allow commercial supersonic flight over land.

There are three phases to the QueSST Mission:

- Phase 1: Aircraft design, fabrication, and envelope expansion flights
- Phase 2: Sonic boom signature data collection and prediction tools from X-59 flights over land from a limited number of airfields
- Phase 3: Sonic boom community response data from X-59 flights over residential areas near major airports

Once the X-59 aircraft completes all the envelope expansion flights (Phase 1) and validates the acoustic sonic boom signature and prediction tools (Phase 2) [3], it will then be flown over several United States cities to collect community response data on the acceptability of the quiet sonic boom (Phase 3). National Aeronautics and Space Administration will provide this community response data to regulators to help approve new rules that potentially could allow future quiet commercial supersonic flight over land, which has been banned since 1973.

NASA Armstrong Flight Research Center (AFRC), Edwards, California) has a unique capability of a self-airworthiness certification program that is a very specialized, risk-based process designed to validate the safety of experimental aircraft, modified research vehicles, and unmanned aircraft that often do not have Federal Aviation Administration or Department of Defense certifications. National Aeronautics and Space Administration AFRC ensures that these unique aircraft are airworthy and are safe to fly via a comprehensive technical airworthiness review process, interdisciplinary analysis, and ground/flight-testing. For many years now, the X-59 Phase 1 activities from aircraft design, fabrication, build, analysis, ground testing, and envelope expansion flights have been reviewed through the AFRC airworthiness process. Currently, the X-59 project is in the final stage of Phase 1: Conducting the envelope expansion flights.

The unique X-59 aircraft design, as shown in Figure 1, has several aeroelasticity and aeroservoelasticity concerns due to the long, slender fuselage, thin wings, large engine mass, and the lift distribution which makes the aircraft statically unstable. All of these concerns drove the need for careful development of the finite element model (FEM) with structural ground testing to validate it to show safe aeroelasticity of the X-59 aircraft. The FEM was continuously improved throughout the life cycle of the project testing and analyses up to the beginning of flight-testing. Several structural ground tests were conducted and led by LM to improve and validate the FEM over a course of four years. This paper discusses three major structural dynamics ground tests: 1) freeplay test; 2) ground vibration test (GVT); and 3) structural coupling test (SCT).



Figure 1: X-59 Low-Boom Flight Demonstrator aircraft.

A modal analysis was conducted on the FEM to determine the natural frequencies and mode shapes of the aircraft. Lockheed Martin and NASA performed numerous aeroelasticity and/or flutter analyses using multiple aeroelastic models and codes for independent verification, but all analyses used the same MSC NASTRAN (MSC Software (formerly The MacNeal-Schwendler Corporation), Newport Beach, California) structural FEM. Linear flutter analyses were done in NASTRAN and ZAERO (Siemens Digital Industries, Plano, Texas (formerly ZONA Technology, Inc.) analyses software and non-linear computational fluid dynamics (CFD)-based flutter analyses were done in the NASA-developed softwares FUN3D and AeroElastic Reduced-Order Modeling (AEROM) and using linearized frequency domain (LFD) solutions. The aeroservoelasticity (ASE) analyses were completed using MATLAB® Simulink® (The MathWorks, Inc., Natick, Massachusetts, USA) state-space models of the flexible FEM, unsteady aerodynamics, and flight control laws.

Prior to first flight, numerous integrated ground tests and taxi tests were completed. The X-59 envelope expansion flights are an interdisciplinary plan of test points and maneuvers using a build-up approach measured in Mach number and knots equivalent airspeed (KEAS) to clear the full operational envelope. Flying a constant KEAS line allows dynamic pressure to be constant while only increasing the Mach number. The X-59 aircraft successfully flew its first flight on October 28, 2025, and continues to fly envelope expansion flights as of May 2026. This paper will describe Phase I activities of the X-59 aircraft journey to show adequate aeroelastic and aeroservoelastic margins for airworthiness clearance, enabling envelope expansion flights to clear the aircraft operational envelope and design point of Mach 1.4 at 55,000 ft for the community response testing.

2 X-59 AIRCRAFT DESCRIPTION

The Low-Boom Flight Demonstrator X-plane, as shown in Figure 2, is a single-pilot aircraft with an overall length of ≈ 99.5 ft, where the nose section is 38 ft (approximately one-third of the entire vehicle). Figure 2 also shows the different aircraft weight configurations and nomenclature. The X-59 aircraft features a highly swept wing (wingspan is ≈ 29.5 ft), a fixed canard, and a single vertical tail, and the design takeoff weight is 24,300 lb [2]. The aircraft is controlled by a digital fly-by-wire flight control system. The flight control surfaces are hydraulically actuated and consist of horizontal stabilators, ailerons and rudder. The flaps are electrically actuated to reduce landing approach speed and are not part of the active control system. At the top of the vertical tail, there is a small, trimmable

T-tail surface which is incorporated for sonic boom reduction and is also not part of the active control system.

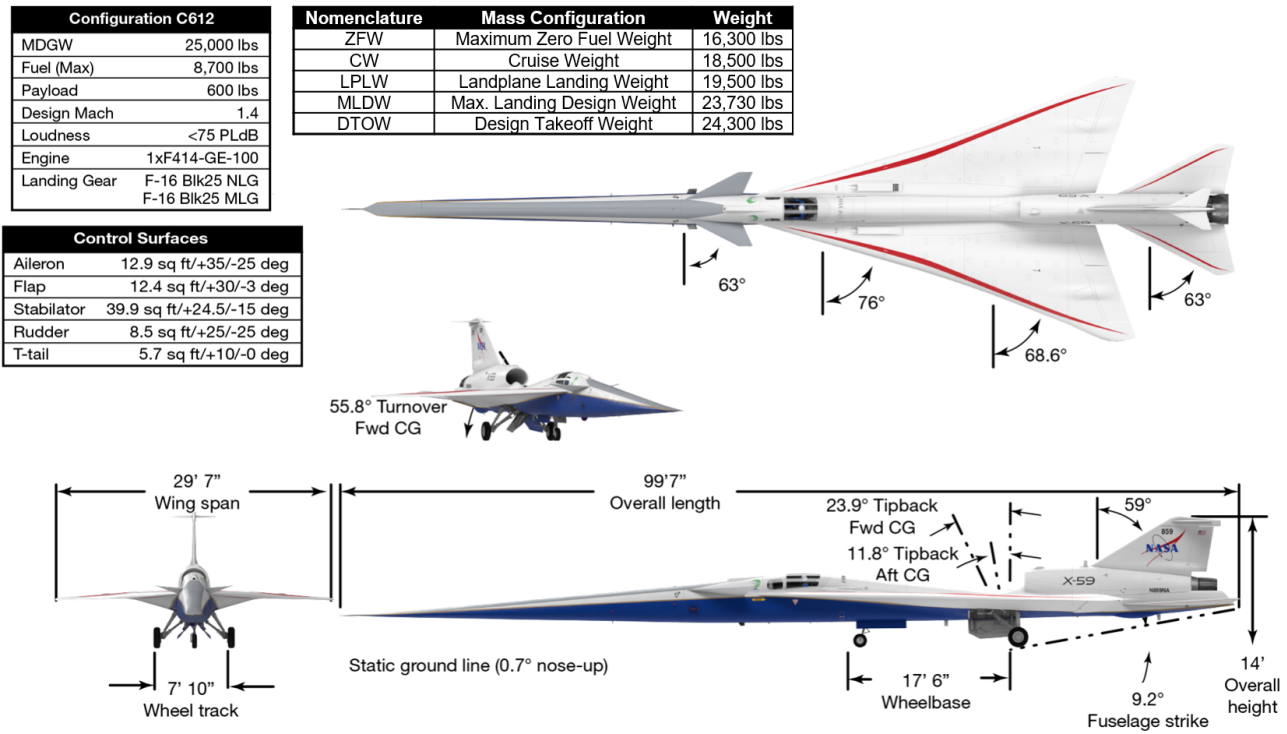


Figure 2: X-59 aircraft dimensions.

The X-59 aircraft is powered by a single General Electric (General Electric Company, doing business as GE Aerospace, Cincinnati, Ohio) F414 engine with the flight envelope expanded out to Mach 1.55, Figure 3 shows the target low-boom demonstration condition of Mach 1.4 at 55,000 ft for the community response testing, as shown in the operational flight envelope.

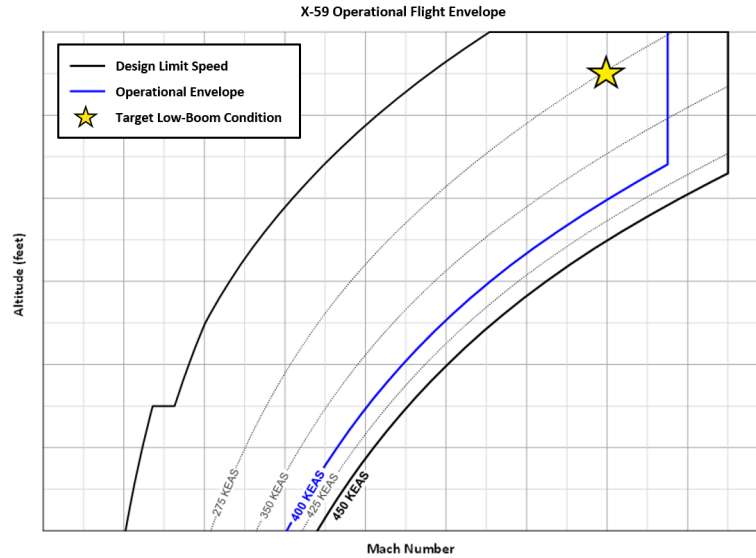


Figure 3: X-59 aircraft flight envelope and target low-boom condition.

3 X-59 FLUTTER DESIGN REQUIREMENTS

The following flutter design requirements, as shown in Table 1, are in the X-59 project structural design criteria (SDC) and airworthiness requirements and criteria (ARC) documentation. These requirements drove the aircraft design and fidelity of the FEM, structural testing, and structural analyses to show acceptable aeroelastic and aeroservoelastic margins for airworthiness clearance for flight.

Table 1: X-59 flutter design requirements for airworthiness clearance.

Criterion	Requirement
Design Limit Speed (Vd/Md)	450 KEAS to Mach 1.7
Cruise Speed (Vc/Mc)	400 KEAS to Mach 1.5
Maximum Altitude	60,000 ft
Maneuver Stalling Speed (V_{S1})	162 KEAS below 25 k ft; Linear interpolated to 195 KEAS at 60 k ft
Landing Stalling Speed (V_{SL})	135 KEAS below 15 k ft
Airspeed Margin	Flutter free up to +15% on Vd/Md
Damping	$\geq 0.03g$ for flutter critical modes
ASE (for all speeds up to Vd/Md)	From ARC: Nominal or single IMU failure: Gain Margin ≥ 6 dB and Phase Margin $\geq 60^\circ$ From SDC: Nominal or single IMU failure: Gain Margin ≥ 8 dB and Peak Gain ≤ -6 dB Dual IMU failure: Peak Gain < 0 dB
Control Surface Buzz	No Buzz up to Vd/Md
Control Surface Freeplay	Aileron: $\leq 0.13^\circ$, flap: $\leq 1.15^\circ$, rudder: $\leq 0.13^\circ$, stabilator: $\leq 0.13^\circ$
Fail-Safe Design	Maintain 15% airspeed margin & ASE margins for single-system failed hydraulic actuators

As shown above, the ASE requirements have two standards: the first (from the ARC documentation) is the conventional 6-dB gain margin; the second (from the SDC setting) is a stricter requirement of 8 dB gain margin with all flexible mode response peak gain at -6 dB or lower. The 8-dB margin value was instituted as a conservative design approach to having strong ASE sensitivities to flexible modes; however, if structural notch filters are too strong, they can have a destabilizing effect on the phase lag of the rigid-body modes. A compromise was adopted where in general the target gain margin was 8 dB, but for those flight conditions where the rigid-body phase response was adversely affected, the ASE margin could be relaxed to 6 dB. In all cases, the requirement remained for flexible mode peak gains to be no greater than -6 dB.

4 X-59 FINITE ELEMENT MODEL

The unique X-59 aircraft design has several aeroelasticity and aeroservoelasticity concerns due to the long, slender fuselage, thin wings, large engine mass, and the lift distribution which makes the aircraft statically unstable. All these concerns drove the need for the development of a highly accurate finite element model (FEM) and structural testing to validate the FEM. The preliminary FEM was used to determine the locations for the flight flutter instrumentation, and the validated FEM was used for the aeroelasticity and aeroservoelasticity analyses showing that the X-59 aircraft was safe to fly.

The X-59 airframe consists of three major segments: 1) the wing, 2) the fuselage, and 3) the empennage, which were further divided into 17 subsegments, as shown in Figure 4. The materials for the aircraft are also shown and color coded (Figure 4): The majority of the airframe is aluminum, composite, and titanium. The overall aircraft design and materials have remained stable since the critical design review in 2019.

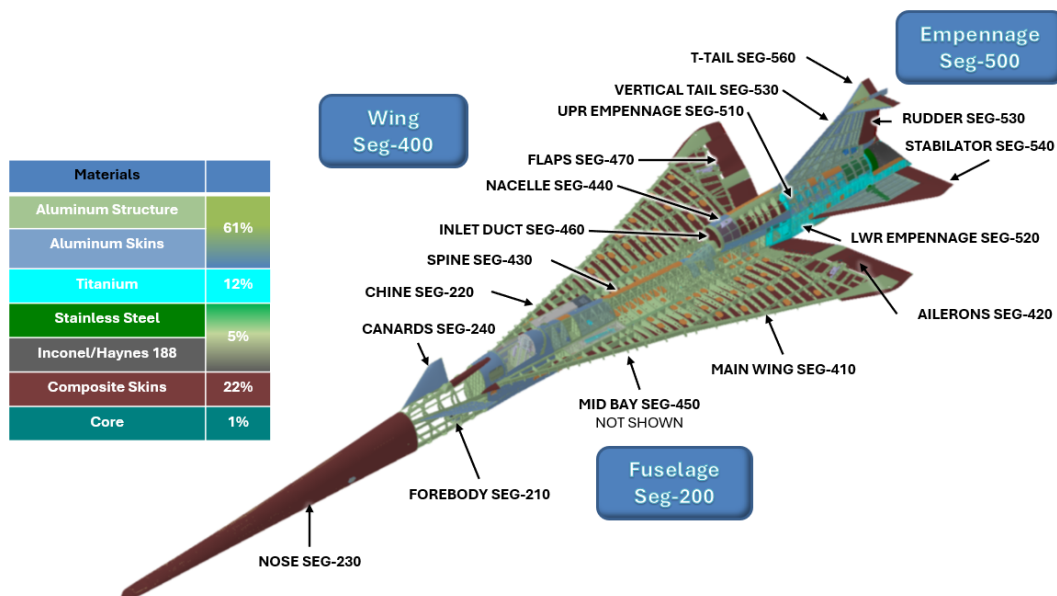


Figure 4: X-59 airframe overview.

The X-59 FEM, referred to as the air vehicle (AV) FEM, was developed in MSC NASTRAN and was used for both static and dynamic analyses. Throughout the course of the program, the AV FEM was

regularly updated to reflect design changes and structural test results. Figure 5 shows the evolution of the AV FEM over many years, starting from contract award to final airworthiness certification.

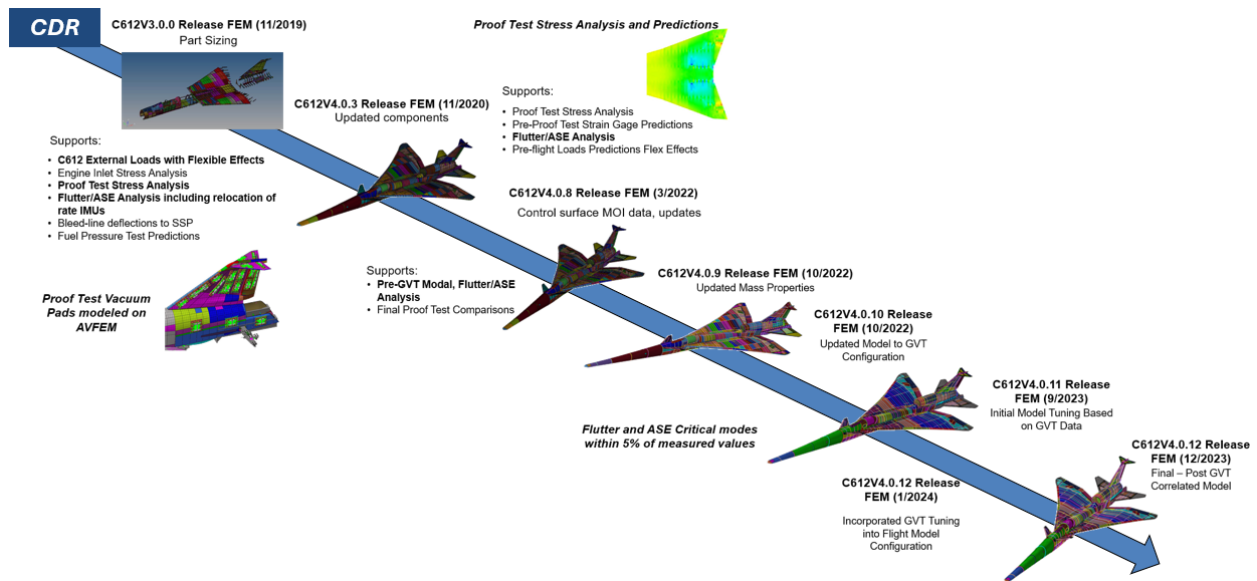


Figure 5: Evolution of the X-59 air vehicle finite element model.

In addition to changes driven by design maturation, as the aircraft was assembled and control surfaces were fabricated, the FEM evolved to reflect empirical test results. Moment of inertia (MOI) testing was used to match FEM mass properties, aircraft proof testing improved the overall FEM stiffness, and the ground vibration test correlated vehicle modal characteristics. It took many years of structural ground testing to acquire the data necessary to tune the FEM to become a validated structural model of the “as-built” one-of-a-kind aircraft. This validated FEM was used in many of the aeroelasticity and aeroservoelasticity analyses. The final post-GVT correlated NASTRAN FEM consisted of 97,321 grid points, 583,926 degrees of freedom (DOF), and 156,044 structural elements (such as bars, shells, solids, rigid and elastic connections) as shown in Figure 6. In the FEM, the control surface actuator stiffnesses are key parameters to tune for accurate modeling of flutter sensitivity, as these parameters directly affect the frequency of the control surface rotation modes. On the X-59 airframe, there are two classes of control surfaces, each driven by separate actuator types: “active controlled” surfaces and “trimmed” surfaces. The active controlled surfaces (stabilator, aileron, and rudder) use dual-system hydraulically powered actuators, whereas the trimmed surfaces (flap and T-tail) employ electro-mechanical actuators, with each surface having different characteristics. The FEM needed to realistically model both types of actuators.

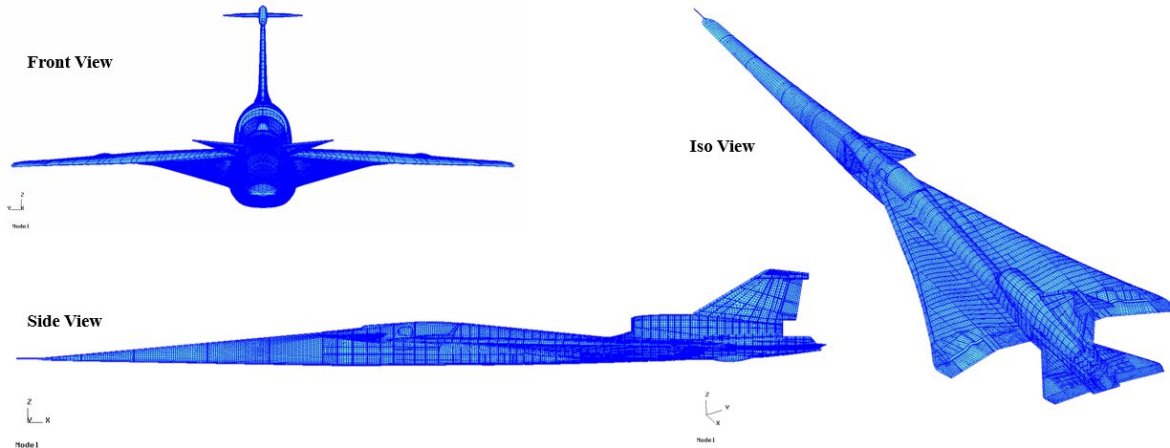


Figure 6: Different views of X-59 air vehicle finite element model.

For the aeroelastic analysis, a simplified, reduced-order A-Set FEM was created from the detailed FEM using the LM proprietary process with Guyan reduction in a way that preserves the modal characteristics (frequencies and mode shapes) from the detailed FEM. By reducing the model size to over 99.7%, the mass and stiffness matrices were significantly reduced, comprised of only 502 grid points with a total of 1,503 DOF, as shown in Figure 7, allowing for faster computational time for the modal, flutter, and ASE analyses. This process is challenging and is an art to reduce the FEM *enough* while still preserving the same frequencies (less than 1% difference) and mode shapes of the detailed model.

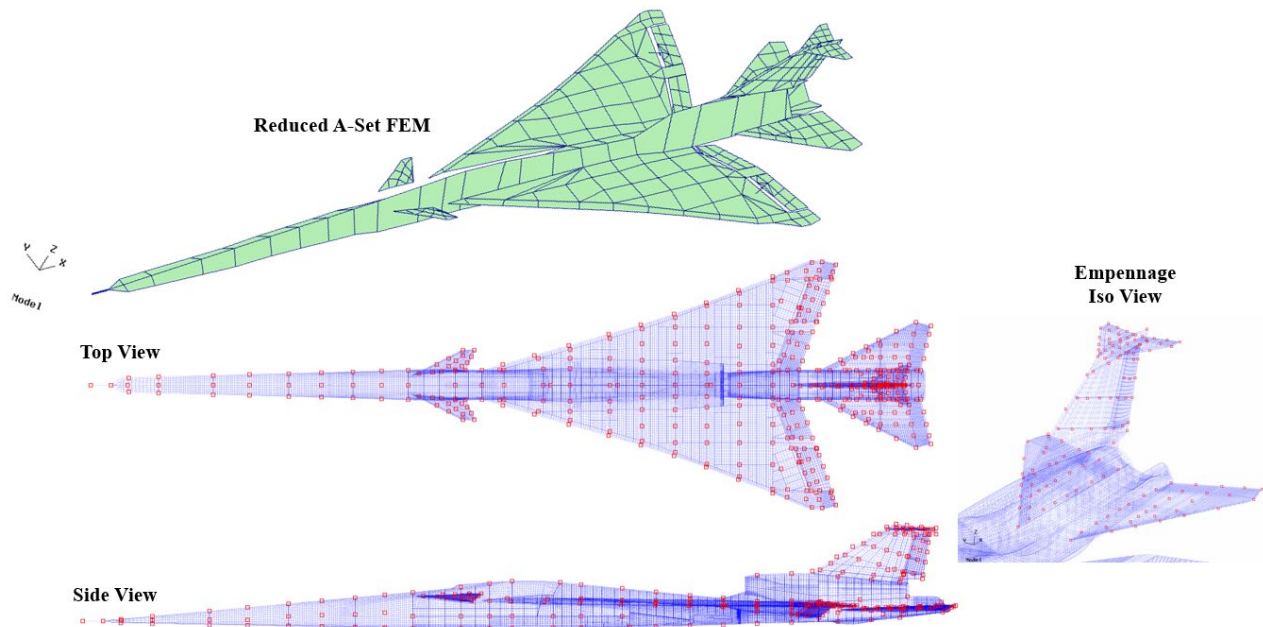


Figure 7: X-59 reduced A-Set FEM grids for aeroelastic and aeroservoelasticity analyses.

5 X-59 STRUCTURAL GROUND TESTS

The FEM provided awareness and pretest predictions for the structural ground tests. There were many structural ground tests (as shown in the following list) that were conducted on the X-59 aircraft to improve and validate the FEM over the course of four years, prior to first flight. Details related to testing focused on structural strength; however, this subject will not be discussed in this paper. Instead, this structural ground test section will only describe the last three X-59 ground tests pertaining to structural dynamics: Freeplay test, ground vibration test (GVT), and the structural coupling test (SCT).

- Landing gear calibration – Summer 2020
- Control surface actuation calibration – Winter 2022
- Fuel tank relief structural test – Fall 2020
- Cockpit pressure proof test – Fall 2021
- Full airframe proof/calibration test – January 2022
- Stabilator proof/calibration test – February 2022

Structural dynamics (focused ground tests):

- Freeplay test – July 2023
- Ground vibration test – July 2023
- Structural coupling test – Started August 2023, completed March 2024

5.1 Freeplay Test

The X-59 freeplay test was conducted by the Lockheed Martin Fort Worth Structural Test Lab personnel over several days, in July 2023, in an outdoor closed canopy at Run Stall 5 at the Lockheed Martin Palmdale, California, site. The objective of this freeplay test was to measure the rotational freeplay (or mechanical lash) about the hinge line of each control surface and landing gear doors. The test measurements were made to compare freeplay design requirements to give baseline freeplay results for future freeplay comparisons, provide data for the freeplay analysis, and to adjust the actuation loop stiffness in analytical models used for flight clearance analyses. The following X-59 control surfaces and doors were tested: Flaps (left hand/right hand), ailerons (left hand/right hand), stabilators (left hand/right hand), rudder, T-tail, nose landing gear doors, and both main landing gear doors. Table 2 shows the tested components with the rotation freeplay design limits.

Table 2: X-59 rotation freeplay design limits.

Component	SDC freeplay design limit (deg)
Flaps (LH/RH)	1.15°
Ailerons (LH/RH)	0.13°
Stabilators (LH/RH)	0.13°
Rudder	0.13°
T-tail	0.13°
Landing gear doors (qty 4)	1.15°

* Left hand (LH); right hand (RH)

Control surface freeplay in the actuation system represents a zero-stiffness region in the loop stiffness inherent to the actuator, control surface, and backup structure. Freeplay can also be caused by bearing clearances in the actuation linkage and possibly loose structural assemblies/joints. Structural dynamic engineers are interested in freeplay because the control surface loop stiffness is a critical parameter for aircraft aeroelastic stability and can influence buzz, limit cycle oscillation amplitude, and other aeroelastic, and aeroservoelastic phenomena.

The aircraft setup for the X-59 freeplay testing required removal of the hinge-line seals from the ailerons, flaps, and rudder. Aircraft full hydraulic and/or actuator power was applied to the surfaces and doors to accurately replicate an in-flight aircraft configuration, and the flight control loops were opened. The control surfaces were powered in the trimmed position prior to test loading. The landing gear doors were in the open position.

Each control surface was individually loaded with the target tension and compressive loads for maximum hinge moment. Measurements of the applied load and response surface deflections provided load versus deflection curves to estimate the loop stiffness in the actuator and structure. The slopes of the load-deflection curves yield the loop stiffness measurement. Figure 8 shows a generic freeplay data stiffness curve (hinge moment versus rotation) calculated from load and deflection measurements. When a control surface is loaded in both directions (e.g., trailing edge up and down), the graphical offset of the stiffness curves represents the freeplay in the system. To compensate for the hysteresis observed when starting from zero load, each control surface was tested with at least four loading cycles; the first load cycle “exercised” the surface, and the final freeplay value was an average from the three other load cycles.

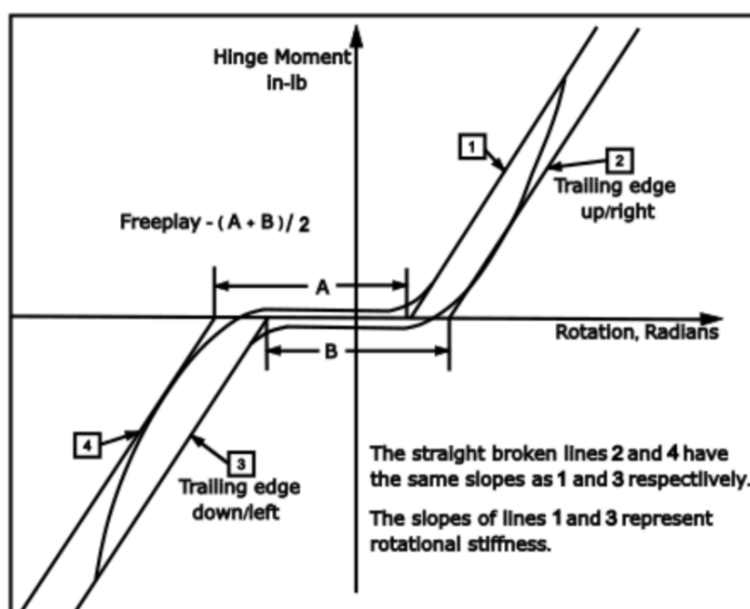


Figure 8: Example of a freeplay graph from stiffness curves calculated from load and deflection measurements.

The LM X-59 freeplay hardware used a vacuum-attached pneumatic actuator to apply tension and compression loads with an in-line load cell, and vacuum-attached deflection measurement hardware housed two direct current differential transformer (DCDT) displacement transducers to precisely

measure deflection (see Figure 9 showing the right aileron freeplay setup). A Hottinger Brüel & Kjær (Duluth, Georgia, USA) QuantumX portable data acquisition (DAQ) system was used to record and monitor real-time load and deflection measurements. Post-test, a MATLAB freeplay tool processed the raw load-deflection data and made graphical freeplay calculations. For the landing gear doors, a dial indicator was used for measuring rotation as the door was manually moved back and forth through the freeplay region. Figure 10 shows the left-main landing gear door freeplay setup.

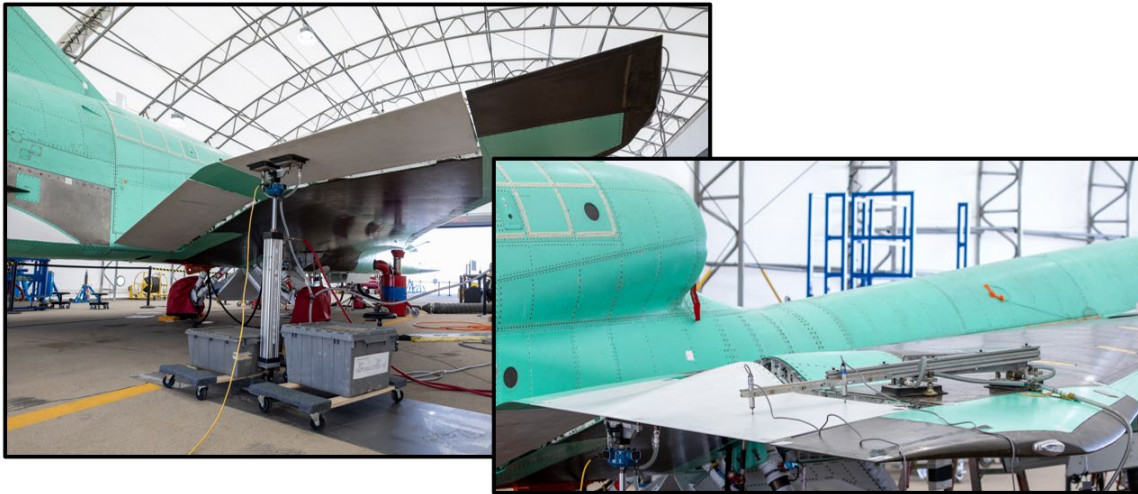


Figure 9: X-59 right aileron freeplay setup with vacuum attached loading and deflection measurement hardware.

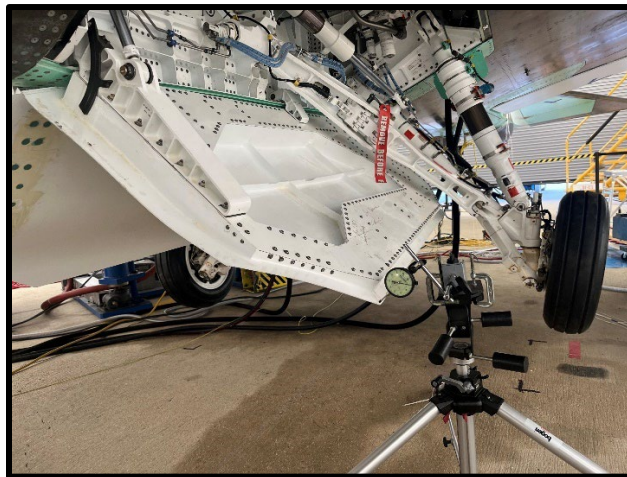


Figure 10: X-59 left main landing gear door freeplay setup using a dial indicator.

Table 3 shows results of the X-59 freeplay test from July 2023 except the flap results which are from a more recent test in July 2024, after the installation of larger pins. Lockheed Martin conducted freeplay analysis that estimates the potential limit-cycle oscillation (LCO) amplitudes that can occur when the control surfaces have zero hinge moment and are in the freeplay region. The resulting oscillatory loads reacted by the actuators should remain low enough to avoid structural fatigue or control problems. Analysis using the measured control surface freeplay values showed that LCO loads are expected to be less than $\pm 10\%$ of the design limit, which should not impact fatigue life, for all

surfaces except the flaps. This is why freeplay on the rudder and stabilators remains acceptable even though the rudder and stabilator measurements exceed the design target values. Initially, freeplay values on both the left and right flaps were less than the design target values in Table 1 (based on MIL-A-8870 criteria) but the LCO analytical predictions indicated loads near $\pm 20\%$ of limit which is unacceptable; therefore, the project fabricated precision/tight tolerance pins for the flap actuator attachments, then repeated the flap freeplay test in July 2024, which provided acceptable results.

Table 3: X-59 freeplay test results.

Component	Measured freeplay
Flap (LH)	0.63°
Flap (RH)	0.59°
Aileron (LH)	0.13°
Aileron (RH)	0.07°
Stabilator (LH)	0.16°
Stabilator (RH)	0.19°
Rudder	0.17°
T-tail	0.07°
Main gear door (LH)	0.06°
Main gear door (RH)	0.08°
Nose gear door (LH)	0.14°
Nose gear door (RH)	0.15°

As the X-59 aircraft continues flying through Phases 1, 2, and 3, freeplay measurements will be repeated every 100 flight hours to ensure freeplay measurements are still acceptable. If freeplay is beyond limits, the actuator linkage hardware will be inspected, retorqued or replaced, and another freeplay check will be performed.

5.2 Ground Vibration Test (GVT)

As previously mentioned in section 5.1, the X-59 ground vibration test (GVT) was conducted prior to freeplay testing by the Lockheed Martin Fort Worth Structural Test Lab personnel, from July 22–29, 2023, in Run Stall 5. The GVT objective was to measure primary and flutter-relevant structural modes of the X-59 airframe in both the empty and maximum allowable fuel configurations. The GVT successfully measured the modal parameters (frequencies, mode shapes, and damping) with the airframe in a free-free state for both fuel configurations. The results were used to validate the FEM used in the flutter analyses, identify modes to track in-flight testing, and refine structural notch filter frequencies used in the aircraft control system, typically referred to as the control law (CLAW). Modal testing on the landing gear (nose and main gear) and aft deck was conducted prior to the airframe GVT utilizing an impact hammer.

For the GVT setup, the X-59 aircraft was in a flight-representative state with all structure and subsystems in place. The control surfaces were powered on and held faired (0° deflection) during testing. Prior to the GVT, aircraft weight and balance measurements were taken in the empty or zero-fuel-weight configuration. Ballast was used for the pilot (200 lb) and any additional missing

equipment. The X-59 aircraft was supported at its three jack points using the LM soft support system (SSS), as shown below in blue in Figure 11. The SSS used three pressurized canisters with jacking capability built in and free-free boundary conditions to simulate the aircraft in-flight. This setup permitted the landing gear to be retracted, which allowed for a direct comparison between GVT versus FEM modal results. Figure 12 shows one of the SSS canisters installed on the right-aft jack point of the aircraft.

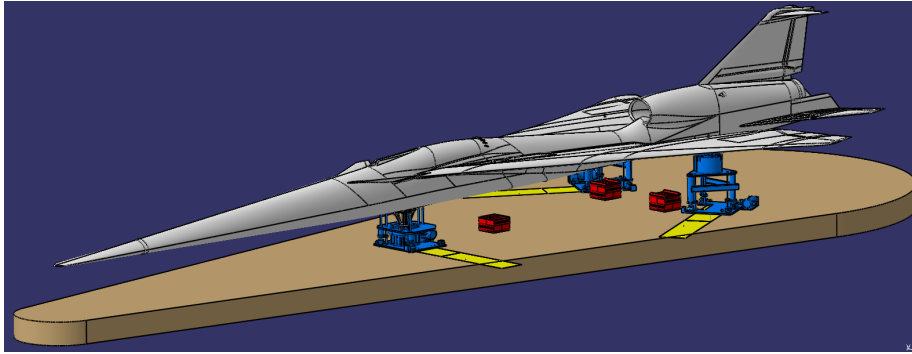


Figure 11: X-59 installed on LM soft support system at aircraft jacking points.



Figure 12: Soft support system installed at left- aft jack point.

The aircraft was excited with burst-random and sinusoidal dwell-decay excitations using the MB Dynamics (MB Dynamics, Inc., Cleveland, Ohio, USA) Modal 50 electromagnetic shakers in different configurations on the wingtips, stabilators, and vertical tail, some of the shaker setups are shown in Figure 13 [4]. Initially, the shakers were installed away from the active control surfaces, but later, shakers were directly attached to the ailerons, flaps, and rudder to characterize their individual surface response and damping.

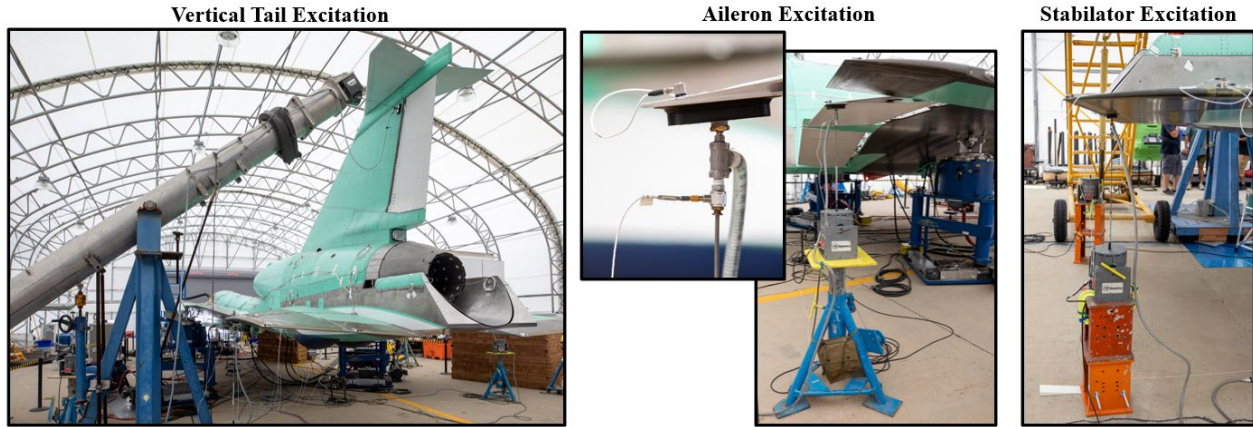


Figure 13: Shaker setup on the vertical tail, right aileron and right stabilator.

External GVT accelerometers, PCB 333A32 (PCB Piezotronics, Inc. in Depew, New York, USA), were installed at approximately 154 locations on the aircraft with approximately 210 degrees of freedom in the aircraft coordinate system. Figure 14 shows a test display model that was created for plotting the GVT mode shapes. The right side of the aircraft had more sensors for increased resolution of mode shape node lines. The LM Siemens/LMS GVT data acquisition system was used to generate the shaker excitation and record the accelerometer responses. Siemens Simcenter Testlab 2019.1 software extracted the modal parameters from frequency response functions (FRFs) and calculated the structural frequency, mode shape, and damping.

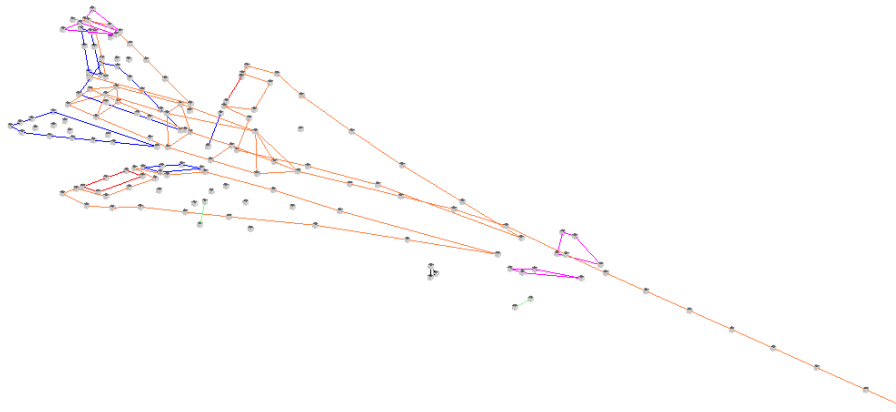


Figure 14: GVT test display model with ≈ 210 DOFs.

Testing started at the minimum weight, no usable fuel configuration. Prior to shaker excitation, the six rigid-body modes of the aircraft on SSS were manually excited and measured with a minimum of ten cycles. Then, simultaneous uncorrelated burst-random excitation was performed with five shakers at various locations (wingtips, stabilators, and vertical tail) with increasing force levels. Next, the control surfaces (ailerons, flaps, and rudder) were tested one at a time with the sinusoidal dwell-decay excitation and abruptly stopped in order to calculate the control surface damping. The last minimum weight test was done to simulate a single-system actuator failure case, where a pressure loss of one of the two aircraft hydraulic systems was reduced to zero in order to measure the differences in hinge-line rotation frequency of the hydraulically powered control surfaces. The aircraft was then fueled

with 7,000 lb of fuel for the design takeoff weight (DTOW) configuration, and the burst-random excitation with five shakers was repeated.

Table 4 shows the results for the DTOW configuration where the GVT frequencies are compared with the FEM pretest predictions, and the GVT mode shapes are compared with the FEM using the modal assurance criteria (MAC). The MAC is a statistical indicator used in vibration and structural analyses to quantify the degree of similarity (or correlation) between two mode shapes, typically from experimental testing and finite element analysis. An important takeaway from Table 4 are the results that are marked with an asterisk, indicating ASE notch filter modes (*) and the primary flutter modes (**). Extra scrutiny during testing was taken to ensure these modes were effectively identified because of their significance for the ASE notch filters and flutter analysis. The flap rotation frequencies show the largest percentage difference when comparing the GVT to the FEM predictions due to freeplay effects. Post-test, the FEM was correlated and tuned to better match the GVT results for the final round of the aeroelastic and ASE analysis.

Table 4: X-59 design takeoff weight (DTOW) GVT results compared to FEM pretest predictions.

Mode shape	Sym/Anti	FEM-predicted frequency (Hz)	Measured frequency (Hz)	Measured damping ratio (%)	% Diff. from predictions	MAC
Fuselage 1st bending *	Sym	5.50	5.22	0.44	-5.1%	99
Fuselage 1st bending	Antisym	7.64	6.93	0.22	-9.3%	88
Wing 1st bending *	Sym	8.54	8.79	0.46	2.9%	97
VT 1st bending	Antisym	10.07	10.28	0.46	2.1%	96
Fuselage 2nd bending *	Sym	10.69	10.21	0.13	-4.5%	77
Wing 1st bending	Antisym	11.06	10.99	1.34	-0.6%	93
Fuselage 2nd bending	Antisym	12.42	13.32	1.04	7.2%	63
Stab rotation **	Antisym	16.06	16.77	3.20	4.4%	91
Stab rotation *	Sym	16.41	17.50	3.99	6.6%	93
Aft fuselage torsion/Stab bending **	Antisym	18.81	20.14	1.25	7.1%	43
Fuselage 3rd bending *	Sym	18.89	17.94	0.43	-5.0%	82
Wing 1st torsion	Sym	20.45	21.77	2.65	6.5%	55
Wing 1st torsion	Antisym	21.08	22.68	1.03	7.6%	65
Fuselage 3rd bending	Antisym	22.59	22.08	1.74	-2.3%	19
Wing 2nd bending	Sym	24.00	24.87	1.95	3.6%	47
Canard bending	Antisym	25.09	26.42	0.81	5.3%	87
Canard bending	Sym	26.00	28.07	0.67	8.0%	91
Wing 2nd bending	Antisym	27.50	26.45	0.81	-3.8%	87
Stab bending	Antisym	28.12	28.32	1.09	0.7%	43
T-tail pitch	Sym	30.59	38.84	1.10	27.0%	71
Stab bending	Sym	31.45	32.43	1.58	3.1%	35
Flap rotation (RHS dominant)	Sym	31.83	21.33	2.48	-33.0%	23
Flap rotation (LHS dominant)	Antisym	32.14	20.21	2.48	-37.1%	5
VT torsion	Antisym	32.32	30.22	0.56	-6.5%	64
Aileron rotation	Antisym	31.83	30.50	1.48	-4.18%	28
Aileron rotation	Sym	32.14	30.15	1.72	-6.19%	22
Aileron torsion	Antisym	33.64	34.21	2.27	1.7%	43
Aileron torsion	Sym	34.17	34.83	1.51	1.9%	53
T-tail pitch	Antisym	37.96	35.98	1.46	-5.2%	67
Aft fuselage torsion	NA	46.90	45.18	0.09	-3.7%	79
Rudder rotation	NA	51.34	52.17	4.2	1.6%	71

* Note: Structural notch filter modes (*); primary flutter coupling modes (**); modal assurance criteria (MAC), stabilator (Stab), vertical tail (VT).

5.3 Structural Coupling Test (SCT)

The X-59 structural coupling test (SCT) initially started after the freeplay testing in Run Stall 5 at the LM Palmdale, California facility. There were three fuel loadings planned for the SCT. In early August 2023, SCT testing of the first aircraft configuration was completed, but then there was an issue with

the aircraft vehicle management system (VMS) hardware that prevented the SCT from continuing. Once the VMS issue was resolved, the SCT resumed in mid-March of 2024 for the remaining test configurations. The X-59 SCT objectives were as follows: (1) measure the open-loop actuator transfer functions (TFs) and surface-to-sensor TFs for the active flight control surfaces (aileron, rudder, and stabilator); and (2) evaluate the performance characteristics of structural filters with open-loop and closed-loop testing. In flight, unsteady airflow and operation of aircraft control surfaces can excite the aircraft structural vibration modes, creating displacements that flight control sensors detect as undesirable feedback. To mitigate this unwanted feedback, structural frequency notch filters are integrated into the aircraft control laws to attenuate these specific frequencies and ensure system stability. The SCT data provided critical X-59 as-built response characteristics data for refining analytical models that were used to predict aeroservoelasticity (ASE) stability and optimize the flight control laws. The ASE analysis and SCT test results verified the flight control system gain and phase stability requirements for airworthiness.

For SCT, the aircraft setup was very similar to the GVT, with a flight-representative aircraft including hydraulic and electrical power for the control surface actuators. The SCT was done with the aircraft on the SSS, and the maximum allowable fuel weight configuration was repeated on-gear. The engine was not running for the SCT, so a continuous slow engine rotator system was used as a mitigation for acceleration spikes on the engine, and engine accelerometers were monitored during test runs. The SCT began with the final, maximum-allowable fuel GVT configuration. The CLAW is sensitive to fuel variations since structural frequencies are dependent on weight, so three fuel loadings were used for SCT configurations: (1) maximum allowable fuel; (2) 37% fuel; and (3) zero usable fuel. See Table 5 for the SCT fuel configurations and boundary conditions.

Table 5: X-59 SCT configurations with fuel loading and boundary conditions.

Configuration	Fuel loading	Weight (lb)	Boundary condition
1	Max allowable fuel	24,300	SSS/gear up
2	37% fuel	19,500	SSS/gear up
3	Zero usable fuel	17,976	SSS/gear up
4	Max allowable fuel	24,300	On-gear, nominal tire/strut pressures

The active controlled surfaces (stabilator, aileron, and rudder) were excited using the aircraft flutter excitation system (FES) with random, swept sine, and multi-sine signals to excite the aircraft structural modes and induce displacements picked up by the flight control sensors as the undesirable feedback. The FES random excitation was 60 seconds (s) from 0 to 50 Hz with a 4-Hz high-pass filter to avoid exciting the SSS. The sine sweep was 30 s from 3 to 33 Hz, and the multi-sine was either a 30- or 60-s excitation with frequency content from 4 to 25 Hz. The SCT began with open-loop test runs and concluded with closed-loop test runs for all test configurations. For the open-loop tests, all loops were broken before the control law commands were introduced to the actuators. This method allowed measuring the command-to-surface and surface-to-sensor transfer functions and obtaining airframe response characteristics for updating the models used in the ASE analysis. The FES excitations stepped through the different surfaces for each test run with varying amplitudes up to 2 degrees (deg). The closed-loop test runs evaluated the gain and phase margin estimates from the open-loop tests. Table 6 shows the test log for all the test runs done for configuration 3, zero usable fuel on SSS.

Table 6: X-59 SCT test log for configuration 3, zero usable fuel on soft support system.

Run date	Start time	Run #	Test type	Excitation input	Excitation signal	Control law		Condition ID
					Type	Degrees	Symmetry	
2024-03-12	10:51 AM	40	Open loop	Rudder	Multi-sine	2	N/A	11008
2024-03-12	11:00 AM	41.1	Open loop	Ailerons	Sine sweep	0.2	Antisym	11008
2024-03-12	11:03 AM	41.2	Open loop	Ailerons	Sine sweep	0.4	Antisym	11008
2024-03-12	11:15 AM	41.6	Open loop	Ailerons	Multi-sine	0.5	Antisym	11008
2024-03-12	11:19 AM	41.7	Open loop	Ailerons	Multi-sine	1	Antisym	11008
2024-03-12	11:22 AM	41.8	Open loop	Ailerons	Multi-sine	1.5	Antisym	11008
2024-03-12	11:34 AM	42.1	Open loop	Stabs	Sine sweep	0.2	Sym	11008
2024-03-12	11:37 AM	42.2	Open loop	Stabs	Sine sweep	0.4	Sym	11008
2024-03-12	11:40 AM	42.3	Open loop	Stabs	Sine sweep	0.6	Sym	11008
2024-03-12	11:45 AM	42.4	Open loop	Stabs	Sine sweep	0.8	Sym	11008
2024-03-12	11:48 AM	42.5	Open loop	Stabs	Sine sweep	1	Sym	11008
2024-03-12	11:51 AM	42.6	Open loop	Stabs	Multi-sine	0.5	Sym	11008
2024-03-12	11:54 AM	42.7	Open loop	Stabs	Multi-sine	1	Sym	11008
2024-03-12	11:58 AM	42.8	Open loop	Stabs	Multi-sine	1.5	Sym	11008
2024-03-12	12:17 PM	42a	Open loop	Stabs	Sine sweep	0.4	Sym	11008
2024-03-12	12:20 PM	42b	Open loop	Stabs	Sine sweep	0.4	Sym	11008
2024-03-12	12:27 PM	43	Closed loop	Rudder	Multi-sine	2	N/A	11008
2024-03-12	12:32 PM	44	Closed loop	Ailerons	Multi-sine	1	Antisym	11008
2024-03-12	12:37 PM	45	Closed loop	Stabs	Multi-sine	1	Sym	11008
2024-03-12	12:43 PM	46	Closed loop	Rudder	Multi-sine	2	N/A	2x gain
2024-03-12	12:49 PM	47	Closed loop	Ailerons	Multi-sine	1	Antisym	2x gain
2024-03-12	12:53 PM	48	Closed loop	Stabs	Multi-sine	1	Sym	2x gain
2024-03-12	12:57 PM	42a.1	Open loop	Stabs	Multi-sine	1	Sym	11008
2024-03-12	1:00 PM	42b.1	Open loop	Stabs	Multi-sine	1	Sym	11008

The freeplay test, GVT, and SCT all provided experimental measured data to update, tune, and correlate the FEM to become a validated FEM that was used for all the aeroelastic and ASE analyses.

6 X-59 AEROELASTIC ANALYSES

Lockheed Martin performed numerous aeroelastic analyses in order to show that the aircraft was airworthy for flight. Over the lifecycle of the program, as the FEM was correlated with testing results, the analyses were repeated and presented at different airworthiness reviews. The reduced-order A-Set FEM was used for the aeroelastic and ASE analyses. The results shown in this section use the final tuned FEM from all structural ground testing. Note, this section only highlights the larger aeroelastic analyses efforts for the flight program. Smaller analysis tasks such as panel flutter, divergence, control surface buzz, and freeplay will not be discussed but were still performed as part of the aeroelastic airworthiness clearance effort.

6.1 Modal Analysis

Modal analysis was conducted using the structural reduced-order FEM with the A-Set grids to analytically determine the natural frequencies and mode shapes of the aircraft. Structural modes of the unconstrained (or free-free) aircraft were calculated for many different weight and center of gravity (CG) conditions. Table 7 shows analytical modal results for the four fuel conditions at nominal

CG, with single-failed actuators. In the Table 7, asterisks denote the ASE notch filter modes (*) and the primary flutter modes (**). The “Mid-Heavy” weight configuration includes ~53% fuel loading for a gross weight of 22,000 lb.

Table 7: X-59 free-free FEM structural modes at various fuel weights at nominal CG.

Mode	Symmetry	Min wt [Hz]	LPLW [Hz]	Mid-Heavy [Hz]	DTOW [Hz]
Fuselage 1st bending *	Sym	5.78	5.69	5.52	5.38
Fuselage 2nd Bbnding *	Sym	11.25	10.98	10.62	10.41
Fuselage 3rd bending *	Sym	19.49	19.37	19.10	18.83
Wing 1st bending *	Sym	9.42	8.96	8.53	8.39
Wing 1st torsion	Sym	27.43	25.20	21.35	20.42
Wing 2nd bending	Sym	28.82	28.33	27.50	29.49
Stab rotation *	Sym	17.25	17.24	17.17	17.09
Stab bending	Sym	30.56	30.39	31.65	31.43
Flap rotation	Sym	23.23	23.34	23.20	23.11
Canard bending	Sym	25.81	25.89	25.84	25.82
Aileron rotation	Sym	38.54	35.72	33.70	32.11
T-tail pitch	Sym	37.64	37.66	37.65	37.65
Fuselage 1st bending	Antisym	7.45	7.42	7.35	7.29
Fuselage 2nd bending	Antisym	12.92	12.71	12.48	12.38
Fuselage 3rd bending	Antisym	24.25	24.07	23.86	22.42
Wing 1st bending	Antisym	13.24	12.60	11.65	10.88
Wing 1st torsion	Antisym	27.89	23.58	21.60	21.26
Wing 2nd bending	Antisym	29.53	29.02	28.30	28.03
Stab rotation **	Antisym	17.00	16.91	16.79	16.72
Stab bending	Antisym	28.52	28.44	28.36	28.33
Flap rotation	Antisym	23.33	23.26	23.11	22.95
Canard bending	Antisym	24.46	26.65	25.55	25.07
Aileron rotation	Antisym	36.39	34.63	33.22	32.18
T-tail pitch	Antisym	39.75	39.52	40.05	39.87
VT 1st bending	Antisym	10.57	10.31	10.05	9.94
Aft fuselage torsion **	Antisym	19.73	19.52	19.19	18.98
VT torsion	Antisym	33.36	33.26	33.62	33.60
Rudder rotation	Antisym	47.34	46.90	47.37	46.88

* Note: Structural notch filter modes (*); primary flutter coupling modes (**); and landplane landing weight (LPLW).

Figure 15 shows some of the critical mode shapes for the notch filters and the flutter modes for the DTOW condition at nominal CG. Two colored versions of the model are overlaid to better highlight the mode shape deflections. The light brown shaded model is undeformed, and the cyan-shaded model displays the deformed mode shapes. The mode shapes represent a visual of the relative displacement

of the aircraft when it vibrates at a specific natural frequency; however, they are not intended to show the exact physical deflection magnitude.

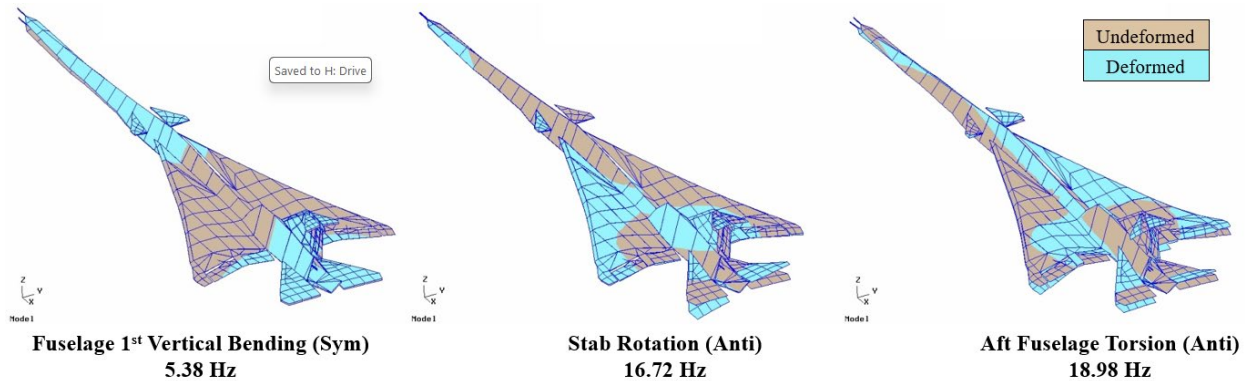


Figure 15: Some X-59 critical mode shapes for DTOW configuration at nominal CG.

6.2 Flutter Analyses

Numerous aeroelasticity / flutter linear and non-linear analyses were done using multiple aerodynamic models and codes for independent verification of the unique X-59 design, but all analyses used the same structural reduced FEM with the A-Set DOFs.

- Linear aeroelastic models
 - NASTRAN
 - ZAERO
- Non-linear CFD-based aeroelastic models
 - FUN3D
 - AEROM
 - Linearized frequency domain (LFD)

6.2.1 Linear Flutter Analysis

Lockheed Martin performed the majority of the linear flutter analysis using NASTRAN-generated mass, stiffness, and aerodynamic influence coefficients (AIC) at the A-Set. The NASTRAN data is imported to the LM in-house p-k flutter code that is used to solve the flutter equation for the baseline FEM and to investigate potential off-nominal conditions in the mass, stiffness, or aerodynamic forces. Several independent flutter analyses were performed by NASA using ZAERO to compare and confirm the LM results due to the unique, one-of-a-kind aircraft design.

For the X-59 aircraft, the aerodynamic forces for the aeroelastic analysis are generated by NASTRAN using the doublet-lattice method for subsonic conditions and the acceleration-potential method for supersonic conditions. The same NASTRAN unsteady aerodynamic model is used for both methods.

To represent the geometry of the vehicle in NASTRAN, the following considerations were made:

- Lifting surfaces – Canards, wings, tails, and control surfaces are modeled with flat-plate panels
- Fuselage – Represented by its vertical and lateral projections with flat-plate panels

- Interpolation – Splines based on the structural reduced model A-Set grids are used to map deflections and forces between the structural and aerodynamic degrees of freedom

The NASTRAN unsteady aerodynamic model, Figure 16, consists of 4,178 boxes, created by subdividing panels to calculate local pressures and forces. The unsteady aerodynamic influence coefficients (AICs) were generated at sixteen reduced frequencies unique to each Mach number; the analysis Mach number range was Mach 0.2 to Mach 2.0 (specifically: Mach numbers 0.2, 0.3, 0.4, 0.5, 0.6, 0.7, 0.8, 0.9, 0.95, 1.05, 1.1, 1.2, 1.3, 1.4, 1.55, 1.7, and 2.0).

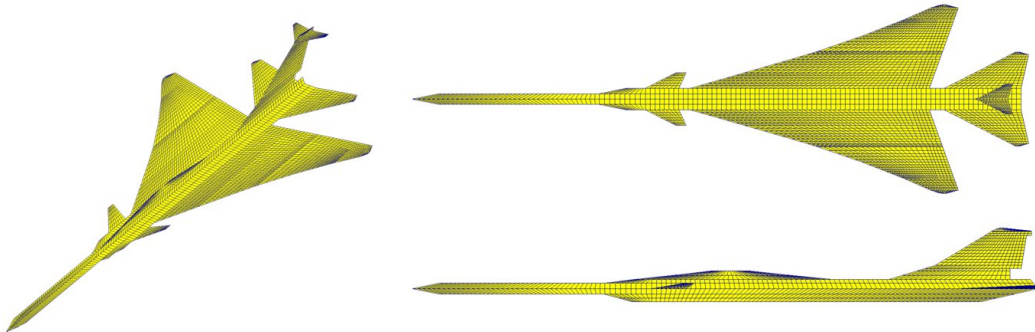


Figure 16: X-59 NASTRAN unsteady aero model.

Figure 17 shows the ZAERO unsteady aerodynamic model which consisted of 10,236 elements and modeled the bodies of revolution for the long fuselage. The ZAERO uses methods like the doublet-lattice method to compute aerodynamic loads on slender bodies in various flow regimes. The ZAERO flutter analysis was done for a smaller Mach number range from Mach 0.9 to 2.0 (specifically: Mach numbers 0.9, 0.95, 0.97, 0.99, 1.05, 1.2, 1.4, and 2.0).

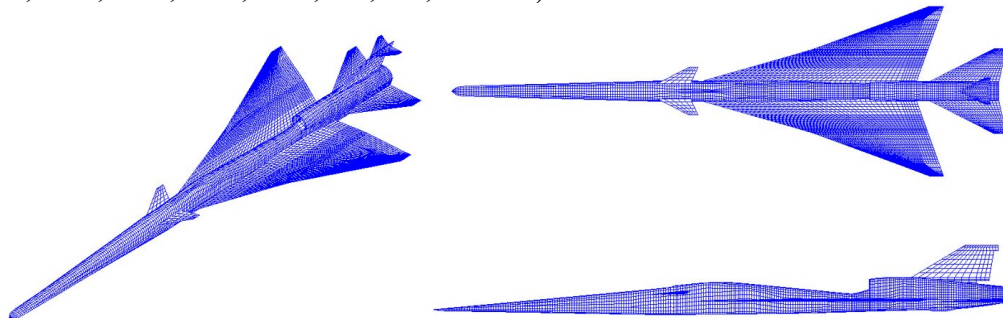


Figure 17: X-59 ZAERO unsteady aerodynamic model.

The X-59 flutter analysis used the p-k method which is an iterative aeroelastic solver that determines aircraft stability by calculating damping and frequency across airspeeds. The method uses doublet lattice aerodynamics to solve complex eigenvalues and tracking modes to find the speed where damping becomes positive at matched conditions of Mach number, velocity, and air density for a chosen speed range.

The X-59 flutter analysis used the following parameters:

- Free-free FEM boundary conditions – all six rigid-body modes were included
- Modes up to 60 Hz were included

- Structural damping of 0.02 g was assumed
- Flutter speeds are calculated where a root crosses the $g=0$ axis (damping)
- Analysis Mach numbers = 0.6, 0.8, 0.9, 0.95, 1.05, 1.1, 1.2, 1.3, 1.4, 1.55, 1.7, 2.0
- Analysis carried out to 1,500 KEAS
- Transonic knockdown factor -10% applied to velocities for analyses at Mach $0.9 < \text{Mach number} < \text{Mach } 1.2$
- Eleven aircraft weight-CG conditions
- Mach 2.0 provides the 15% flutter margin on the design limit of Mach 1.7

The results of the flutter analysis show no flutter instabilities of the canards, rudder, or wings out to 1,500 KEAS. The main flutter mechanism on the X-59 aircraft is a coupling of the antisymmetric stabilator rotation mode and the aft fuselage torsion mode, as shown in Figure 18. Flutter of the T-tail is predicted as a high velocity ($>1,200$ KEAS) flutter mechanism with a minimum flutter speed in the transonic region showing 167% flutter margin. The stabilator flutter mechanism is a hump mode and only occurs at Mach 2.0 with 52% flutter margin, which is well above the 15% requirement. The flap exhibits a flutter hump mode at supersonic conditions with significantly reduced damping within the X-59 flight envelope, though not unstable or catastrophic, see Figure 19. This flap hump mode combined with high-measured flap freeplay may result in limit cycle oscillation (LCO) of the flap at Mach 1.2. The baseline flutter analysis assumed 0.02 g damping for all modes; however, when the flutter analysis uses the measured GVT flap damping (~ 0.05 g), results are significantly improved, and the flap hump mode is eliminated. During envelope expansion flights, data collected during the flutter excitation maneuvers will be utilized to verify damping of the X-59 flutter critical modes (stabilator rotation at and aft fuselage torsion / stabilator bending at 20 Hz)

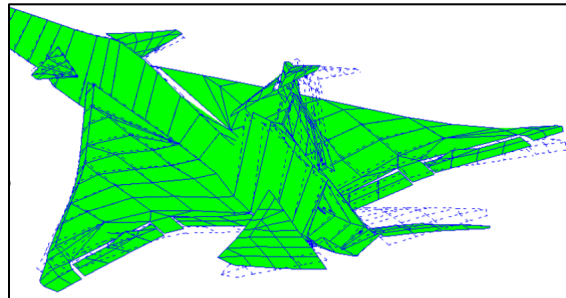


Figure 18: X-59 flutter mechanism, coupling of antisymmetric stabilator rotation and aft fuselage torsion modes.

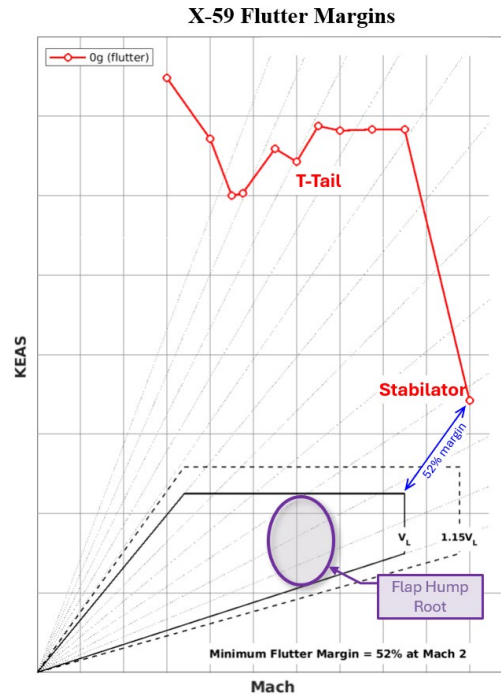


Figure 19: X-59 flutter margins, flap hump mode is eliminated using GVT flap damping.

6.2.2 Non-Linear CFD-Based Flutter Analyses

For the X-59 aircraft, non-linear computational fluid dynamics (CFD)-based flutter analyses were done in FUN3D, AEROM, and linearized frequency domain (LFD) solutions to verify the linear flutter analysis at transonic speeds. A CFD-based flutter analysis is expensive, computationally heavy, and time consuming but may be preferred over linear methods primarily because linear models can be nonconservative and inaccurate in complex flight regimes, such as the transonic regime. While linear flutter methods (e.g., doublet-lattice method) are computationally efficient, they can fail to capture critical nonlinear and viscous phenomena. With the unique design of X-59 aircraft it was uncertain whether linear flutter analysis would be sufficient for the transonic region. The NASA Langley Research Center (LaRC) (Hampton, Virginia, USA) focused on the CFD-based flutter analyses [5] while LM focused on the linear flutter analysis. Multiple CFD-based flutter solutions were employed because each had its strengths and weaknesses.

The X-59 CFD-based flutter analyses used the following solutions:

FUN3D

- Steady rigid at each Mach number (M)
- Static aeroelastic at each M/dynamic pressure (Q)
- Dynamic aeroelastic at each M/Q
- Dynamic aeroelastic at each M/Q for different initial conditions

AEROM

- NASA-patented AEROM software
- Unsteady aerodynamic system ID solution per Mach number
- Generate linearized state-space models of unsteady aero
- Couple with structural dynamic state-space model (modal frequencies and dampings)

- Generate root locus plots and SIMULINK time histories (comparison with FUN3D)

Linearized frequency domain (LFD)

- Steady rigid at each Mach number
- LFD at each Mach number
- Generate linearized generalized aerodynamic forces (GAFs)
- Use flutter solvers (p-k, g)

The FUN3D solutions are the most accurate; however, they are also the most expensive and time-consuming. Each Mach number and dynamic pressure point is independently calculated and checked for stability, with multiple dynamic pressure points needed for a given Mach number to verify that there is no indication of flutter. For example, Figure 20 shows the generalized displacement history for two structural modes at $M=0.95$ and a dynamic pressure of 216 psf. The FUN3D analysis was performed for Mach numbers equal to Machs 0.95, 0.97, 0.99, 1.05 and 1.40.

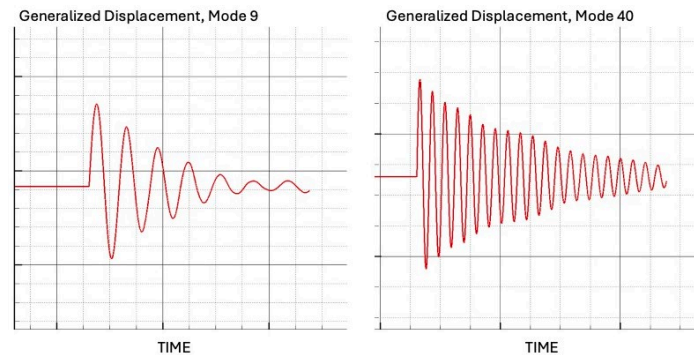


Figure 20: FUN3D generalized displacement time history for two modes (modes 9 and 40) for $M=0.95$ and $Q=216$ psf.

The AEROM solution is much more efficient than a FUN3D solution. For a given Mach number, a reduced order model (ROM) can be constructed for the cost of a single FUN3D solution; the resulting model is valid for a large range of dynamic pressures. As the name implies, however, the ROM approximates the full system. Typically, a ROM is validated by comparing the results with a FUN3D test case. Figure 21 shows the comparison of generalized forces for an AEROM and a FUN3D solution for structural mode 7 at $M=0.99$ and dynamic pressure of 216 psf. Once the ROM has been generated and validated, analysis can be done in the frequency domain, such as the root locus plot using 49 structural modes for $M=0.99$ with dynamic pressure ranging from 0 to 432 psf, as shown in Figure 22.

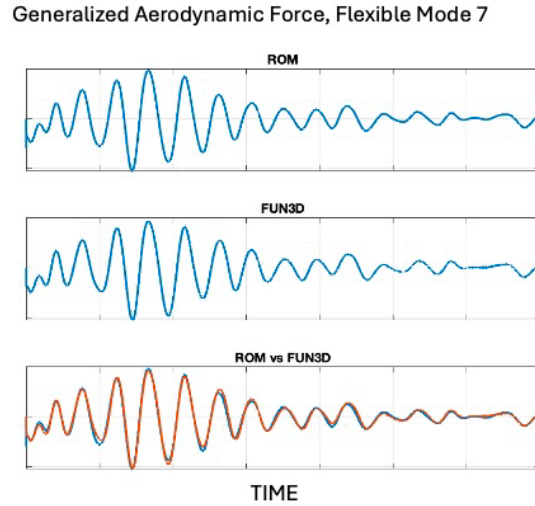


Figure 21: AEROM versus FUN3D generalized displacements time history for Mode 7 for $M=0.99$ and $Q=216$ psf.

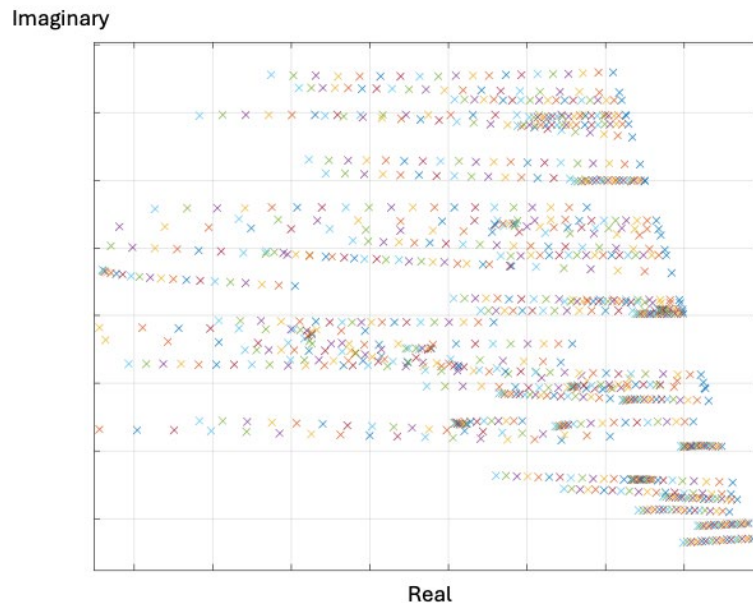


Figure 22: AEROM generated root locus for $M=0.99$ and Q ranging from 0 to 432 psf, for 49 Modes.

The LFD method has the advantage of working in the frequency domain, which makes it easier to compare to linear methods; however, since linearization is performed, like the ROM, it approximates the full system too. The LFD method is also computationally expensive, having a much higher computational memory requirement than the other two methods. Because of the expense, LFD analyses were done only at two Mach numbers: Mach 0.97 and Mach 1.05. To help validate the LFD results, a comparison between the linear ZAERO results and non-linear LFD was performed at $M=1.05$ (outside of the transonic regime); Figure 23 shows the velocity versus damping (V-g) flutter plots for the two methods. When applied to the transonic regime, the non-linear LFD solution showed no indication of flutter within the X-59 flight envelope.

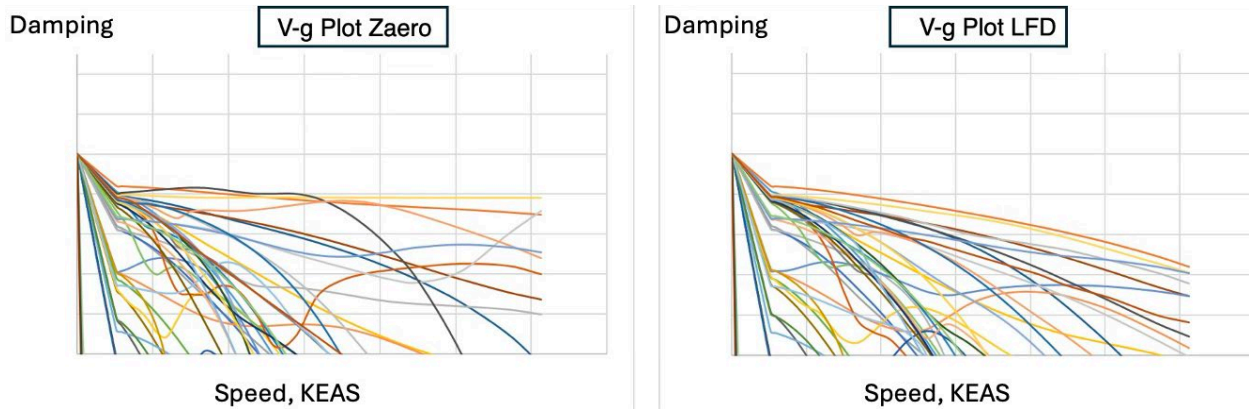


Figure 23: Comparison of linear ZAERO versus non-linear LFD velocity versus damping (V-g) plots.

The other two non-linear FUN3D and AEROM CFD-based flutter analyses likewise showed no indication of flutter within the X-59 flight envelope and similar flutter mechanism. It was surprising to learn that the linear flutter analysis was sufficient for the transonic region of the X-59 aircraft, given its unique, long-nosed design.

7 X-59 AEROSERVOELASTICITY ANALYSIS

Actuation of flight control surfaces generates inertial and aerodynamic forces that excite the aircraft structural vibration modes. If these oscillations are picked up by onboard sensors (accelerometers and rate gyros), they are fed back into the control laws. Depending on the system gains, this coupled response of the aerodynamics, control system, and airframe structural modes can trigger command instabilities and excessive structural deflections, this phenomenon is known as aeroservoelastic (ASE) interaction. Aircraft structural mode responses can be mitigated using two different ASE analyses methods: (1) geometric filtering that involves installing sensors where motion for the most critical mode is minimized; and (2) structural notch filters, which are used in the flight control laws to reduce motion signal gains at specific modal frequencies.

The long fuselage of the X-59 aircraft made the lowest frequency, fuselage 1st bending mode (~ 5.5 Hz), the primary concern from the flexible structural modes. Flexible mode response can be reduced through geometric filtering, which involves installing sensors at locations where structural motion is minimal. The X-59 aircraft optimized the placement of its forward inertial measure unit (IMU), which measured accelerations (N_x , N_y , N_z) by placing the unit near the fuselage 1st bending mode node point (or zero vertical motion) to minimize the feedback response. The aft IMU, which measured gyroscope angular rates (P_b , Q_b , R_b) and the optimized location to minimize the feedback response, was installed near the fuselage 1st bending mode anti-node point (or zero rotation), as shown in Figure 24. These optimized IMU locations minimized the amount of structural notch filtering needed at lower frequencies to alleviate the unfavorable phase shift that can reduce the rigid-body control margins.

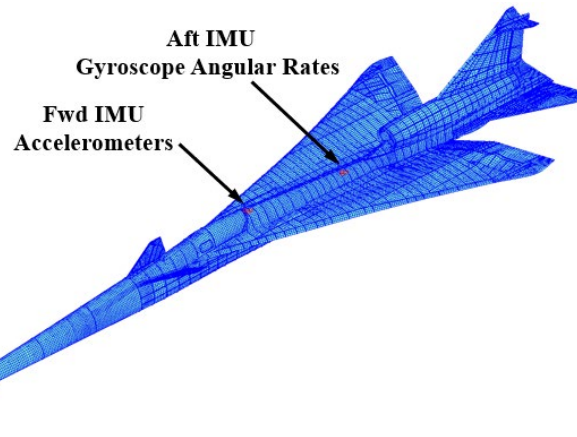


Figure 24: X-59 geometric filtering for inertial measure unit (IMU) locations.

The ASE analysis is done using a state-space approach to model the rigid-body and flexible airframe motions due to control surface actuator commands. The airframe state-space model is derived from the modes of both mass and stiffness matrices at the reduced A-Set FEM nodes degrees of freedom. Surface actuators are modeled in MATLAB Simulink with transfer functions to represent frequency roll-off characteristics. The unsteady aerodynamic AIC matrices in the frequency domain are fit using a rational function approximation to obtain an explicit function in the time (or Laplace) domain.

The aircraft flaps and T-tail are only trimmed surfaces and not involved in the active flight control system; therefore, no frequency response models for the flaps and T-tail were needed for the ASE analysis. The active control surface (aileron, stabilator and rudder) rotation modes were tuned to the measured GVT data to represent accurate modal responses in the state-space aircraft models.

The ASE analysis used the FEM and state-space models to minimize the feedback of the structural vibration response through the flight control system. Due to the extremely long fuselage, several structural frequency notch filters are applied to the X-59 control laws to reduce the feedback magnitude.

The X-59 structural notch filters used in the flight control laws:

- Fuselage 1st bending (Sym)
- Wing 1st bending (Sym)
- Fuselage 2nd bending (Sym)
- Stabilator rotation (Sym)
- Fuselage 3rd bending (Sym)

Simulink provides a block diagram approach for connecting the various components of the ASE model such as a coupled airframe/aerodynamic model, actuator frequency response characteristics, CLAW commands, CLAW feedback signals, and test signals (or external excitation inputs). The X-59 Simulink model was constructed so that the input/output signals will create the desired transfer functions (TF): (1) the surface to sensor responses (open-loop plant data); (2) the sensor to surface command responses (CLAW gains); and (3) the surface input to surface command responses (Bode data for stability).

The stabilator actuator transfer function, which converts the CLAW actuator position command to apply force at the stabilator actuator ends, is shown in Figure 25. The CLAW position command passes through the frequency response (roll-off) function and is then converted from degrees to inches based on the stabilator crank arm length. Next, the old position value is subtracted so the actuator force will only be what is needed to move to the new stabilator position. The freeplay effect function has two options (selected by variable $fp_type_STAB_LH$): (1) use a freeplay block which puts out zero force when inside freeplay (can only be used for time simulation); or (2) a reduction gain to simulate a stiffness reduction effect of freeplay (used for frequency analysis, i.e., Bode). The freeplay effected displacement is then turned into an actuator force by a gain equal to the stabilator dynamic stiffness.

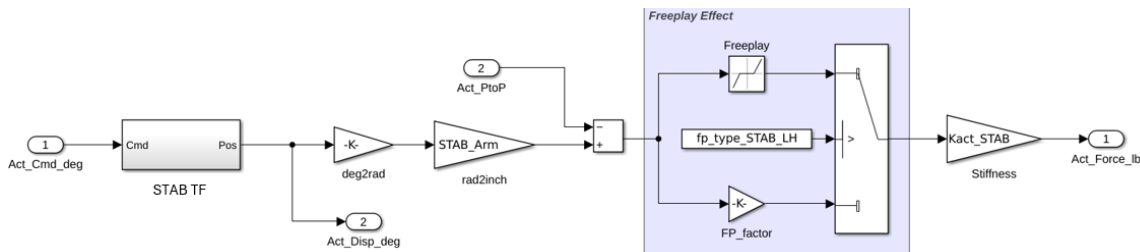


Figure 25: X-59 ASE state-space model of the stabilator actuator block.

The final ASE analysis incorporates the most recent CLAW which are expected to remain frozen through the start of flight-testing. The ASE results, summarized below, represent the Up & Away condition for the nominal and single fail IMU. In all cases, the aileron and rudder command paths satisfy all ASE stability requirements. The stabilator command path minimum Up & Away gain margins for the nominal and single failure IMU states are shown in Figure 26 and contain the stop-light grade chart showing the minimum gain margin values across the flight envelope. Some of the subsonic conditions, between 250 to 350 KEAS, have gain margins that are less than 8 dB but greater than 6 dB (purple grade); these are conditions where the rigid-body phase-lag issues required relaxation of the structural notch filters, as previously discussed. The stabilator maximum Up & Away peak gains for the nominal and single-failure IMU states are shown in Figure 27 and contain the stop-light grade chart showing maximum peak gain values across the flight envelope. All conditions within the design flight envelope are at or below -6 dB. Future flight-testing will inform the analysis regarding the effect of aerodynamic damping, which is expected to be greater than predicted with the linear aerodynamic panel methods.

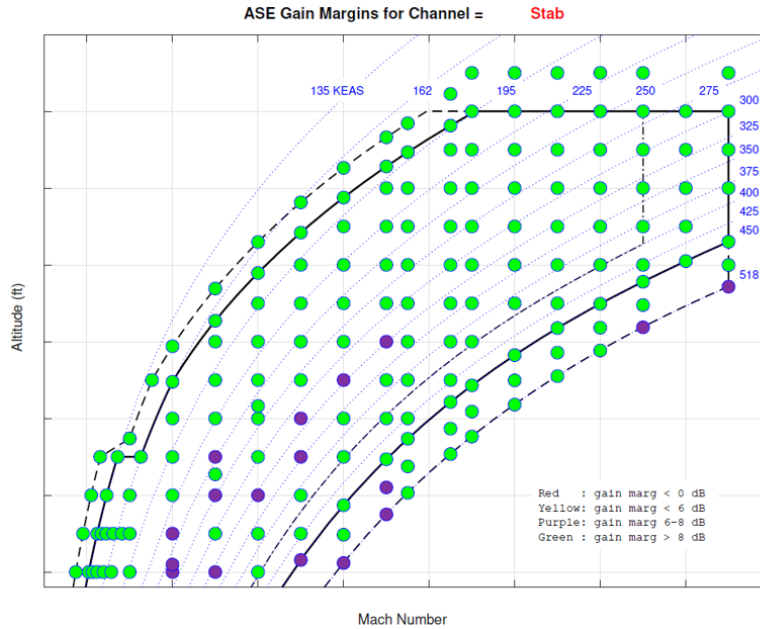


Figure 26: X-59 stabilator Up & Away gain margins, stop light grades for nominal- and single-fail IMU.

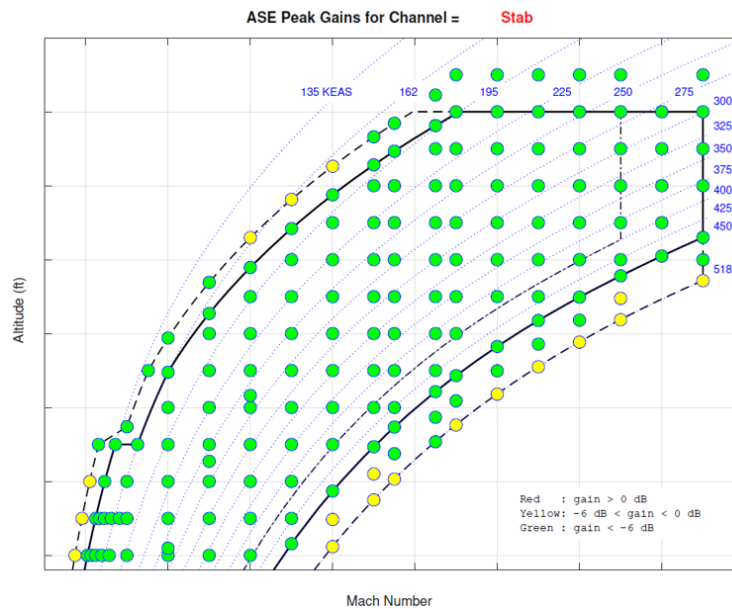


Figure 27: X-59 stabilator Up & Away peak gains, stop light grades for nominal- and single-fail IMU.

8 X-59 FLUTTER AND AEROSERVOELASTIC FLIGHT-TEST METHODOLOGY

Along with all the structural ground testing and analyses, prior to first flight, numerous integrated ground tests and taxi tests were completed. The X-59 envelope expansion flights are an interdisciplinary plan of test points and maneuvers using a build-up approach in Mach number and equivalent airspeed (KEAS) to clear the full operational envelope for the desired Mach 1.4 at 55,000-ft test point for the community response testing. The flutter flight-test objectives are to verify that the X-59 aircraft is free from flutter, buzz, divergence, and any ASE instabilities. Flight data will be used

to correlate ASE analysis showing required stability margins are satisfied. The X-59 aircraft successfully flew its first flight on October 28, 2025, and will continue to expand the envelope using the flight flutter excitation system (FES) to excite the structural modes of the aircraft while flight instrumentation is monitored real time in the control room.

8.1 Flight Test Approach

The X-59 flutter and ASE clearance flight-testing utilizes a build-up approach along constant equivalent airspeed lines. Flight-testing targets three specific KEAS lines, 275, 350, and 425 KEAS, to clear out the envelope up to Mach 1.55. Along each airspeed line, the Mach number will be increased by increments of Mach 0.1. Progression to a higher airspeed line is contingent upon completing the equivalent Mach number at the lower airspeeds. Flying a constant KEAS line allows dynamic pressure to be constant while only increasing the Mach number. Figure 28 shows the flutter flight-test envelope expansion plan with the flutter/ASE test points (pink circles and blue squares) along the constant KEAS lines, and at each test point, the pilot will fly a steady 2 knots/second acceleration to the test point condition, a 1-g trimmed steady-state maneuver for 60 s, and FES maneuvers.

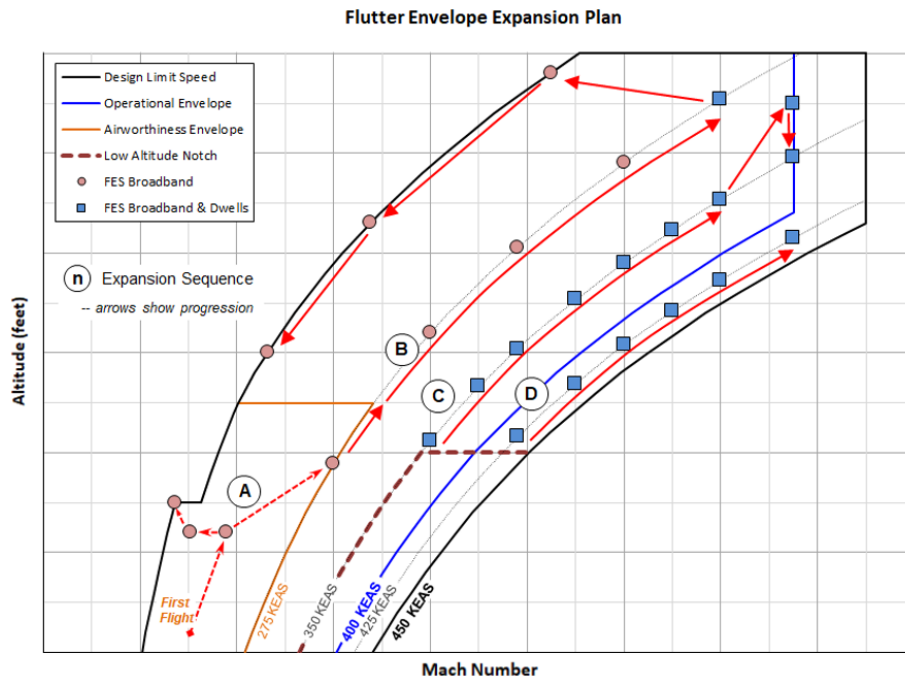


Figure 28: X-59 flutter flight test envelope expansion plan.

Flight data from the onboard accelerometers, vehicle management computers (VMC) control law commands, IMU body accelerations and rates, and control surface actuator positions will be collected during a 1-g trim ambient response and FES maneuvers. Ambient data will identify the frequency and damping of the X-59 primary structural modes while ensuring no adverse aeroelastic behavior occurs as the flight envelope is expanded. The FES data will be utilized specifically to verify damping for flutter critical modes and to evaluate aeroservoelastic margins.

An initial assessment of ASE response and stability was conducted during the high-speed 100-kgs taxi testing prior to first flight from an FES ± 2 -deg multi-sine 40-s excitation of the stabilators. The

Bode plot of the stabilator command path showed a peak gain of -11.3 dB at 6.7 Hz, which meets the requirement for ASE peaks to be less than -6 dB.

8.2 Flutter Excitation System (FES)

The flight flutter excitation system (FES) executes preprogrammed oscillatory signals to the control surface actuators (stabilators, ailerons, and rudder) to excite the aircraft structural response in flight for engineering evaluation. The desired FES excitation settings are defined in the preflight briefing, but settings such as excitation type, control surface grouping, frequencies, amplitudes, and durations may be modified in flight based on observed flight data and with direction from the flutter lead. When on a test point, the pilot will load the desired FES run, the FES setting will be verified and cleared by the control room with the FES interactive analysis and display system (IADS), as shown in Figure 29, the pilot will then verbalize and engage the FES signal. The control room will observe the aircraft response during the FES maneuver and evaluate the control system ASE gain margins and critical mode identification. Then, the control room flutter lead may request a repeat of the excitation, make changes to FES settings, or continue to the next planned maneuver.

The X-59 flutter excitation system has the following signals:

- Broadband: Multi-sine signals expected to be the primary type, with random and sine sweeps as back-up options
- Dwell/decay: A single-frequency burst with sudden stop to estimate mode damping from a decay rate

FES, 1st Program is Running



FES Slot	1	2	3	4	5
Prog #	111	112	113	107	106
Type	FES-2	FES-2	FES-2	PID-2	PID-1
Rud	-	-	On	On	On
Ail	-	Both	-	Both	Both
Stab	Both	-	-	Both	Both
Amp	1.5	1.5	1.5	2.0	2.0
Filter				5	4
F1 (Hz)					
F2 (Hz)					
Duration	30	30	30	60	20
nBursts					
Interval					

Figure 29: Control room flutter excitation system (FES) settings in IADS table.

During initial low-speed envelope testing, optimal multi-sine settings will be done by first exciting individual control surface pairs (symmetric stabilator or anti-symmetric aileron) followed by simultaneous multi-surface excitation, and finally, adding rudder excitation. Post-flight analysis will evaluate if a single FES run with multiple surfaces provides sufficient flutter and ASE data to reduce the total excitation time and increase test efficiency for subsequent conditions. Different FES signals at a lower frequency range, targeting the aircraft rigid-body modes, will also be done by the flight controls and flying quality engineers to assess stability margins and collect data for model calibration.

8.3 Flight Instrumentation

The flutter flight instrumentation on the X-59 aircraft consists of 39 accelerometers. The accelerometers are mounted on the canards, wings, and tail; flight control surfaces measure the out-of-plane response while the accelerometers on the fuselage at the nose, cockpit, aft wing carry-thru, and tail measure vertical and lateral responses. The X-59 accelerometer locations are shown in Figure 30, and the layout is mostly symmetrical, but the left side of the aircraft has slightly more accelerometers for mode shape extraction. The ASE flight instrumentation consists of the VMC bus parameters of the CLAW command signals, body accelerations and rates, and control surface actuator positions for ASE margin calculations. The flutter accelerometers are recorded onboard and telemetered to the control room at 500 samples per second (sps); however, the ASE parameters are lower at 100 sps.

The parameters that are critical to the safe execution of a test point are designated as safety of test (SOT) parameters. In Figure 30, 22 out of the 39 accelerometers are SOT parameters, as indicated with the filled-in red and green circles, and the other 17 accelerometers are not as critical and will be used for post-flight analysis purposes. There are 10 SOT, ASE, and CLAW input and output parameters. All of these SOT parameters must be monitored real time for the entire flight and must be operational in order to proceed with any flutter or ASE flight-test points.

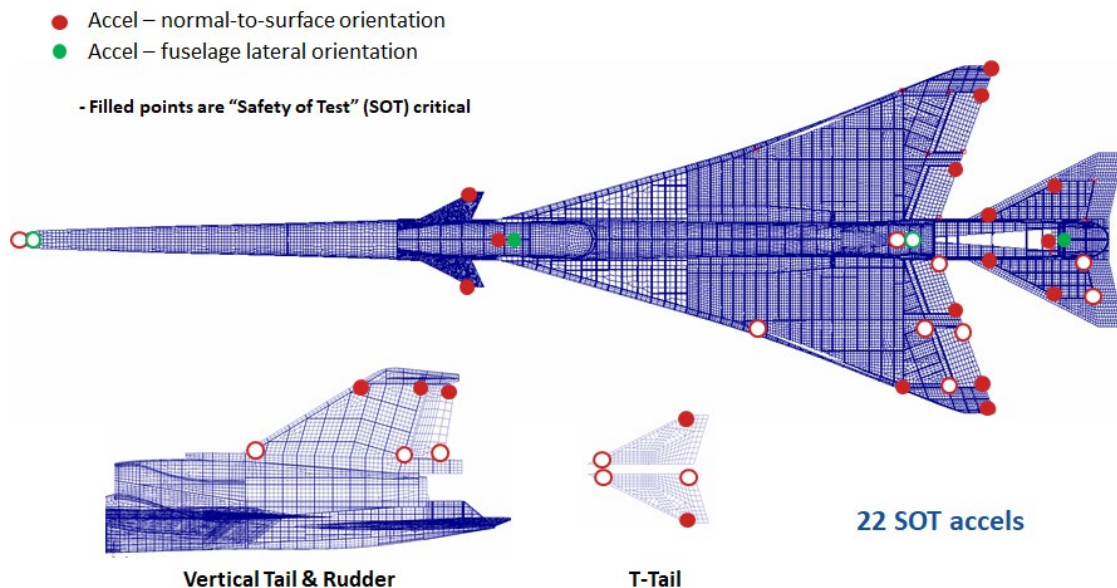


Figure 30: X-59 flight flutter accelerometer locations.

8.4 Control Room Monitoring

All of the X-59 envelope expansion flights will be monitored and controlled from the NASA Armstrong Mission Control Center (MCC). Instrumentation and video will be telemetered down to the control room. Project engineers will staff the MCC in order to monitor real-time data and verify the aircraft response is consistent with predictions as flights expand out the envelope to the full

operational envelope. Clearance to proceed to the next test point requires verification from the control room that the aircraft response aligns with predefined flight-test continuation criteria and predictions.

For the X-59 first flight and initial airworthiness checkout flights, flutter/ASE will have three seats in the control room for monitoring data on custom-built interactive analysis and display system (IADS) screens. For envelope expansion flights, flutter/ASE will have five seats in the MCC. The flutter/ASE IADS monitor plan for each station is shown on Figure 31. The flutter/ASE custom-built IADS screens for each of the five seats were laid out so that each station has a dedicated part of the airframe component to monitor along with dedicated roles and responsibilities. The flutter lead or call sign “Flutter1” is the primary communicator back to the test conductor and coordinates and confirms FES settings, maintains the log documentation of test points and event markers as maneuvers are completed, polls the flutter team to confirm that a maneuver is good, and has all the flutter/ASE SOT parameters to monitor when call sign “Flutter2” through “Flutter5” are busy completing frequency and damping or ASE margin calculations following a trim maneuver or FES run. The Flutter1 station is the only person with two IADS screens, which allows the lead to watch the main lead IADS screen and another IADS screen with all the SOT parameters. Flutter2 sits on the left side of Flutter1 and is responsible for anything related to the stabilator and aft fuselage flutter. Flutter3 sits on the right side of Flutter1 and also has two screens: one is an IADS monitor dedicated to ASE stabilator, and the second is a real-time MATLAB monitor for ASE stability margin calculations, which are computed slightly different than IADS and has more options for cleaning up the Bode plots if necessary. Flutter4 has one monitor but will choose between two different IADS screens depending on the flight condition being either a low- or high-speed test point. At lower speeds the aileron ASE IADS screen will be monitored, otherwise at higher speeds the flutter vertical tail and T-tail IADS screen will be monitored. Flutter5 is responsible for monitoring flutter on the canards and wings.

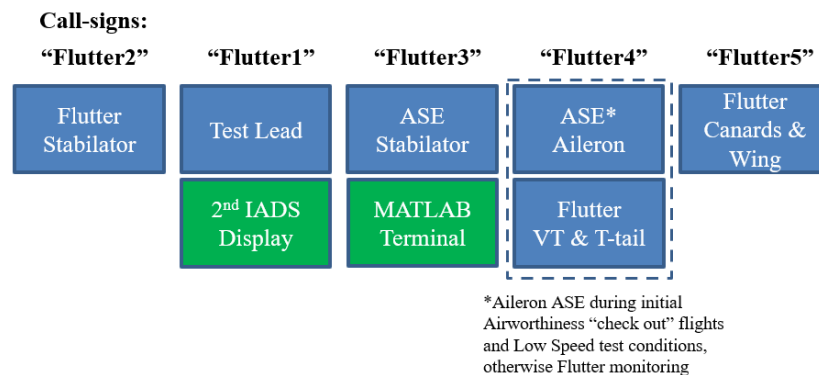


Figure 31: Flutter/ASE monitoring plan layout.

During the slow acceleration to the next test condition, each flutter station will be monitoring the time history of their respective airframe component SOT parameters to ensure the oscillation remains damped and not divergent/growing as the Mach number is increased. The 1-g trimmed steady-state 60-s maneuver is done as soon as the pilot is on the desired test condition. Upon the trim maneuver completion, each flutter station, except for the flutter lead who is now responsible for all the SOT parameters, will evaluate and document the critical flutter modes (frequency and damping) using the IADS randomdec capability and power spectral density (PSD) plots. When each flutter station is finished with analysis and documenting, they will resume SOT responsibilities. After completed FES runs, the ASE monitors will complete stability margin calculations through transfer function Bode analysis of CLAW command paths due to FES multi-sine signals, and the Stabilator monitor will

review IADS automated dwell-decay damping from FES bursts on the flutter critical modes. Flutter2 through Flutter5 will report back to Flutter1 upon completion of analysis and give an all-clear for next maneuver if data is looking as expected, or report their concern if data is not as expected. Then, Flutter1 will report back to the test conductor if testing is good to continue or not. This control-room monitor and analysis process will continue as each flutter/ASE test point is flown through the expansion of the flight envelope to finalize the Phase I flights with an X-59 operational flight envelope. The Phase 2 sonic boom signature flights and the Phase 3 sonic boom community response flights will both be flown inside the operational envelope without any flutter/ASE real-time monitoring since the airframe will be fully cleared.

9 CONCLUDING REMARKS

This paper presented the journey of X-59 aircraft to demonstrate adequate aeroelastic and aeroservoelastic margins for airworthiness clearance as well as the flutter flight-test envelope expansion methodology. Initially, there were several aeroelasticity and aeroservoelasticity concerns due to the one-of-a-kind, long, slender fuselage, thin wings, large engine mass, and statically unstable X-59 aircraft design. Over the course of many years, Lockheed Martin (LM) and the National Aeronautics and Space Administration (NASA) structural dynamics engineers continued to improve the X-59 finite element model (FEM) with structural ground test data and run updated flutter analysis with each FEM update. Moreover, updated aeroservoelasticity (ASE) analysis with each of the FEM and control law (CLAW) updates were performed to show that our design requirements were satisfied. Currently, the X-59 aircraft continues envelope expansion flights and hopes to gather the community response test data for regulators as soon as possible. These flights will assist with new rules which may allow for future quiet commercial supersonic flight over land.

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