



AFRL-RQ-WP-TR-2018-0088

**EXPLORING THE RESPONSE OF A THIN, FLEXIBLE
PANEL TO SUPERSONIC TURBULENT FLOW AND
SHOCK BOUNDARY-LAYER INTERACTIONS (SBLI)**

**Stephen Michael Spottswood
Hypersonic Sciences Branch
High Speed Systems Division**

**MARCH 2018
Final Report**

**DISTRIBUTION STATEMENT A. Approved for public release.
Distribution is unlimited.**

**AIR FORCE RESEARCH LABORATORY
AEROSPACE SYSTEMS DIRECTORATE
WRIGHT-PATTERSON AIR FORCE BASE, OH 45433-7542
AIR FORCE MATERIEL COMMAND
UNITED STATES AIR FORCE**

NOTICE AND SIGNATURE PAGE

Using Government drawings, specifications, or other data included in this document for any purpose other than Government procurement does not in any way obligate the U.S. Government. The fact that the Government formulated or supplied the drawings, specifications, or other data does not license the holder or any other person or corporation; or convey any rights or permission to manufacture, use, or sell any patented invention that may relate to them.

Qualified requestors may obtain copies of this report from the Defense Technical Information Center (DTIC) (<http://www.dtic.mil>).

AFRL-RQ-WP-TR-2018-0088 has been reviewed and is approved for publication in accordance with assigned distribution statement.

This report is published in the interest of scientific and technical information exchange and its publication does not constitute the Government's approval or disapproval of its ideas or findings.

REPORT DOCUMENTATION PAGE					Form Approved OMB No. 0704-0188	
The public reporting burden for this collection of information is estimated to average 1 hour per response, including the time for reviewing instructions, searching existing data sources, gathering and maintaining the data needed, and completing and reviewing the collection of information. Send comments regarding this burden estimate or any other aspect of this collection of information, including suggestions for reducing this burden, to Department of Defense, Washington Headquarters Services, Directorate for Information Operations and Reports (0704-0188), 1215 Jefferson Davis Highway, Suite 1204, Arlington, VA 22202-4302. Respondents should be aware that notwithstanding any other provision of law, no person shall be subject to any penalty for failing to comply with a collection of information if it does not display a currently valid OMB control number. PLEASE DO NOT RETURN YOUR FORM TO THE ABOVE ADDRESS.						
1. REPORT DATE (DD-MM-YY) March 2018		2. REPORT TYPE Final		3. DATES COVERED (From - To) 30 March 2016 – 30 March 2018		
4. TITLE AND SUBTITLE EXPLORING THE RESPONSE OF A THIN, FLEXIBLE PANEL TO SUPERSONIC TURBULENT FLOW AND SHOCK BOUNDARY-LAYER INTERACTIONS (SBLI)				5a. CONTRACT NUMBER In-house		
				5b. GRANT NUMBER		
				5c. PROGRAM ELEMENT NUMBER 61102F		
6. AUTHOR(S) Stephen Michael Spottswood				5d. PROJECT NUMBER 3002		
				5e. TASK NUMBER		
				5f. WORK UNIT NUMBER Q185		
7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES) Hypersonic Sciences Branch (AFRL/RQHF) High Speed Systems Division Air Force Research Laboratory, Aerospace Systems Directorate Wright-Patterson Air Force Base, OH 45433-7542 Air Force Materiel Command, United States Air Force				8. PERFORMING ORGANIZATION REPORT NUMBER AFRL-RQ-WP-TR-2018-0088		
9. SPONSORING/MONITORING AGENCY NAME(S) AND ADDRESS(ES) Air Force Research Laboratory Aerospace Systems Directorate Wright-Patterson Air Force Base, OH 45433-7542 Air Force Materiel Command United States Air Force				10. SPONSORING/MONITORING AGENCY ACRONYM(S) AFRL/RQHF		
				11. SPONSORING/MONITORING AGENCY REPORT NUMBER(S) AFRL-RQ-WP-TR-2018-0088		
12. DISTRIBUTION/AVAILABILITY STATEMENT DISTRIBUTION STATEMENT A. Approved for public release. Distribution is unlimited.						
13. SUPPLEMENTARY NOTES PA Clearance Number: 88ABW-2018-1164; Clearance Date: 9 Mar 2018						
14. ABSTRACT There do not currently exist reasonable means to accurately simulate the complex and often nonlinear response of an aircraft structure to the dynamic, high-temperature and transient environment of high-speed flight. While methods do of course exist to predict aspects of the static and dynamic responses of such structures, once the response becomes nonlinear (geometric, material, etc.) and the transient nature of the loading requires a long time history, the simulation problem proves quickly intractable. As a consequence, predictive techniques rely on simplifying assumptions, oftentimes overlooking or losing the ability to capture extreme responses or local effects that lead to damage and failure.						
15. SUBJECT TERMS aerothermoelastic, digital image correlation (DIC), fluid-structure interaction, hypersonics, shock boundary-layer interaction (SBLI)						
16. SECURITY CLASSIFICATION OF:			17. LIMITATION OF ABSTRACT: SAR	18. NUMBER OF PAGES 50	19a. NAME OF RESPONSIBLE PERSON (Monitor) Stephen M. Spottswood 19b. TELEPHONE NUMBER (Include Area Code) N/A	
a. REPORT Unclassified	b. ABSTRACT Unclassified	c. THIS PAGE Unclassified				

TABLE OF CONTENTS

Section	Page
LIST OF FIGURES	II
LIST OF TABLES	IV
1 INTRODUCTION & BACKGROUND	1
2 EXPERIMENTAL OVERVIEW	6
3 TEST PROCEDURE	16
4 FIRST RC-19 TUNNEL ENTRY (2010)	18
5 SECOND RC-19 TUNNEL ENTRY (2012)	22
6 FINAL RC-19 TUNNEL ENTRY (2016 - 2017)	25
7 CONCLUSION	38
LIST OF SYMBOLS, ABBREVIATIONS, AND ACRONYMS.....	39
END NOTES.....	40

LIST OF FIGURES

Figure 1: (a) Geometric (hard-spring) nonlinear response typical of acoustic fatigue [6]; (b) Ceramic matrix composite panel failure (172 dB) [7], and (c) C-130 aluminum aircraft panel failure (160 dB) due only to acoustic/dynamic resonant fatigue.	2
Figure 2: Panel deformation and buckling on the underside of the YF-12 experimental aircraft at the USAF Museum.	3
Figure 3: RC-19 modified test section, denoting flexible panel, shock generator, optical access and details of early “bonded” thin-panel arrangement.	6
Figure 4: RC-19 modified test-section showing pressure taps, shock generator locations and thin panel test specimen.	7
Figure 5: Acceleration PSD showing effect of tunnel bracing, ($P_o = 270$ kPa).	7
Figure 6: Static pressure measurements along the tunnel-wall.	8
Figure 7: Modal test arrangement for the RC-19 thin panel specimen.	9
Figure 8: Boundary layer measurement locations upstream (1 & 2) and downstream (3) on the rigid control specimen.	11
Figure 9: Boundary layer measurement results for 6 tests and three locations denoted in Figure 8.	11
Figure 10: Year 1 3D DIC speckle pattern applied at discrete points across the bonded panel. ..	13
Figure 11: Years 2 & 3 3D DIC pattern applied across the machined panel surface and the tunnel/specimen frame.	13
Figure 12: RC-19 tunnel test arrangement for the first year of experiments.	18
Figure 13: Shadowgraph image of adjustable shock generator in RC-19 Tunnel "Block 4."	19
Figure 14: 3D DIC displacement results with shock generator at two settings - flush with the tunnel wall and elevated 10° . Also shown is the corresponding sample PSP image denoting the shock impingement location on the panel ($P_o = 270$ kPa).	20
Figure 15: RC-19 Tunnel shut-down with normal (λ) shock and corresponding panel strain gage time-history	21
Figure 16: Panel center displacement PSD with corresponding full-field deflected shapes measured using 3D DIC.	22
Figure 17: Sensitivity of panel response to location of Shock Boundary-Layer Interaction (SBLI).	23
Figure 18: Year 2 (a) unheated and (b) heated flow panel center displacement response.	24
Figure 19: RC-19 experimental setup for simultaneous filming (DIC) of the panel dynamic response through the cavity and through the SBLI flow.	26
Figure 20: Full-field PSP (pressure) and TSP (temperature) images corresponding to the flow/SBLI conditions.	27
Figure 21: High-speed shadowgraph image and associated shock dynamics for (a) one point upstream of the wedge shock-generator, and (b) a single point near the location of initial impact on the thin-panel.	28
Figure 22: Captured panel dynamic response (red curve) in down-stream tunnel Kulite pressure measurement.	29
Figure 23: Filming the panel displacement response from the cavity and panel flow-side using a dual-DIC camera arrangement. There is a shock impingement at the panel midpoint, $x/L=0.5$	30
Figure 24: Heated flow test panel specimen with DIC pattern.	31

Figure 25: Transient panel response to start of RC-19 tunnel and heated flow.....	32
Figure 26: Resulting POD vectors and associated proper orthogonal values for Region 1 and 2 full-field displacement time regions in Figure 25.	32
Figure 27: Large amplitude panel response to heated supersonic flow when cavity pressure was reduced by 2.1 kPa (0.6 psi, nominal = 7.8-psi).....	33
Figure 28: Heated flow (12.7 cm/5-in panel) strain displacement and strain response at the panel 1/4-point ($x/L=0.25$). Pressure in cavity reduced by 3 kPa (0.44-psi, nominal = 7.9-psi)	34
Figure 29: Top surface of thin panel (12.7 cm/5-in panel) with span-wise through-crack in the region of the trailing edge.	35
Figure 30: Displacement time-history near the 12.7 cm (5-in) panel trailing edge for three consecutive heated flow test conditions. Notes: (a) intermittent large displacement, (b) continuous large displacement, and (c) intermittent and now damaged panel.....	35
Figure 31: A comparison of Kulite and PSP dynamic pressures with/without spatial averaging ($U_{\infty} = 490 \text{ m/s}$, $q_{\infty} = 123 \text{ kPa}$, $\delta^* = 8.6 \text{ mm}$).....	36
Figure 32: RC-19 (a) measured pressure spectra; (b) PSP measurement location and the variance of measured pressure; and a comparison (c) between the rigid control and thin-panel specimens (Points 1 & 2: $U_{\infty} = 490 \text{ m/s}$, $q_{\infty} = 123 \text{ kPa}$, $\delta^* = 8.6 \text{ mm}$, Points 3 & 4: $U_{\infty} = 287 \text{ m/s}$, $q_{\infty} = 123 \text{ kPa}$, $\delta^* = 8.6 \text{ mm}$).....	37

LIST OF TABLES

Table 1: Thin panel modal frequencies and damping	9
Table 2: RC-19 Boundary-layer properties.....	12

Abstract

There do not currently exist reasonable means to accurately simulate the complex and often nonlinear response of an aircraft structure to the dynamic, high-temperature and transient environment of high-speed flight. While methods do of course exist to predict aspects of the static and dynamic responses of such structures, once the response becomes nonlinear (geometric, material, etc.) and the transient nature of the loading requires a long time history, the simulation problem proves quickly intractable. As a consequence, predictive techniques rely on simplifying assumptions, oftentimes overlooking or losing the ability to capture extreme responses or local effects that lead to damage and failure. Structural, and even coupled, reduced order modeling methods certainly have their place [1]; however, very little relevant experimental data exists for the validation of these methods. The following experimental campaign was thus started for three principal reasons: (1) to observe and measure the effect of turbulent, SBLI interactions and heated flow on a thin, aircraft-like panel; (2) to explore severe structural events (geometric nonlinear, dynamic instabilities and material failure); and (3) to develop and/or refine full-field and non-contacting experimental measurement techniques necessary to characterize the high-speed flow environment and corresponding structural response. All of the objectives were achieved, including the failure of the panel due to the extreme, large-amplitude response experienced during heated flow testing. 3D Digital Image Correlation (DIC) techniques were refined for this particular experimental series with success in capturing full-field dynamic displacement and strain across the panel specimen. For the first time, the 3D DIC technique was also used to record the panel behavior while filming through the flow and SBLI environment. Fast reacting pressure sensitive paint (PSP) was used, concurrently with 3D DIC, to record the dynamic surface pressure on the flow-side of the panel. The PSP results were compared (spatial averaging considered) with the more traditional Kulite dynamic pressure sensors, as well as with early experimental data sets. Temperature sensitive paint (TSP) was used for the last series of experiments, as it was demonstrated [2] that a single temperature (thermocouple) data point was of course insufficient for modeling purposes. Finally, the panel response to heated flow and sensitivity to panel back-pressure modulation was studied, with transient, large-deformation limit cycle behavior measured. The experimental results are being assembled, distilled and prepared for release to the larger aerospace research and development communities.

1 INTRODUCTION & BACKGROUND

Sonic, or acoustic fatigue, has long been and continues to remain [3] an issue for civilian and military aircraft. Sonic fatigue is a resonant high-cycle fatigue issue for constrained and lightly damped skin-panels exposed to the high-amplitude (sometimes > 170 dB) broad-band dynamic loading from engine noise, turbulent boundary layer noise amplified by SBLI, and outer mold-line (OML) structures exposed to buffeting from external pods, fuel tanks and other protuberances, internal cavity (weapon bay) structures, and structures downstream of engine/ducted exhaust, e.g., the B-2, UCAV, etc. While it has largely remained a maintenance and cost issue for more traditional metallic civilian and military aircraft, the acoustic fatigue phenomenon does limit the effectiveness of military aircraft by removing them from potential missions. Further, acoustic fatigue could prove catastrophic for the case of high-speed vehicles, i.e., OML skin/leading-edge structural failure/fatigue allows hot-gas intrusion. Figure 1 displays (a) the typical nonlinear response of constrained aircraft OML panels, and (b,c) the types of structural failure that can occur when panels are exposed to this broadband dynamic loading. Solutions for this problem often start with the thought to increase the bending stiffness of the fatigue prone area through either modification to the panel skin thickness, the stiffener orientation, placement density or design. This can often force the high-cycle fatigue issue to another trouble-prone area. The best solution is one that damps, or removes, the offending dynamic loading. For example, Liguore and Beier [4] present some of the characteristics of sonic fatigue and then detail the results of a flight test of a pre-emptive damped, composite repair patch in a fatigue prone region of the USAF F-15 aircraft.

As the flow speed moves from sub- to supersonic, the fluid and structural disciplines are oftentimes coupled. Frendi notes that acoustic damping, which increases with vehicle speed, will lead naturally to a reduction in the dynamic response of aircraft panels, particularly panel response at higher frequencies [5].

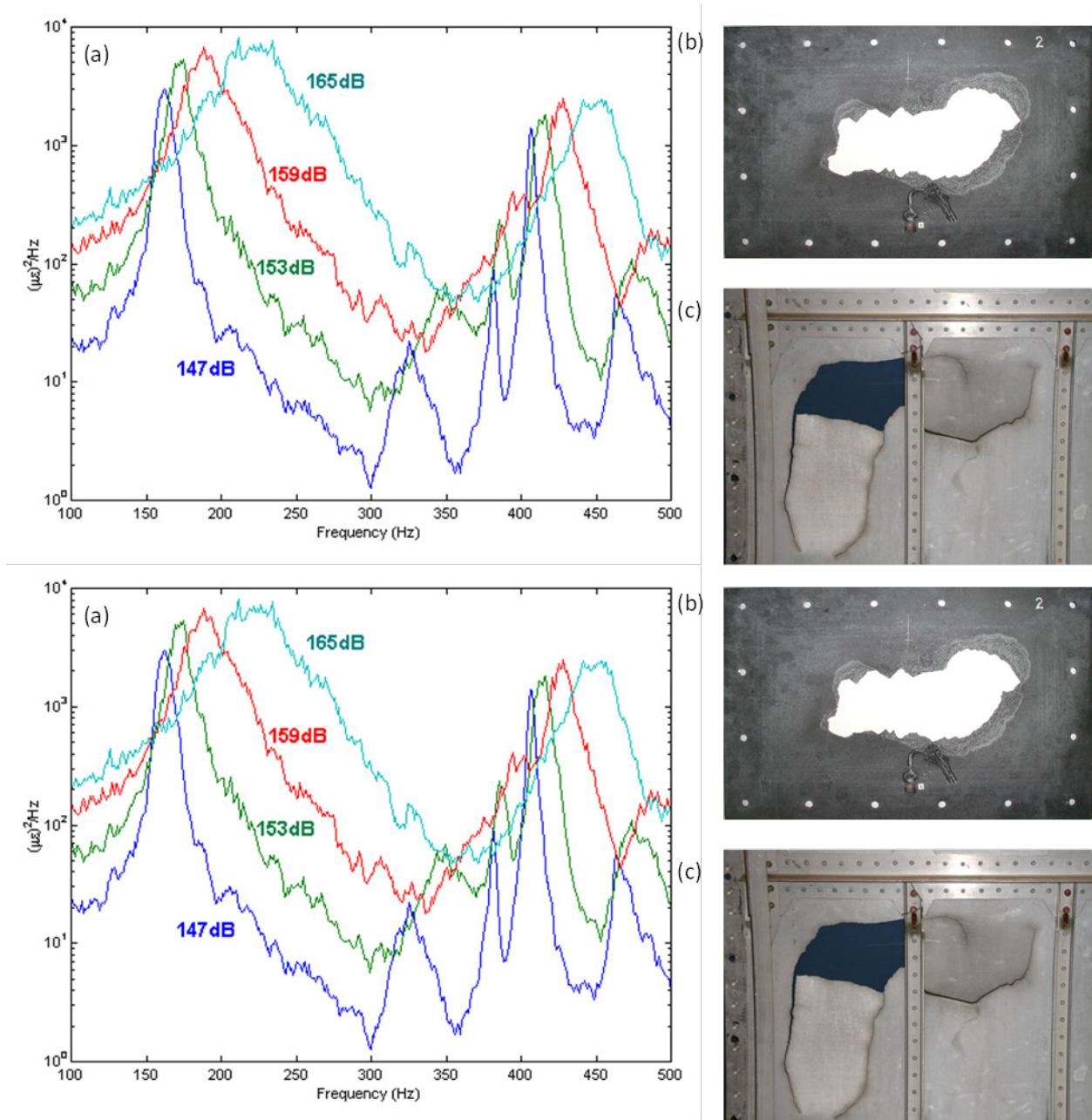


Figure 1: (a) Geometric (hard-spring) nonlinear response typical of acoustic fatigue [6]; (b) Ceramic matrix composite panel failure (172 dB) [7], and (c) C-130 aluminum aircraft panel failure (160 dB) due only to acoustic/dynamic resonant fatigue.

For structures that are exposed to the random dynamic loading of engine noise or turbulent SBLI, combined with high-temperatures, the problem grows in complexity. The addition of thermal effects reduces the strength and stiffness of the aircraft structure, and introduces time-dependent material behavior [8]. Because aircraft panels are constrained and not free to grow unimpeded, initially mono-stable structures can realize a state where multiple stable equilibria exist. As a consequence, the addition of the boundary layer and/or engine-induced dynamic loading can drive the response between these equilibria and thus significantly shorten the life of the panel. A

nice example of large panel deformation is seen in Figure 2, where the Titanium skin panels are deformed/post-buckled over much of the underside of the aircraft. Now this is the case of an aircraft on the ground (indoors no less) and therefore the post-buckled deformation is likely from the static weight of the aircraft. It is quite possible that once in flight the panels would return to a "smoother" state. However, the figure is instructive by noting just how much deformation these constrained panels could realize when exposed to the extreme aeroheating at Mach 3.5+.

There have been very few flight test cases where boundary layer noise has been measured for the purposes of studying and quantifying the effect on aerospace structure. There are even fewer cases involving very high-speed aircraft. In the mid-1960's, NASA researchers instrumented OML panels on the X-15 hypersonic research vehicle, representing one of the few instances where structural weight is tied directly to the magnitude of dynamic loading in the boundary layer - a 10-dB increase in overall sound pressure levels would result in a 75% increase in structural weight using their design approach [9].



Figure 2: Panel deformation and buckling on the underside of the YF-12 experimental aircraft at the USAF Museum.

A good example of the effects of extreme heating on constrained aircraft structure is the Mach 6.7 flight of the X-15 [10]. In this case, vehicle X-15-2 was modified to carry a "dummy" ramjet. The ramjet pylon suffered a failure due to shock-shock interference heating. The fuselage structure at the root of the pylon experienced buckling and permanent deformation - both geometric and material nonlinearity. The best observations and calculations, calibrated with wind-tunnel experiments, predict that the heating rate in the vicinity of the pylon/fuselage-structure increased by a factor of 7 due to the impingement/interference heating thus leading to the unexpected structural response. Thornton and Oden studied the thermo-viscoplastic response of a notional hypersonic structure, and note that the predicted large inelastic deformation would augment locally the aeroheating [11].

It is also well known that SBLI is a dynamic event, and that the motion of the shock occurs at a much lower frequency (several orders of magnitude lower) than the characteristic frequency of the boundary layer [12, 13]. The low frequency shock motion could also drive an aircraft panel at resonance, leading to the aforementioned fatigue [14]. In their review paper, Clemens and

Narayanaswamy highlight the results of an earlier Mach 5, 280 compression-ramp experiment where it was shown that the separation shock foot exhibits low-frequency motion near 500-Hz. This low frequency unsteadiness is ubiquitous, having been observed in other interactions such as reflected shocks (like the present experiment) and blunt fins (like the X-15-2 example discussed previously).

In the 1980's and 1990's the USAF spent considerable time and effort on a single-stage-to-orbit (SSTO) vehicle. Blevins et al. note the interconnectedness of the various disciplines involved in the design and analysis of such a vehicle [15]. Blevins' team at Rohr Industries was contracted by the U.S. Air Force Research Lab to design and fabricate representative OML panels to study the response to combined, extreme and representative acoustic and thermal loading. The SBLI environment can amplify the aeroheating and aeroacoustic pressures by an order of magnitude [16]. The authors used a generic transatmospheric vehicle shape as the basis for identifying the loads and deriving the resulting material and structural requirements. The aft wing region of the vehicle is prone to SBLI for speeds of Mach 10 and greater. In the case of shock-impingement, the fluctuating pressure overall sound pressure level (OASPL) is predicted to be 166 dB (Ref. 20 μ Pa) or greater depending on the ascent trajectory dynamic pressure. Engine noise on the compression ramp panel region leading to the scramjet engines was estimated to be 160-170 dB range, necessitating a redesign of the panels in that region (increase in panel thickness or a decrease in stiffener spacing). The compression ramp panel region, up-stream of the scramjet engines, was not designed to carry pass-through loads, e.g., primary aircraft mechanical loads. In fact, the authors note that experience dictates that structures designed for "primary load and stiffness" [15] will not be susceptible to dynamic, fluctuating pressures in the boundary layer. However, those structures that, like the compression ramp region, serve as "fairings" will be sized by the dynamic loading. This last comment must be taken with some caution for two principle reasons: (1) hypersonic vehicles are inherently multi-disciplinary often with unexpected responses, and (2) there isn't sufficient experience designing and flying in-atmosphere hypersonic vehicles to warrant sweeping statements regarding their design. The results of the present study indicate some surprising responses for the various conditions considered.

Constrained hot structure will also deform into the flow, and could lead to a scenario where the aerothermal heating is amplified locally leading to additional deformation. Kontinos and Palmer analyzed insulated metallic TPS panels for the X-33 NASA flight test vehicle using a very loosely coupled simulation approach [17]. The X-33 experimental vehicle program was designed to integrate and reduce the risk of key technologies to be used on a Space Shuttle Orbiter replacement, the next generation reusable launch vehicle (RLV). One of the key technologies to be studied on the X-33 was the use of a metallic TPS design. This innovative design, desired due to its enhanced durability, was expected to be an evolution of the ceramic tile TPS design used on the Space Shuttle. In the case of the X-33, it was anticipated that entire regions of the vehicle would experience significant panel deformation into or away from the vehicle body. This could lead to augmented heating, in-turn affecting panel deformation and so Kontinos and Palmer undertook a coupled fluid-thermal-structural approach to quantify this effect. Glass and Hunt executed an experimental campaign studying the same scenario in the NASA Langley High Temperature Tunnel (HTT) using various configurations of rigid dome protuberances [18,19]. Kontinos and Palmer showed that there was not a significant difference in the uncoupled and loosely coupled quasi-static analysis because the deformation did not significantly alter the local aerothermal heating or the integrated heat load. There were however

noted differences in the panel surface temperature leading to higher in-plane stresses. While their analysis did not show a pressing need for a coupled analysis approach for this application/assumed loading conditions, this would likely not remain the case if the in-plane resistance (i.e., boundary conditions) were to increase unexpectedly as will next be discussed.

Culler and McNamara [20] studied a problem similar to the X-33 one just described. They quantified the effect of one- versus two-way coupling combined with variations in the in-plane boundary resistance (stiffness) of the SSTO compression ramp panel designed as part of Blevins' study [15]. The results are striking. It was shown for a simulated Mach 12 trajectory that a 2-way, reduced-order/analytical fluid-thermal-structural coupled model, combined with sufficient in-plane boundary resistance, uncovered a failure mode not observed in a one-way, or prescribed loading approach. The maximum panel deformation was on the order of 3-thicknesses and so well into the geometric nonlinear regime. Compare this to the X-33 panel analysis where the peak deformation was less than a single thickness. This is not unexpected given the different intent and philosophy of the X-33 TPS design – the panels were not directly connected but rather on flexible isolating pads attached to the substructure. This finding demonstrates: (1) the extreme sensitivity of the response of hot-structures to boundary conditions; and (2) the seeming correlation between large deformation nonlinear response and the requirement for coupled aerothermoelastic analysis.

For these reasons, an experimental campaign was started to explore the response of an aircraft-like structure to supersonic turbulent boundary-layer dynamics, SBLI and heated flow conditions. While similar experiments have previously been conducted, i.e., Coe & Chyu [16], Maestrello [21], DLR [22,23,24], none offer the combination of extreme loading conditions and/or the use of full-field measurement techniques to record the response.

2 EXPERIMENTAL OVERVIEW

All of the experiments were carried out in the Air Force Research Lab (AFRL), Research Cell 19 (RC-19) supersonic tunnel [25]. The RC-19 research facility was designed and built in the mid-1990's to study scramjet-inspired fuel-air mixing and combustion. It is a variable Mach (1.5-3) continuous flow facility with superb optical access. The RC-19 test section consists of four separate interchangeable walls that can be configured to meet a variety of experimental requirements. The rectangular tunnel test section is 127x152-mm (5x6-in) and was modified for the present series of experiments beginning in 2008 to accommodate a flexible panel flush with the top-wall. Details of the modified test section, including the shock generator, test panel and cavity locations, are shown in Figure 3 (an early bonded and not machined test panel configuration as used later in the testing campaign). For the present series of experiments, the single top-wall was replaced with three distinct sections. The upstream section contained pressure ports before the flexible panel specimen. The middle section contained the flexible panel and a cavity behind the panel (see Figure 3). The cavity was “ported” to the tunnel wall downstream of the compliant panel, connecting the tunnel to the panel cavity so as to equalize the pressure on both sides of the panel and avoid yielding the specimen during tunnel start-up. The cavity section had a 19-mm thick clear quartz window through which the backside panel dynamic response was measured using 3D Digital Image Correlation (DIC). The dimensions of the cavity, 254x127x51 mm (10x5x2 in), did result in a minor stiffening effect, i.e., panel-acoustic coupling estimated to be 6 Hz in the first mode based on modeling of the combined panel and cavity system. The first predicted cavity acoustic mode, in the panel normal direction, is 3 kHz. The stiffening effect was most pronounced in the first mode as noted by Dowell and Voss [26] and Pretlove [27].

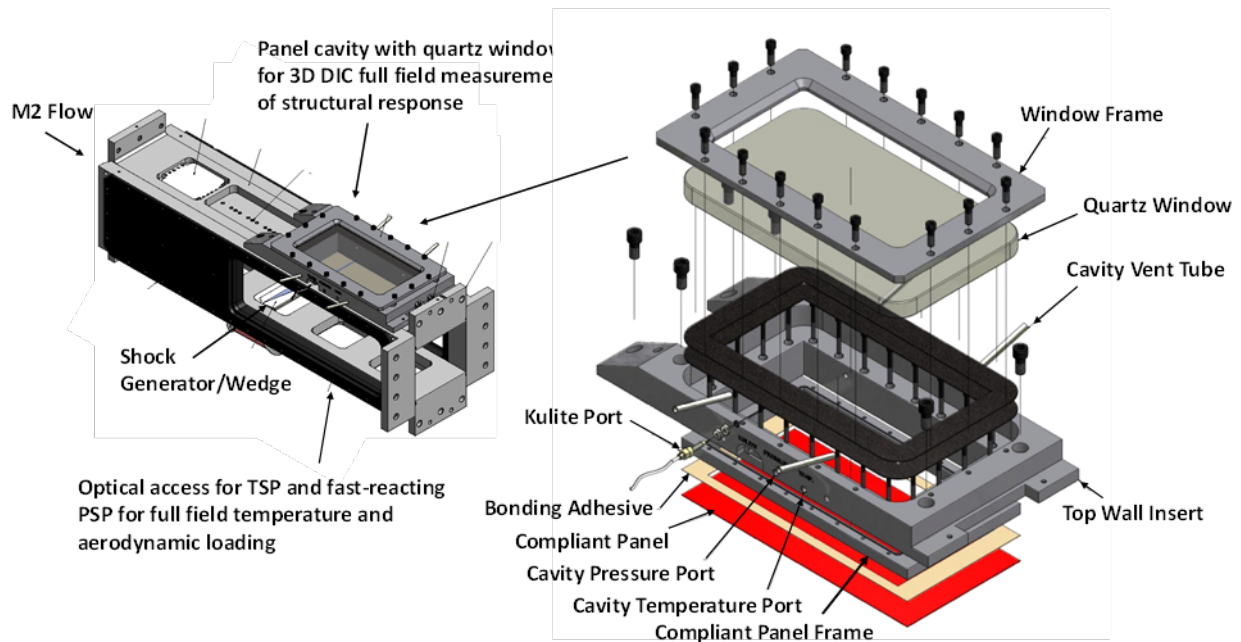


Figure 3: RC-19 modified test section, denoting flexible panel, shock generator, optical access and details of early “bonded” thin-panel arrangement.

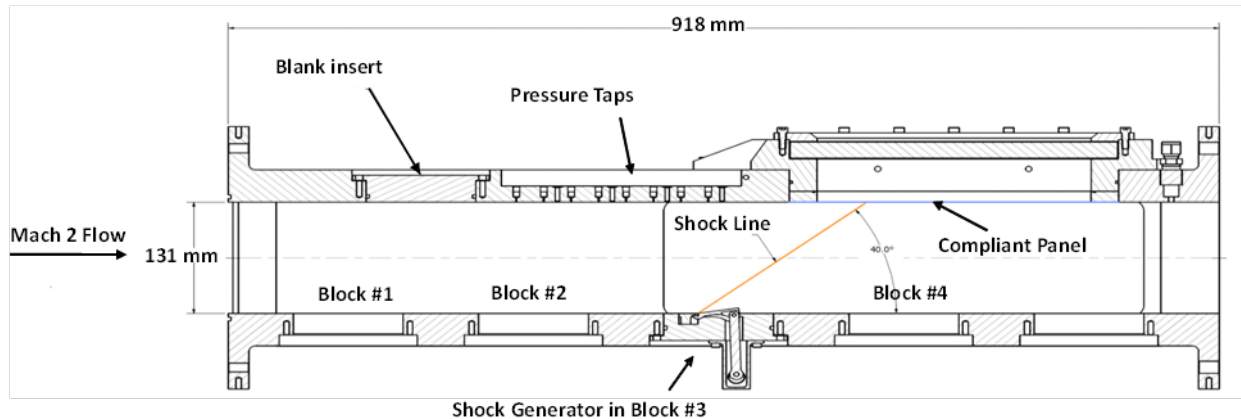


Figure 4: RC-19 modified test-section showing pressure taps, shock generator locations and thin panel test specimen.

Early and preliminary tests identified facility mechanical noise measured during tunnel operation and so additional bracing was added to each side of the tunnel wall. The end result of the tunnel modifications was a 40% reduction in measured acceleration at the location of the panel specimen - from 0.89 to 0.53 g's RMS (0-2 kHz) – see Figure 5.

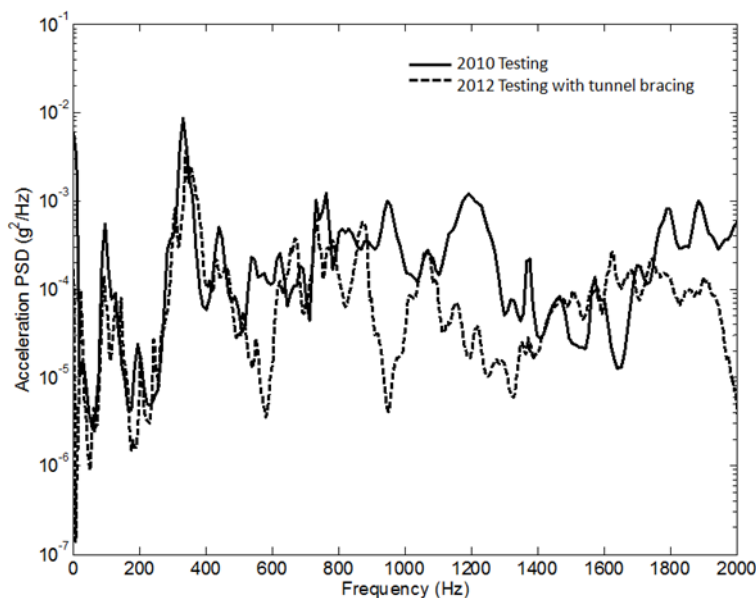


Figure 5: Acceleration PSD showing effect of tunnel bracing, ($P_o = 270$ kPa).

All testing was conducted at Mach 2.0 (nominal), free-stream dynamic pressures of 61.7, 91.4, and 123 kPa, and total pressures of 172, 270, and 345 kPa. While Mach 3 was within the RC-19 testing capability, the Mach 2 tunnel configuration allowed for the design of a larger thin-panel specimen. During the final year of testing in 2016/2017, only a free-stream total pressure of 345 kPa was considered. The Mach 2 nozzle is located upstream of the test section shown in Figure 3. The average, static pressure measurements, taken from stream-wise locations along the nozzle wall ($x < 0$) and in the portion of the test section upstream of the test panel ($x > 0$), are shown in

Figure 6. The Mach number in the test section was calculated to be 1.92 using isentropic relations at the most downstream static pressure measurement (50.2 kPa at location $x = 0.45$ m).

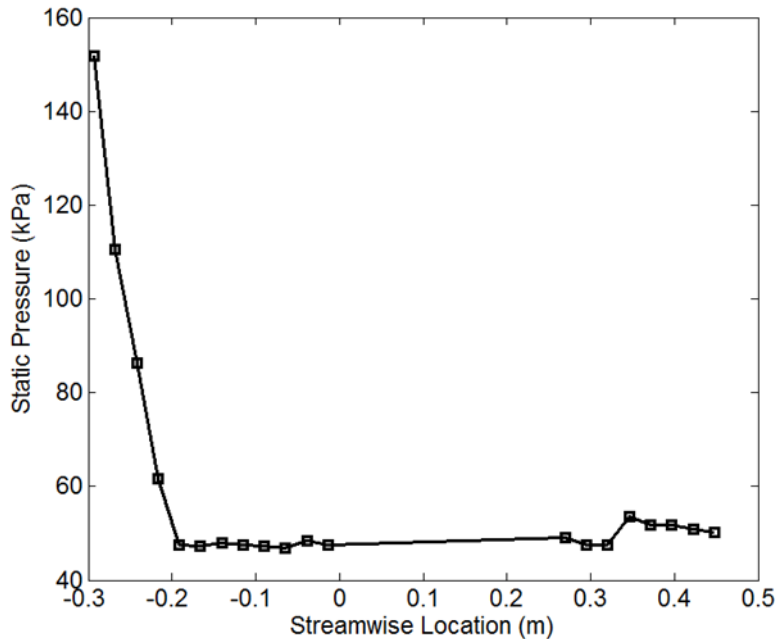


Figure 6: Static pressure measurements along the tunnel-wall.

The sides of the tunnel contained large windows to allow for pressure/temperature sensitive paint (PSP/TSP) illumination, as well as the use of a high-speed shadowgraph system to visualize the SBLI flow characteristics. A window was also added to the bottom wall test section so as to view the PSP on the flow-side of the compliant panel. When the shock generator was used, it was placed in the bottom wall of the tunnel opposing/upstream of the thin-panel. During the first year of testing, the shock generator was kept in a fixed position and the angle between the tunnel bottom wall and shock generator “ramp” was adjusted in-situ from a flush-wall condition to a maximum angle of 10° thus impinging at different stream-wise panel locations. In the second two tunnel entries, a fixed 8° shock generator was used to turn the flow resulting in a shock wave angle of approximately 39° as measured from the tunnel bottom wall. In that case, the shock generator was moved/translated 170 mm in the flow direction to allow for the shock wave to impinge from the compliant panel leading edge to nearly the panel mid-point in the stream-wise direction.

Three different thin-panel specimens were tested during the experimental campaign. In the first year of testing, a 0.635 mm panel was bonded to rigid frame for an effective dimension of 254x127 mm (10x5-in). In the second and third years of testing, a 305x152x12.7 mm (12x6x0.5-in) block of AISI 4140 alloy steel was machined to create the compliant panel. A pocket was machined into the block leaving 0.635 mm (0.025 in) for the compliant panel thickness over an effective vibration dimension of 254x127 mm (10x5 in) and 254x102 mm (10x4 in). A rigid (non-responsive) panel, i.e., without the thin machined region, was also manufactured to serve as the control specimen. This panel included both pressure ports and the ability to mount a boundary layer rake at three locations so as to characterize the tunnel boundary

layer in the location of the thin panel. The thin, 127 mm wide flexible panel specimens were sized so that three modes of vibration would be below 500 Hz, thus the earlier reference to “aircraft-like” panels, i.e., acoustic-fatigue like resonant response.

Modal testing was performed on each panel used in the wind tunnel experiments. An example of the modal test arrangement is shown in Figure 7. Note that the panel is resting on elastic cords (free-free condition), and is being excited by a small magnetic driver on the underside of the panel. The magnetic driver is located off-center so as to excite many low-frequency symmetric and asymmetric modes. The velocity response of the panel was recorded using a Polytec OFV-552 fiber-optic, dual-beam (differential measurements) laser Doppler vibrometer (LDV) for 45 measurement locations denoted by the reflective tape pieces on the top surface of the panel. The pre- and post-test measured frequencies and damping for the machined (not bonded) 127 mm wide, thin-panel specimen used in the second year of testing are shown in Table 1. There were negligible differences in the pre- and post-test frequency measurements while the measured viscous damping ratio did increase a small amount. This slight pre- and post-test increase in measured damping was typical of all of the machined specimens used throughout the multiple wind-tunnel entries.

Table 1: Thin panel modal frequencies and damping.

Mode	Frequency (Hz)		Damping Ratio (%)	
	Pre-test	Post-test	Pre-test	Post-test
(1,1)	208	209	0.16	0.22
(2,1)	225	225	0.14	0.21
(3,1)	334	335	0.07	0.13
(4,1)	499	499	0.06	0.11
(1,2)	565	563	0.07	0.11
(2,2)	612	608	0.06	0.09

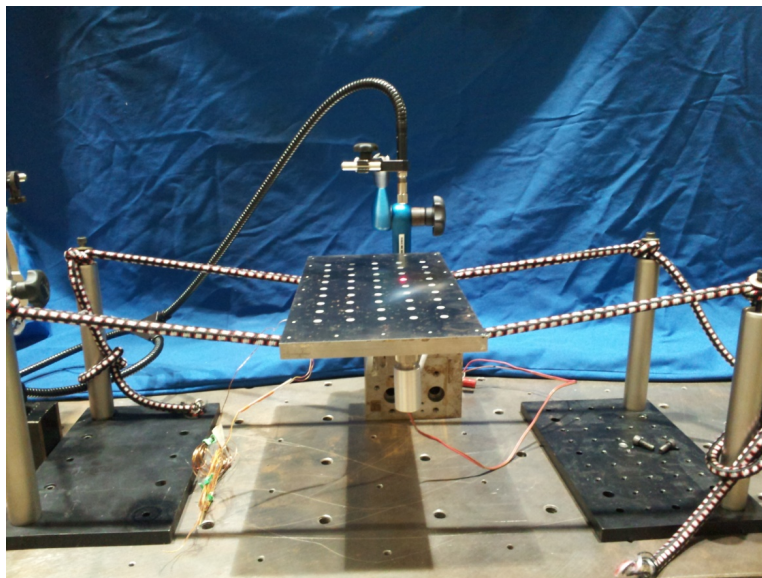


Figure 7: Modal test arrangement for the RC-19 thin panel specimen.

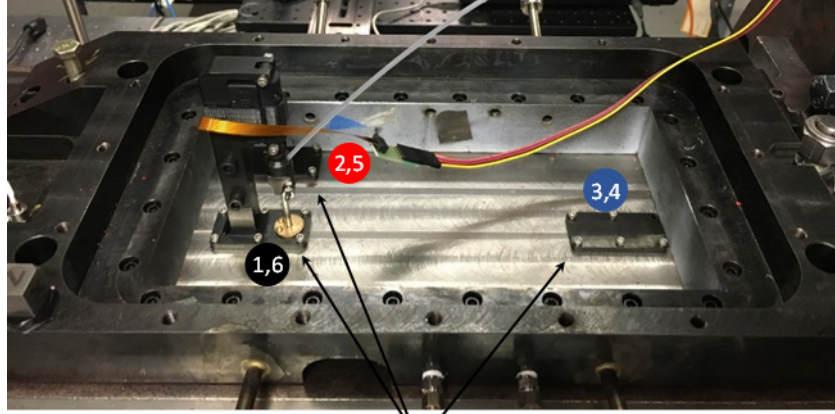
As noted earlier, a rigid control specimen was used for characterizing the boundary layer in the region of the thin-panel specimen. The purpose was to check the state of the boundary layer against earlier measurements [25] now that the RC-19 tunnel test-section was modified for the present series of thin-panel experiments. The ratio of total pressure between the test-section core flow and the near wall were used to compute the boundary-layer profile. The total pressure measurements were made using a traversing pitot tube as shown in Figure 8. Note also the three locations along the rigid control specimen where total pressure measurements were taken - two upstream locations (denoted by the black and red markers) and one downstream (blue marker). Because a bow-shock forms in front of the Pitot tube, adjustments must be made prior to comparing the measured total pressures with the free stream measurements. This can be done by using the measured post-shock total pressure and the upstream static pressure to calculate the Mach number using the Pitot-Rayleigh formula:

$$P_{0_2} = P_1 \left[\frac{(\gamma + 1)M^2}{4\gamma M^2 - 2(\gamma - 1)} \right]^{\frac{\gamma}{\gamma - 1}} \left(\frac{1 - \gamma + 2\gamma M^2}{\gamma + 1} \right) \quad (1)$$

where P_{0_2} is the pitot tube measurement and P_1 is the upstream static pressure. The Mach number is first calculated and then the upstream total pressure (P_{0_1}) can be solved as follows:

$$P_{0_1} = P_1 \left(1 + \frac{\gamma - 1}{2} M^2 \right)^{\frac{\gamma}{\gamma - 1}} \quad (2)$$

The profile colors in Figure 9 correspond to the colors and locations in Figure 8. The boundary layer profile for Run #4 (Figure 9) is significantly different than the other profiles and this is due to the presence of a wedge-generated shock impingement on the rigid panel approximately 1/4 of the panel length downstream of the leading edge ($x/L = 0.25$).



BL rake locations across rigid control specimen

Figure 8: Boundary layer measurement locations upstream (1 & 2) and downstream (3) on the rigid control specimen.

Note, the colors are coordinated with the boundary layer measurement results of Figure 9.

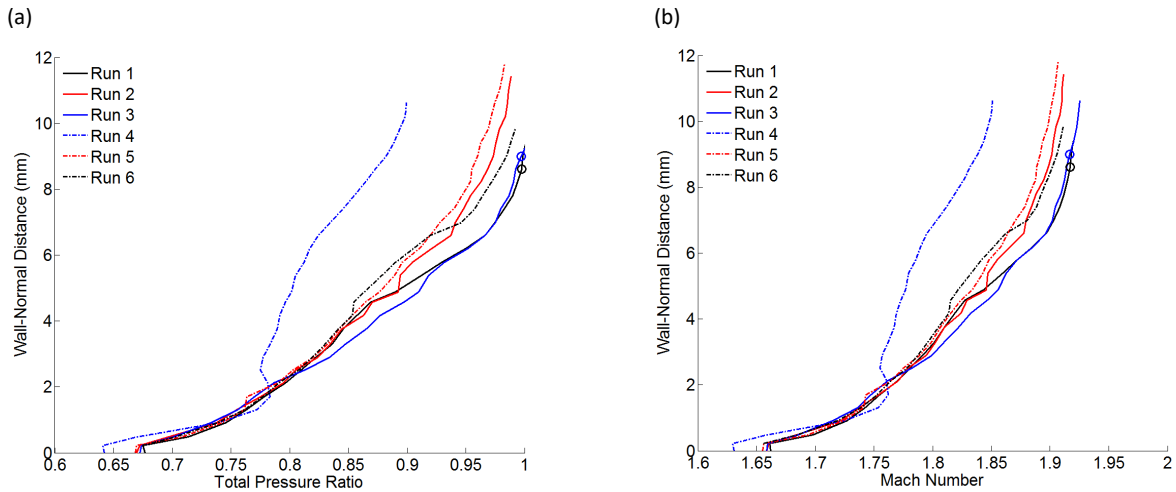


Figure 9: Boundary layer measurement results for 6 tests and three locations denoted in Figure 8.

The boundary-layer thickness was approximated as that location where the total pressure (P_{0l}) first exceeds 99.5% of the freestream value (mean value of 346 kPa). These locations are denoted by the circle markers in Figure 9. The data runs for case numbers 2, 4, 5 and 6 did not satisfy this edge criterion and so for these runs the boundary-layer thickness was approximated using following analytical expression for the experimental velocity ratio:

$$\frac{u}{U} = \left(\frac{y}{\delta}\right)^A$$

(3)

where δ is the boundary-layer thickness, u the velocity at the probe elevation, U the free-stream velocity, and y the probe height. A least-squares approximation, given the measured velocity profile and probe height, determines the boundary layer thickness and exponent, A . The results of the boundary-layer thickness estimates are displayed in Table 2.

Table 2: RC-19 Boundary-layer properties.

Run Number	x (m)	δ (mm)
*1	1.19	8.6
2	1.19	11.1
*3	1.38	9.0
4	1.38	39.0
5	1.19	12.4
6	1.19	11.7

Run 1 and 3 (highlighted in Table 2) compare well with the Gruber's earlier RC-19 tunnel characterization [25].

The previous discussion alluded to some of the measurement techniques used during this multi-year experimental effort – 3D DIC, PSP and TSP along with single-point strain, temperature thermal couples, and panel velocity via the Polytec dual beam LDV. A major part of this effort was to apply, and extend where necessary, the 3D DIC, Fast Reacting PSP, and TSP full-field measurement techniques. These full-field techniques were desired so that a detailed and very-well characterized data set could be provided to the larger aerospace research community. As one example of improving the measurement techniques, an attempt was made to combine the PSP and TSP to capture simultaneous temperature and fast-response pressure measurements, but the technique (currently) proved unsuccessful. The effect of combining the paint resulted in reduced temperature sensitivity for what were already small changes in panel temperature for the test conditions considered. This issue is still being investigated.

The high-speed 3D DIC set-up also evolved throughout the experimental effort. The first two tunnel entries used dual Photron SA-5 cameras, each with 32 GB, to record the panel dynamic response. The cameras were later upgraded to Photron SA-Z cameras, each with 64 GB, for the final year of testing. In that last year of testing, the original SA-5 cameras were used to film the panel dynamic displacement through the supersonic flow. This is a first step for the community in solving a real problem with this type of experimentation – that is to film real-time structural dynamic response when the structural specimen is in the flow and not a part of a wind-tunnel side-wall. Details will follow.

In addition to DIC camera hardware improvements, the optimal speckle pattern, and application of that pattern, was improved as well. The top (cavity-side, see Figure 3) of each panel was prepared for DIC by first uniformly painting the specimen using a standard flat-white paint followed by a hand-applied random pattern of flat-black dots using a fine-tipped permanent marker. The randomness of the 3D DIC pattern is crucial to avoid spatial aliasing of the digital images which can then lead to displacement measurement bias. In the first year, the DIC pattern was applied to only 21 discrete regions evenly spaced over the panel surface plus one additional

speckle region applied over the foil strain gage for direct strain comparisons – see Figure 10. A reference point (see Figure 11a) was also added to the panel frame/tunnel to remove tunnel motion from the measured panel displacements.

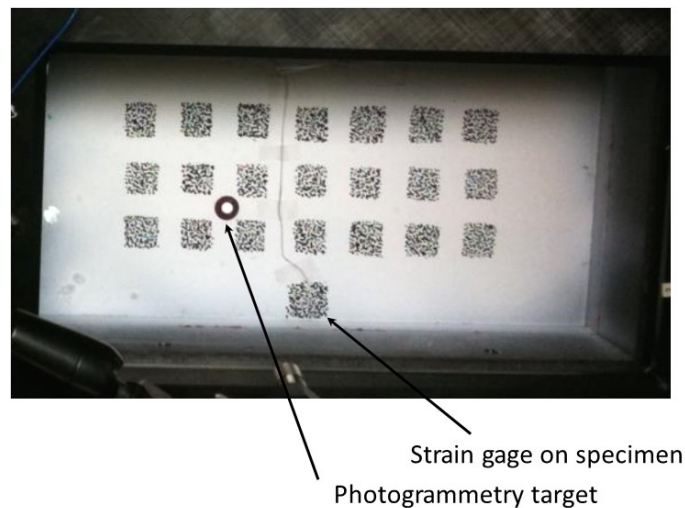


Figure 10: Year 1 3D DIC speckle pattern applied at discrete points across the bonded panel.

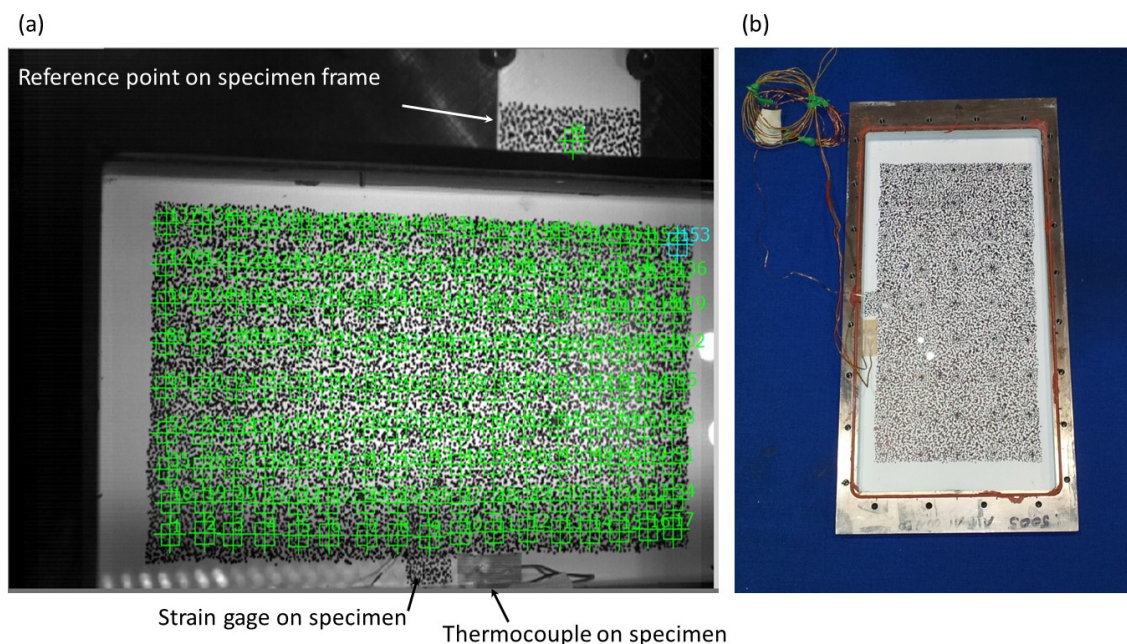


Figure 11: Years 2 & 3 3D DIC pattern applied across the machined panel surface and the tunnel/specimen frame.

Note there are 153 “facets” identified in (a) across the random pattern, while (b) shows the machined panel prior to test and prepared for measurement using DIC.

For the second and third years of wind tunnel testing, the DIC speckle pattern was applied to the majority of the machined panel surface as shown in Figure 11, thus allowing for displacement

measurements across the entire surface. A speckle pattern was not applied right to the edges of the panel since the transverse displacements near the clamped boundaries were quite small and typically lost in the measurement noise. Testing in the final two years used a 153-point 20x20-averaged pixel facet array (see the green squares in Figure 11a) so as to survey the response across the panel and provide sufficient measurement locations for operational modal analysis (OMA) based panel deformation identification. For reference a DIC facet is an area of selected pixels over which the panel response is averaged in turn reducing system noise. For all 3D DIC data acquisition the following analysis software, calibration, cameras, and settings were used. The camera calibration object used was a GOM specific 250x200 mm panel with a 150 mm calibration scale. The lenses used were 28 mm set at an aperture of 2.8. The resulting camera calibration produced a measurement volume of 425x450x450 mm. The calibration deviation was 0.021 pixels and the reported camera angle was 25.8 degrees. The camera resolution for specimen measurement was 768x496 pixels. Specimen illumination consisted of a 305x305 mm Litepanel which is a panel array of 1152 LED bulbs for a consistent distribution of lighting that also minimizes specimen heating and therefore pre-stress. Two high intensity area lights (model LEDRA70D4-XQ) were also used to help alleviate shadows caused by the cavity walls. The DIC software used in this study was the GOM Optical Measuring Technique's IVIEW Real-Time Sensor for dynamic analysis and ARAMIS for static analysis. Through a collaborative effort between the AFRL and GOM the IVIEW program was modified to accommodate the large volumes of images required for random-vibration measurement. The 3D-DIC sampling was increased from 500 Hz, to match early PSP sampling, to 5000 Hz by the last tunnel entry in 2017.

Temperature sensitive paint and fast-reacting PSP were used to obtain full-field unsteady panel surface temperature and pressure measurements for these RC-19 experiments. Conventional pressure sensors/transducers are placed at pre-determined discrete locations. PSP and TSP approaches are relatively new and still being developed/extended. Their benefit is that they provide a non-intrusive means to simultaneously capture the temperature and pressure across the structure. PSP is made of luminescent molecules suspended in an oxygen permeable paint/binder. When the paint is illuminated the brightness of the resulting luminescence is recorded with a camera. The intensity of the luminescence is inversely proportional to the partial pressure of oxygen because the oxygen that diffuses into the paint layer absorbs the excess energy of the luminescent molecules in a process called quenching. Through pre-test calibration imaging, that intensity is related to pressure thus providing the non-contacting and full-field surface pressure. The TSP operates in a similar fashion to the PSP. One of the goals of the RC-19 experiments was to estimate the error in the PSP measurements. This was done by comparing the PSP pressure measurements with data from the discrete Kulite pressure transducers at different locations on the wind tunnel wall as well as across the rigid, non-responsive control specimen.

In addition to the dynamic full-field panel pressure and displacement measurements, a myriad of other measurement techniques were used to better characterize the tunnel environment and panel response - LDV, High-Speed Schlieren, Kulites, accelerometers mounted on the tunnel, tunnel test-section and camera mounts, a single foil-type strain gage placed on each test panel, and thermocouples on the panel and the panel frame to quantify temperature differences. The dual-beam Polytec LDV was used as a single point reference to compare with the 3D DIC measurements. The laser velocity measurement location can be seen on the early test panel in

Figure 10, denoted as “photogrammetry target.” Small pieces of reflective tape were applied in a similar location for the panels shown in Figure 11.

3 TEST PROCEDURE

The testing procedure throughout all three tunnel entries largely proceeded in the following manner. The bottom or flow-side of the rigid (control) and flexible panel specimens were painted with the fast-response PSP or the TSP paint from Innovative Scientific Solutions Inc. on the day of testing. The thin-panel specimen was removed (or not installed yet) from the tunnel and painted in an appropriate painting facility due to the confined area within the tunnel. After drying, the panel specimen was reinstalled and a structural modal test was performed on the panel to ensure proper installation. Illumination for the PSP/TSP was provided by two LM2X-400 LED heads entering through the side windows and triggered five seconds prior to image acquisition and turned off after the acquisition of all images was completed. The return lighting from the panel exited through a window in the bottom of the tunnel and was reflected by a mirror to the PSP/TSP camera. The camera, also a Photron SA5 with 32 GB of memory, was used to record the full-field PSP/TSP pressure/temperature measurements triggered simultaneously with the DIC measurements. The fast reacting PSP was recorded for 14.5 seconds and sampled at 4 kHz (for all but the first year of testing). The PSP/TSP field of view was 195 mm in the flow direction and 155 mm in the transverse direction with the each pixel representing a physical sensor size of 0.278 mm by 0.278 mm. The images were cropped in the transverse direction to match the physical plate width. The experimental procedure began by pulling a vacuum on the tunnel to an approximate absolute pressure of 20.5 kPa. A series of “wind-off” images for the PSP/TSP were collected at that time. The tunnel was then started by setting the total pressure at the nozzle to the desired value. Before starting the tunnel, the test section was at room temperature and so the compliant panel frame required some time for the thin-panel and panel-frame to thermally equilibrate. The temperature difference between the panel and frame induced a transient thermal stress that decayed over time altering the dynamic response of the panel. This temperature difference was monitored using the two previously mentioned panel and frame thermocouples. The LDV was used to record the dynamic response of the compliant panel while awaiting this thermal equilibrium. Once the spectral response of the panel showed little difference, i.e., had “reasonably” converged, the actual testing could begin. At this point, the cameras used for the 3D DIC and PSP/TSP (wind-on images) were triggered at the same time and images recorded for approximately 22.5 seconds at a 4-to-5 kHz rate for the DIC and 14.5 seconds at 4-to-5 kHz for the PSP images. The TSP images were sampled at 60 Hz. One consequence of these long 3D DIC and PSP time records (required for spectral analysis and averaging purposes, and to record the transient start-up during heated flow testing) was the tremendous amount of data (camera images) generated for each RC-19 wind tunnel test run. Simultaneous LDV velocities and strain gage data were recorded on a separate data acquisition system for 60 seconds at a sampling rate of 20 kHz. Additionally, two more data acquisition systems were employed. One was used to gather static data while the other was dedicated to the fluctuating pressure measurements. Once the test was complete, i.e., the 3D DIC, PSP/TSP and or shadowgraph camera memory was full, the images were then downloaded from the respective cameras using separate computers to decrease the download time. This process was then repeated for the next desired testing condition. For the heated flow test runs, the RC-19 tunnel gas-fired heat exchanger was employed. While the facility is capable of providing temperatures up to 650°C [25], the present heated flow experiments used the minimum facility settings (approximately 110°C). A constrained (clamped) panel with an aspect ratio of 2:1 has a critical buckling temperature of only 5°C. The purpose of the heated-flow portion of the thin-panel

experiment was to explore the transient response while the panel thermally equilibrated, and then to study/capture the dynamic response for "mildly" post-buckled behavior. If heated too excessively the panel would be well into the post-buckled regime and there would be little chance of seeing the more interesting dynamic behavior.

4 FIRST RC-19 TUNNEL ENTRY (2010)

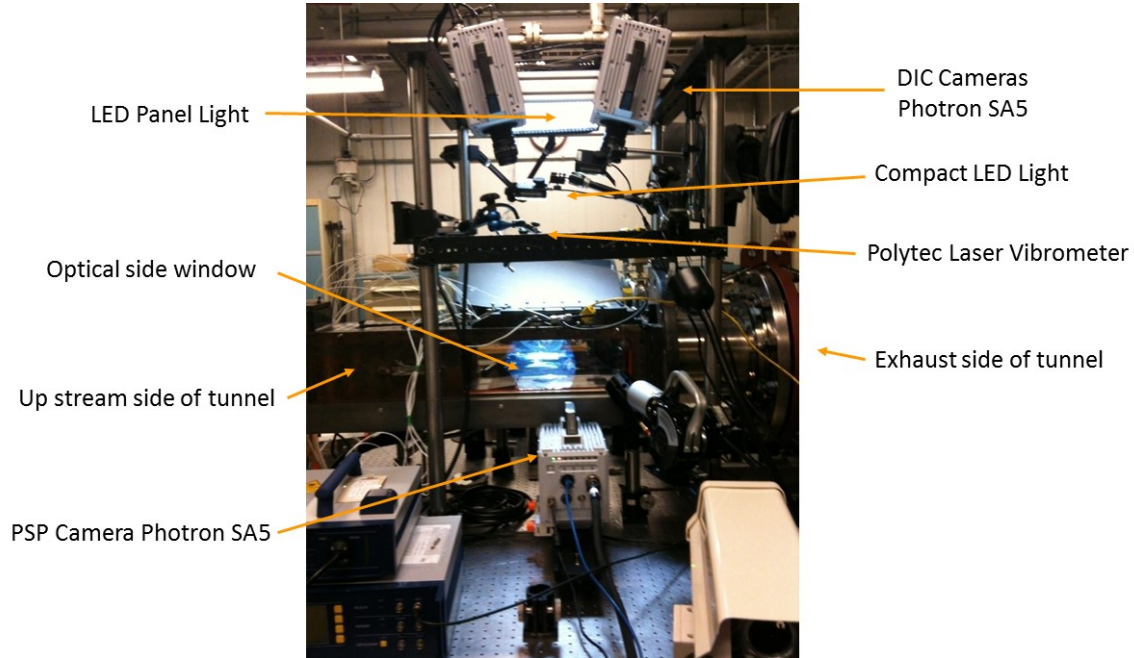


Figure 12: RC-19 tunnel test arrangement for the first year of experiments.

The crowded and cramped experimental configuration for the first (2010) tunnel entry is detailed in Figure 12. The first series of experiments included three Photron SA5 high speed cameras - two for 3D DIC and one for the fast reacting PSP measurements. A combination of miniature and panel LED lights were manipulated to provide the appropriate lighting for the DIC displacement filming. In this first series, an adjustable wedge – from flush to the tunnel surface to a maximum of 10° rotation provided a "no-shock" to a shock impingement beyond the panel 1/2-point or $x/L = 0.5$ (Figure 13).

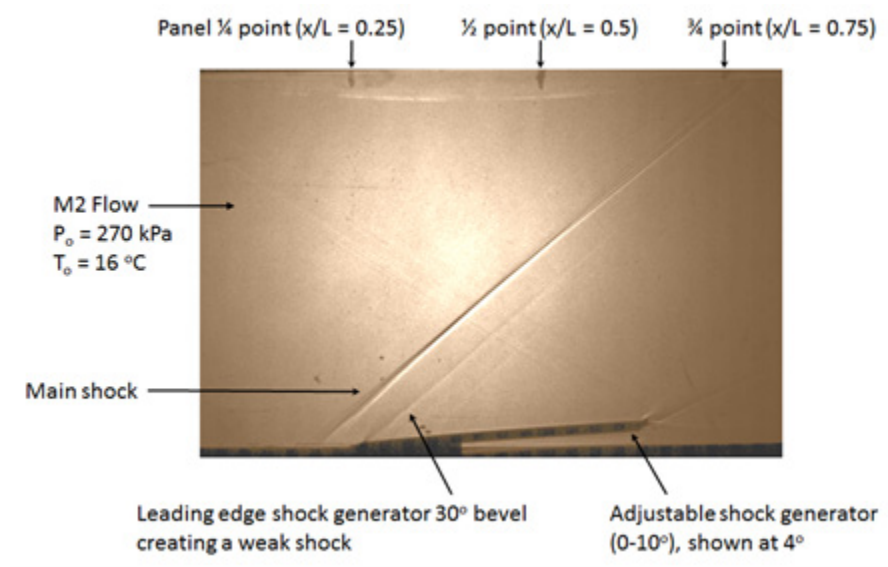


Figure 13: Shadowgraph image of adjustable shock generator in RC-19 Tunnel "Block 4."

The response of the panel to the supersonic flow and SBLI is shown in Figure 14. Note the same hard-spring nonlinear response characteristics as displayed in Figure 1a, albeit much less pronounced. The corresponding averaged (steady) PSP image is also shown in Figure 14, and the shock impingement location is clearly denoted. The 3D DIC sampling rate was limited this first year of testing to 2 kHz, and the PSP was limited (due to facility illumination constraints) to 500 Hz.

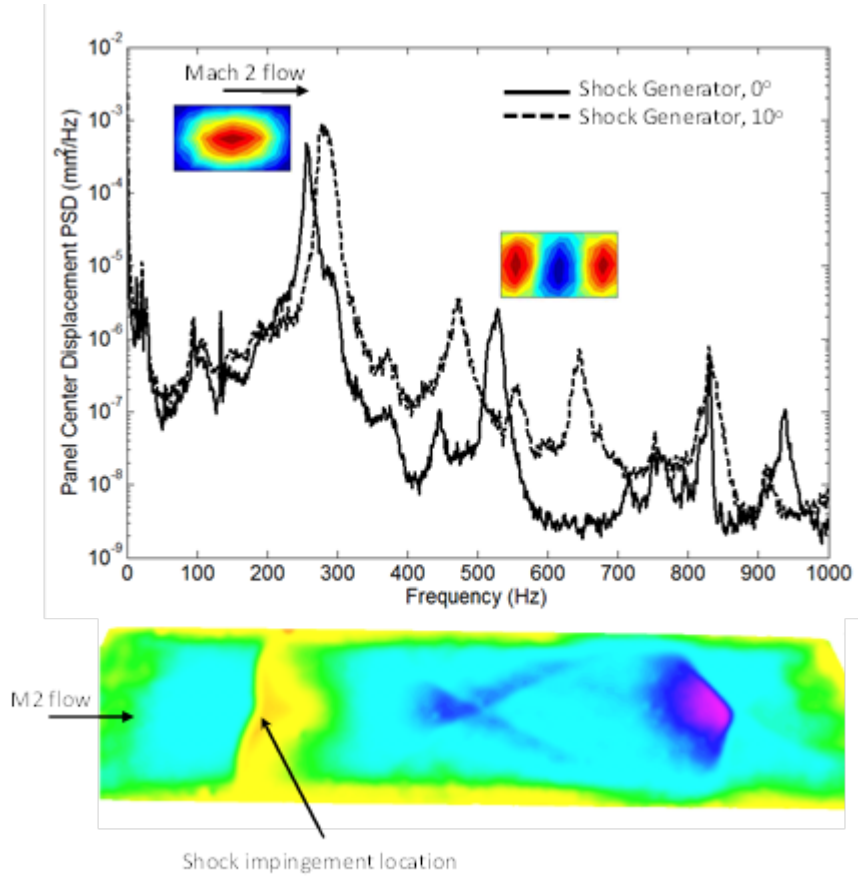


Figure 14: 3D DIC displacement results with shock generator at two settings - flush with the tunnel wall and elevated 10°. Also shown is the corresponding sample PSP image denoting the shock impingement location on the panel ($P_o = 270$ kPa).

An obvious benefit of the 3D full-field DIC measurements is the ability to simultaneously interrogate a number of points and thus describe the deflected shapes as a function of frequency. It is clear from the deflected shapes shown in Figure 14 that the first panel mode of vibration dominates the response. The deflected shapes displayed in Figure 14 are the result of applying output-only modal analysis to the full-field displacement (output) results using the 3D DIC measurements versus traditional input-output modal analysis [28]. The presented deflected shapes are the singular vectors resulting from the decomposition of the Hermitian auto- and cross-spectral density matrix. To arrive at the shapes, the DIC measurement points are first transformed into the frequency domain via the Fast Fourier Transform (FFT). The FFT spectral estimation was processed using a blocksize of 2048 data points and a Hanning window with 50%

overlap. Next, the auto- and cross-spectral density matrix, G_{yy} , was populated at discrete frequencies. The main diagonal terms are the auto-spectral densities while the off-diagonal terms are the cross-spectral densities of the output DIC measurement points. The spectral

density matrix is Hermitian so $G_{pq}(\omega) = G_{qp}^*(\omega)$ for $p \neq q$, where the asterisk denotes the complex conjugate. Finally, the spectral density matrix is decomposed at each i^{th} frequency using singular value decomposition (SVD). The singular vectors resulting from the decomposition of the power

spectral density matrix approximate the system mode shapes for a lightly-damped system under broadband-excitation. This approach is similar to the Complex Mode Indicator Function (CMIF), where the FRF matrix is decomposed at each frequency using SVD. The magnitude of the singular values at each frequency denotes the relative modal contribution. Each of the operational deflected shapes in Figure 14 corresponds to the maximum singular value for that particular frequency. These resulting deflected shapes were a bit crude that first year of testing due to the limited DIC facet locations across the panel (see Figure 10), but the methods used certainly showed promise and were used quite successfully to interrogate the results of later testing.

The first year of testing concluded when the bond failed between the panel and the panel frame. The failure occurred when a normal shock was held fixed on the panel during tunnel start-up/shut-down. Figure 15 shows the normal shock and resulting strain gage reading at the panel $\frac{1}{2}$ -point ($x/L = 0.5$, see Figure 11a). Notice the peak strain reading – nearly $800 \mu\epsilon$. More on the measured panel strain later, but it goes without saying that the consequence of such an abrupt, singular event could prove catastrophic for a high-speed aerospace vehicle.

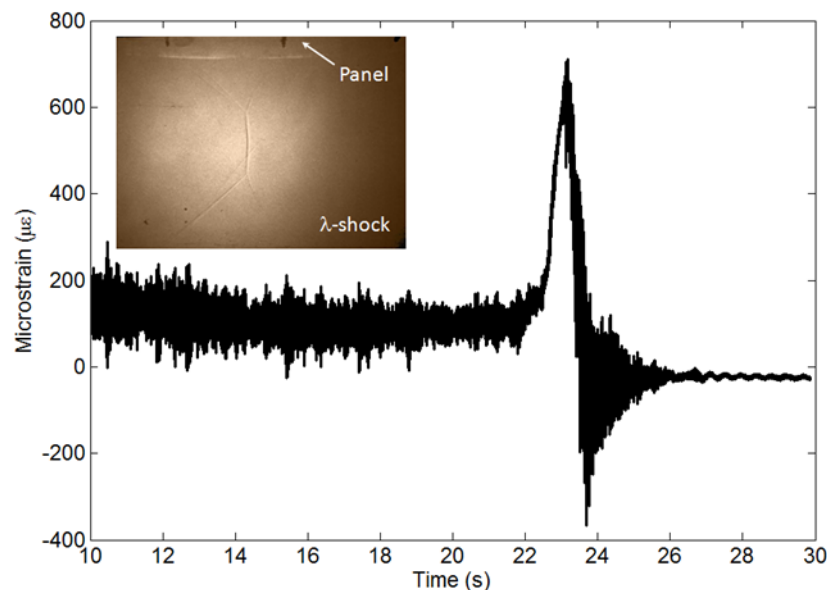


Figure 15: RC-19 Tunnel shut-down with normal (lambda) shock and corresponding panel strain gage time-history

5 SECOND RC-19 TUNNEL ENTRY (2012)

The second year of testing provided an opportunity to refine the specimen design (manufacture an integral frame thin-panel) to avoid a “bond-line” failure to and improve the 3D DIC and PSP sampling and time record length to explore longer transients and record longer (steady) records for averaging. Further, the team began to focus on the heated flow conditions that would lead to interesting post-buckled dynamic behavior. The tunnel illumination was improved for the fast reacting PSP and as a result the sampling rate was increased from 500 Hz to 4 kHz, likely beyond the dynamics relevant to the thin-panel. There were nearly 15 seconds of fast reacting PSP data recorded for every test condition. The 3D DIC was also sampled at 4 KHz to match the PSP with 22 seconds of full-field displacement data recorded for each test condition. The tunnel test section was further modified to use a solid 8° wedge, 10.2 mm in height, to turn the flow and impinge a shock at different stream-wise panel locations.

The typical panel spectral response to supersonic flow and shock impingement is shown In Figure 16. The deflected panel shapes (operational deflected shapes) provide much more detail due to the greater number of points used during the post-processing of the DIC images - 153 points versus the 21 used in the first year of testing. Also included in Figure 16, on the frequency axis, are the unstressed measured modal frequencies. The stiffening effect of the aeroacoustic loading can be observed.

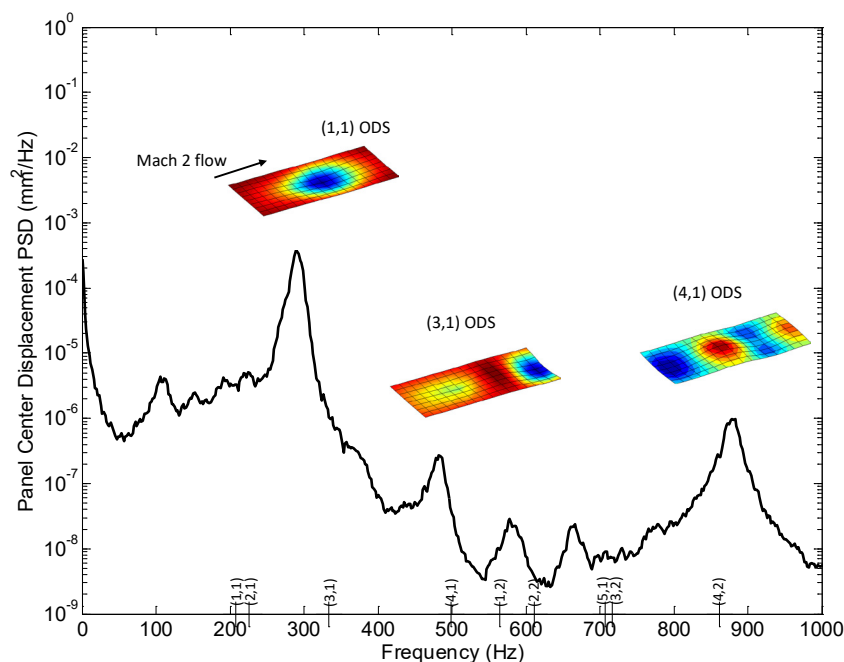


Figure 16: Panel center displacement PSD with corresponding full-field deflected shapes measured using 3D DIC.

Next, in order to gauge the sensitivity of the panel response to the location of SBLI, the solid 8° wedge was moved to two different locations. The panel response was then measured at those different shock impingement locations for the same tunnel operating conditions (total pressure, $P_o = 345$ kPa). The results are shown in Figure 17. The response power spectral density (PSD)

is significantly reduced when the shock impingement location was in "Position #2" or near the panel stream-wise mid-point ($x/L = 0.5$). In addition to the panel center response PSD, the full-field, averaged PSP image for each case is also displayed. The average pressure increases by a factor of two as a result of the shock impingement and then subsides to the pre-shock pressure near the trailing edge of the panel.

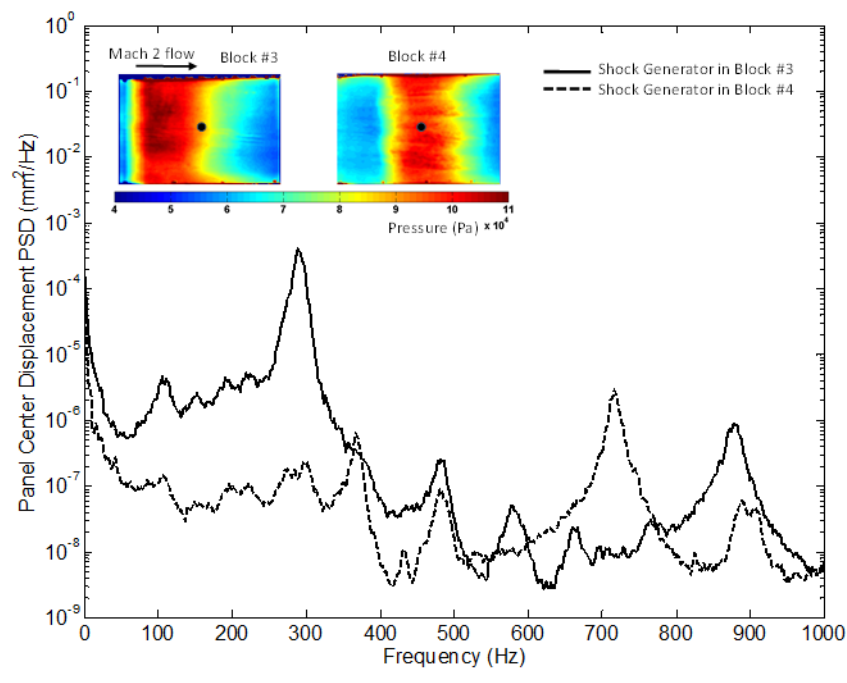


Figure 17: Sensitivity of panel response to location of Shock Boundary-Layer Interaction (SBLI).

Finally, the panel response (full-field DIC and single point velocity and strain) was recorded for a single heated flow test condition. There was also a shock impingement at "Position #1." The authors were attempting to identify the conditions that would lead to post-buckled and possibly snap-through or limit cycle aeroelastic behavior. The tunnel flow was initially heated to approximately 93°C (200°F), the minimum setting/temperature for operating the tunnel heaters. Flow to the tunnel test section, at the desired testing conditions (total pressure, $P_0 = 345$ kPa) was initiated once the target flow temperature was reached. The effect of the heated flow on the panel response is shown in Figure 18. Surprisingly, there was not an appreciable difference in response for the selected testing conditions. The dominant modal frequencies, and the corresponding deflected shapes, are quite similar between the unheated and heated tunnel conditions. For the heated flow case, the panel did deform into the tunnel and away from the cavity by 2 mm (a bit more than 3 panel thicknesses), but there was no way to determine whether the panel deformation was beyond a critical point thus indicating multiple stable equilibrium. The temperature difference between single thermocouples on the panel and frame was approximately 17°C. The displacement response PSD does indicate a broadening-behavior in the first mode which could be the effect of geometric nonlinearity at play.

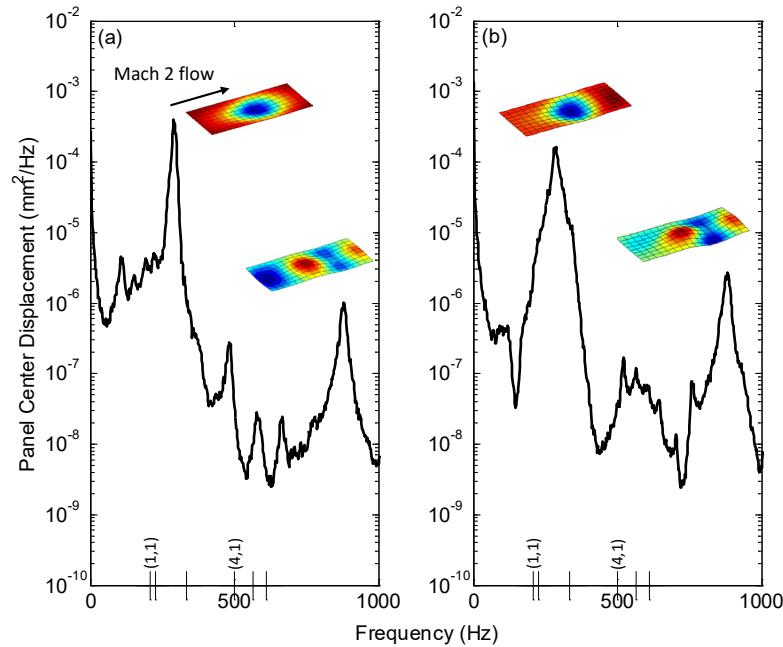


Figure 18: Year 2 (a) unheated and (b) heated flow panel center displacement response.

In conclusion, the second year of testing allowed the authors to further refine the 3D DIC methods resulting in an improved understanding of the panel response (see the difference in the deflected shapes in Figure 14 and Figure 16). The machined panel behaved very well and did not experience a failure like the bonded one used during the first year of testing. Pre- and post-test modal measurements indicated no appreciable differences in the panel frequencies and damping (see Table 1). The illumination for the fast-reacting full-field PSP was also improved, resulting in an order of magnitude increase in sampling rate. Finally, a heated flow/SBLI condition was also investigated in order to (hopefully) identify testing parameters that would result in more extreme dynamic behavior, i.e., snap-through dynamic buckling, limit cycle oscillations, etc.

6 FINAL RC-19 TUNNEL ENTRY (2016 - 2017)

The final year of testing in the RC-19 facility provided the most complete and substantive data set yet. Full-field 3D DIC, PSP, TSP and shadowgraph dynamics were all recorded for unheated flow conditions with varying shock location. Panel displacement using 3D DIC, along with discrete temperature, strain and velocity were measured for the heated flow conditions. Unfortunately, the PSP and TSP were not able to survive even the lower heated flow testing conditions which is a notable limitation for these paint-based full field techniques. One obvious solution to the inability of TSP to survive the higher-temperature flow environment was to use Forward Looking Infrared Radiometer, or FLIR. That was the intent, but it was quickly discovered that the quartz viewing windows were exceptionally good at filtering infrared light! The setup in this final year of testing was similar to the 2012 experimental campaign with several exceptions. Importantly, three new SA-Z Photron high-speed cameras were purchased for this experiment. Two were used for the 3D DIC measurements while the third was used to film the fast-reacting PSP or TSP for the cold flow conditions. Two SA-5 Photron high-speed cameras were used to film the response of the thin panel through the flow so as to gauge the error introduced into the displacement measurements by distortions (density gradients) in the supersonic flow. It is anticipated that future experiments, in different wind tunnels, will not allow for the panel to be incorporated into the facility side-wall and will therefore require filming the response of a structural test article through the flow. A photograph of the facility with all of the cameras is shown in Figure 19. All of the cameras, with the exception of the TSP measurements, were sampled at 5 kHz. The total record length for the 3D DIC was now extended to 30 seconds.

There were five different SBLI conditions - no-shock, a shock impingement at the panel leading edge, 1/8-point, quarter-point, and mid-point in the stream-wise direction.

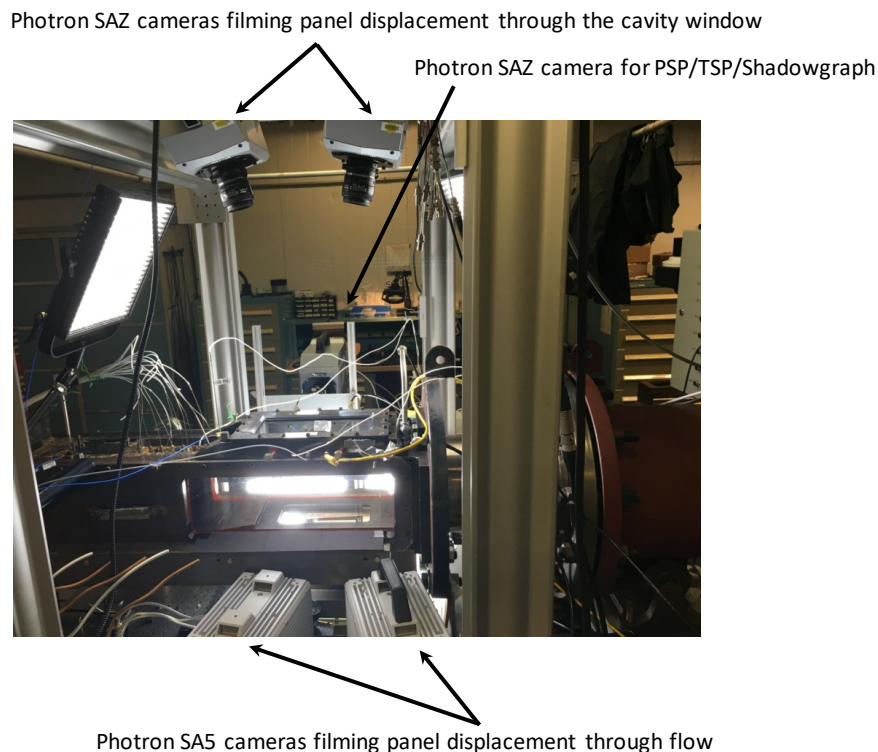


Figure 19: RC-19 experimental setup for simultaneous filming (DIC) of the panel dynamic response through the cavity and through the SBLI flow.

Traditional measurement techniques were also employed as in past RC-19 thin-panel experiments. Their purpose was twofold: (1) to provide a check with the full-field methods, e.g., single point LDV and the corresponding 3D DIC location, and (2) to track the "real-time" response of the panel during test as a measure of consistency and repeatability. For example, since PSP and TSP could not be used simultaneously, the panel velocity and strain were monitored to ensure consistent conditions and results between testing nights. The traditional techniques included the same Polytec dual-beam LDV to record the panel velocity, a single strain gauge at the panel stream-wise edge/mid-point perpendicular to the flow direction, thermocouples on the thin-panel edge and the panel frame, and Kulite dynamic pressure sensors at various points in the tunnel test section.

This final year of testing was also used to explore the response of two different thin-panel widths - 10.2 cm (4 in) and 12.7 cm (5 in) wide. The panels were the same 25.4 cm (10 in) length and 0.635 mm (0.025 in) thickness. A rigid, and instrumented, control specimen was also fabricated for this last series of experiments. The intent of the control specimen was to provide a means to directly compare the dynamic, fast-reacting PSP pressure paint to the more common, traditional Kulite pressure sensors and characterize the boundary-layer as was previously discussed.

It was noticed during this last test year that a significant pre-load was introduced into the test article when the integrated panel/frame was attached to the tunnel top-wall. This was the first time this initial deformation was both noted and measured. This pre-load resulted in a maximum transverse deflection of the (12.7 mm wide) thin-panel, 0.8 mm into the tunnel. This was merely a result of the installation, i.e., before even starting the RC-19 tunnel and certainly before any

thermal influence. This initial deformation was measured using the 3D DIC and will be used subsequently to update and inform planned modeling efforts [29].

The PSP and TSP full-field images from the rigid control panel, corresponding to a shock location near $x/L = 0.25$, are shown in Figure 20. The TSP image (Figure 20b) shows quite a bit of measurement noise. While the general temperature trend follows the accompanying pressure one, it is likely that two factors contributed to the poor temperature results. There were difficulties in applying the paint consistently to the rigid control specimen in this particular instance, and there was significant condensation on the viewing window for that test night.

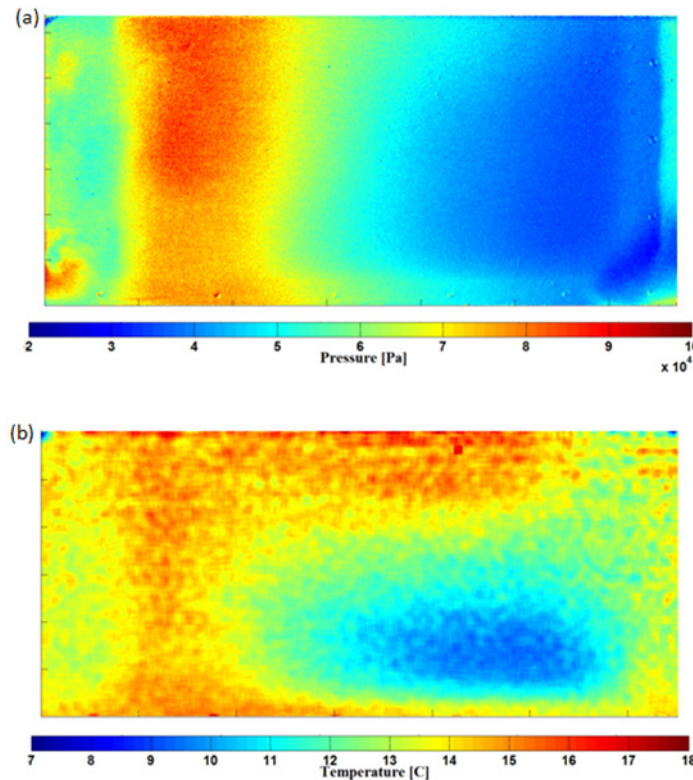


Figure 20: Full-field PSP (pressure) and TSP (temperature) images corresponding to the flow/SBLI conditions.

A single high-speed camera was also used to record the dynamics of the shock, and SBLI using a shadowgraph setup during several of the RC-19 test conditions (see Figure 21). This is a case where the shock impingement is near the panel $\frac{1}{2}$ -point ($x/L = 0.5$). It is quite clear from the shadowgraph image spectra that the overall dynamic/energy content is significantly greater than the pre-shock location, and that there is appreciable low-frequency content well-below 500 Hz – well in the range of the first vibration mode of the thin-panel. This is the type of potentially troublesome dynamic loading that Dolling [13] and Pozefsky [14] have earlier referred to.

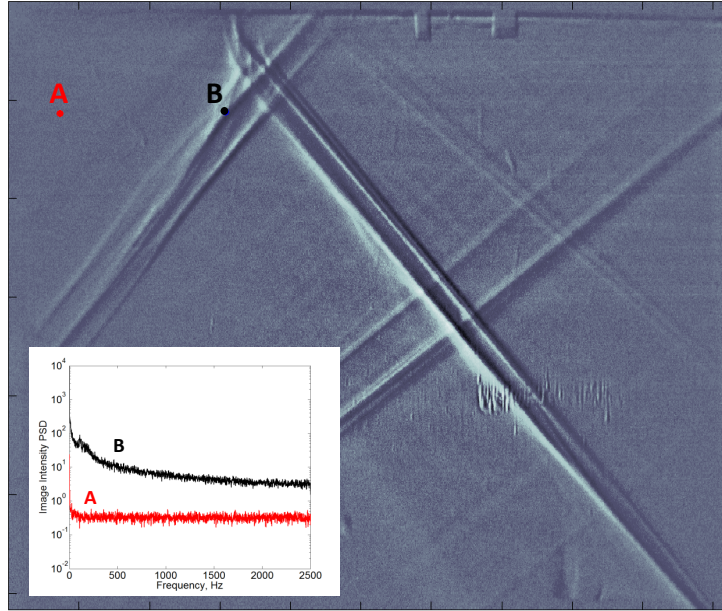


Figure 21: High-speed shadowgraph image and associated shock dynamics for (a) one point upstream of the wedge shock-generator, and (b) a single point near the location of initial impact on the thin-panel.

The use of the Kulite dynamic pressure transducers proved quite useful for several reasons. In addition to providing general tunnel characterization, the dynamic response of the thin panel was captured in one Kulite 51 mm down-stream of the thin-panel trailing edge (see the red curve in Figure 22). The noticeable peak in the pressure PSD occurs at 275 Hz, the dominant first-mode response of the panel to these flow conditions. To the authors' knowledge, this is one of the few instances of this phenomena being observed experimentally, and at such a great distance from the dynamically responding thin-panel – approximately 3 boundary layer thicknesses.

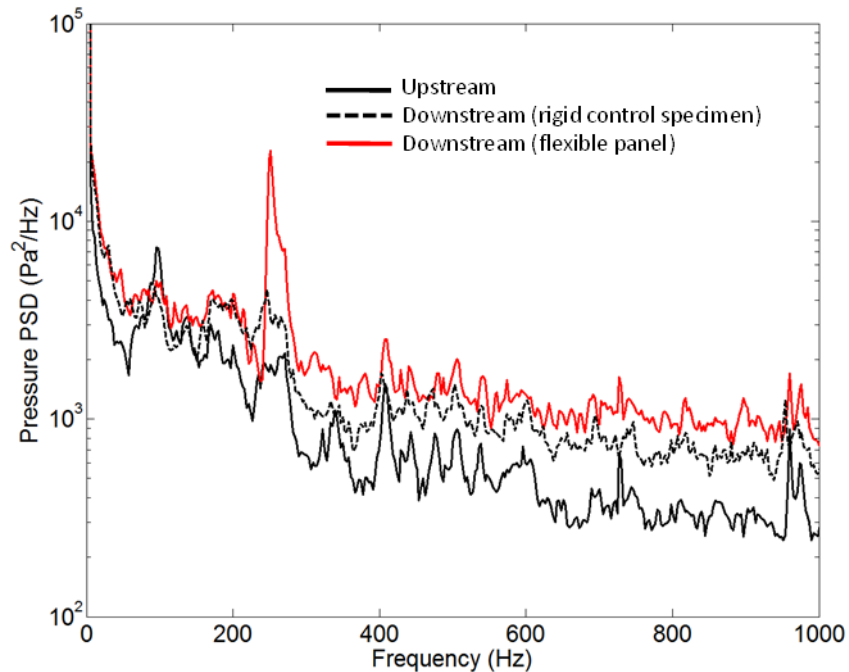


Figure 22: Captured panel dynamic response (red curve) in down-stream tunnel Kulite pressure measurement.

Note: The dashed black curve denotes the same measurement location with the solid, control specimen in place upstream, while the solid black curve is the upstream dynamic pressure measurement.

The RC-19 facility has exceptional access for optical measurement techniques and greatly simplified the testing (from a measurements point of view) of the thin-panel. Since the thin-panel specimens were integrated into the top wall of the tunnel, it was relatively straightforward to create the cavity with a window (Figure 3) and measure the response from the panel backside.

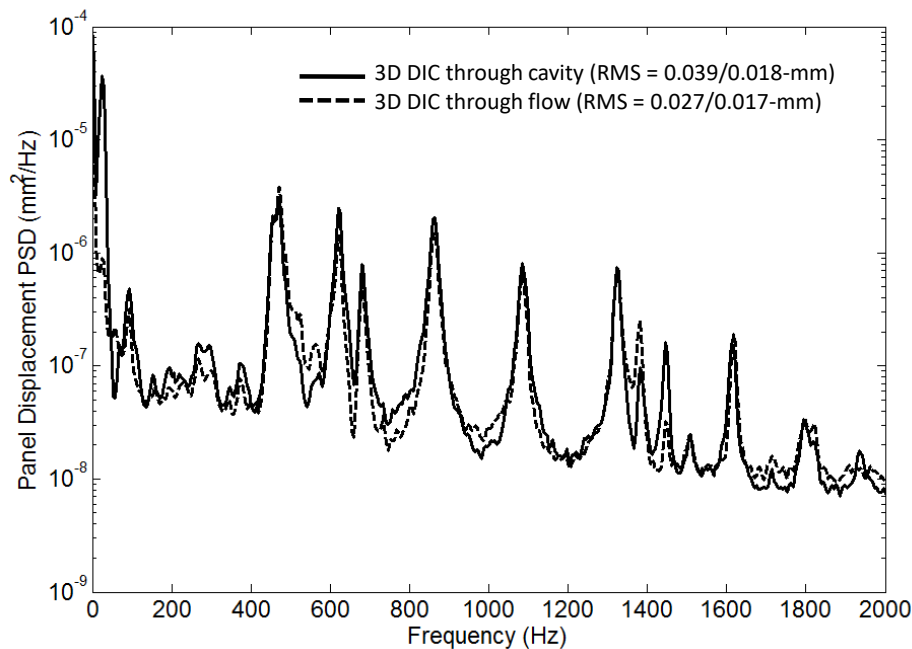


Figure 23: Filming the panel displacement response from the cavity and panel flow-side using a dual-DIC camera arrangement. There is a shock impingement at the panel midpoint, $x/L=0.5$.

This will not always be the case. Many wind tunnel facilities use a test “wedge” or some other specimen fixture mechanism to insert the flexible test article [22,23,24], material sample or other rigid specimen, into the core flow of the tunnel. In this scenario, measuring the response of a flexible specimen using 3D DIC becomes an exercise of balancing appropriate optical access and lighting, all while managing or quantifying the noise due to measured distortions/density-gradients in the flow. The present series of RC-19 experiments provided a ready means of gauging the efficacy of filming/measuring the panel dynamic response simultaneously from the cavity side and through the flow on the panel flow-side. The experimental tunnel configuration for this case was at the same Mach 2 flow and total pressure (345 kPa) and included an oblique shock-impingement at the panel mid-point. The camera arrangement for this dual-DIC setup is shown in Figure 19. The results of this exercise is quite promising as seen in Figure 23 – there is very little difference in spectra between the flow and back-side measurements. It was noted that one effect of filming the displacement through a shock-wave was an increase in the measurement noise, particularly below 1 kHz [30]. This is an area of active research, and future tests with more severe environments/distortions are planned.

Next, the response of the thin-panel to heated flow was studied. Two different scenarios were considered: (1) the panel response to transient pressure loading and heating, i.e., heated-flow tunnel start-up, and (2) the panel response to steady total-temperature/pressure flow conditions with cavity pressure modulation. It was this second scenario, the steady heated flow pressure modulation study, that led to excessively high panel strain and ultimately to the failure of the thin-panel thus ending the experiments. Interestingly, it was not the real-time discrete foil gage that indicated the impending high-cycle fatigue, but post-processing of the 3D DIC data after the fatigue crack was noticed at the completion of the test. The lesson there is that a discrete sensor,

used for in-situ monitoring of a test specimen, could very well miss critical response. The following discussion on panel displacement and strain are taken from the region highlighted in Figure 24.

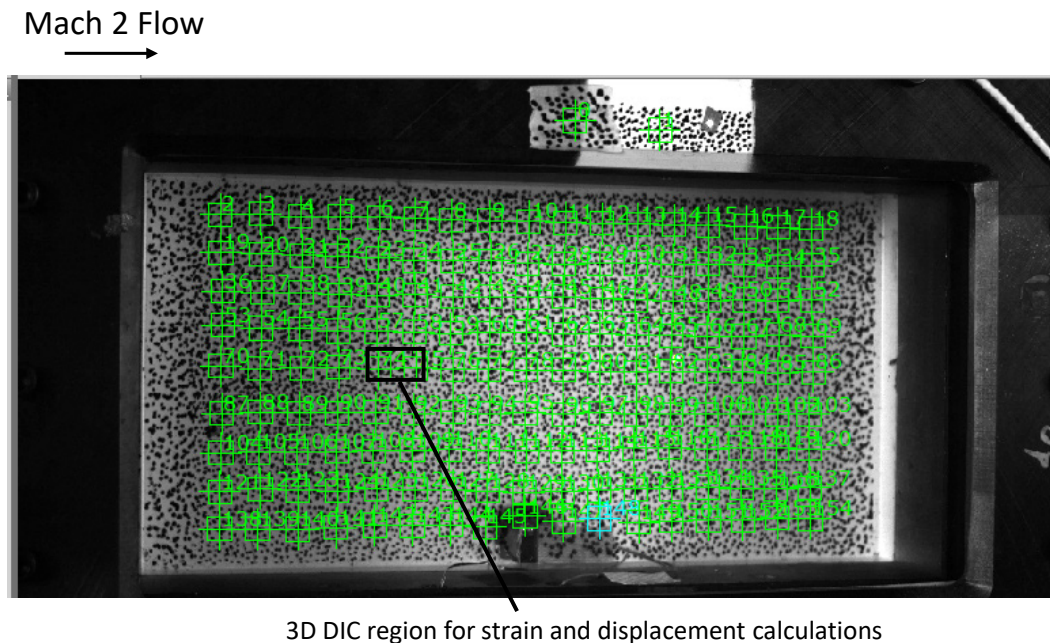


Figure 24: Heated flow test panel specimen with DIC pattern.

The time history of the tunnel start-up for heated flow is shown in Figure 25, and there are several key features worth discussing. First, note that the total time history of this recorded transient is 30 seconds, limited by the total record length capture of the Photron SAZ cameras/3D DIC setup. The tunnel never reached the maximum (desired) total pressure during that 30 second transient, as evidenced by the plot of total pressure. Once the 3D DIC camera memory was full, and the recording complete, the tunnel was shut-down. The start of the tunnel can be seen in the displacement time history, where the effect of a normal shock traversing the panel can be seen just before the 5 second mark. The pressure rises quickly after the normal shock crosses the thin-panel, while the panel temperature rise is of course delayed by several seconds. Even with these tunnel start-up and camera timing issues, some interesting behavior was noted. The panel response moves through a large-deformation region (Region 1 in Figure 25) before settling into what appears to be the single well of a bi-stable system (Region 2). It is not known if indeed the potential at the end of the transient time-history is truly “double-well” as there was no way to gauge this during testing. Needless to say, the response is certainly quite interesting as denoted by the zoom-in window/region between 23 and 24 seconds. The appearance is of snap-through buckling behavior, but even if not snap-through, the deformation is certainly excessive. The panel full-field displacement data, for each of the “Regions” called-out in Figure 25, was analyzed using proper orthogonal decomposition in order to determine the dominant deflected shapes (see Figure 26). Not only did the order of the dominant proper orthogonal mode change, but so too did the relative importance (weight of the proper orthogonal values) and therefore the dimension of the response [31]. This type of complex, transient response change will have a direct effect on the structural modeling requirements.

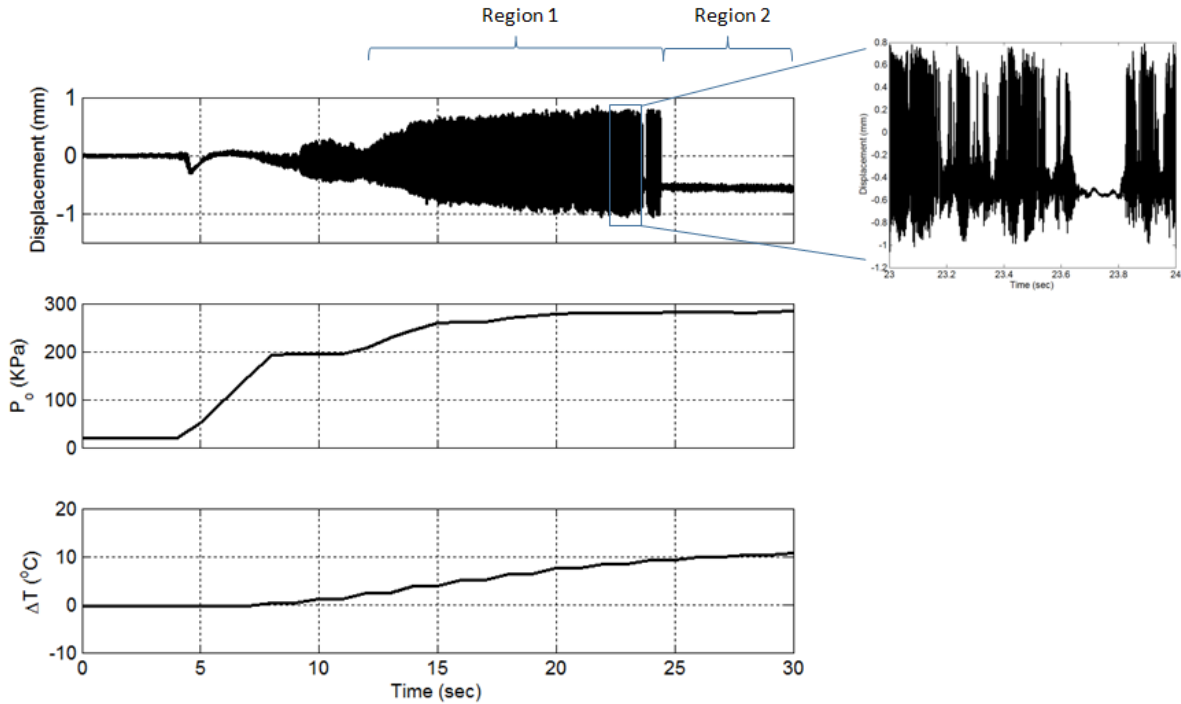


Figure 25: Transient panel response to start of RC-19 tunnel and heated flow.

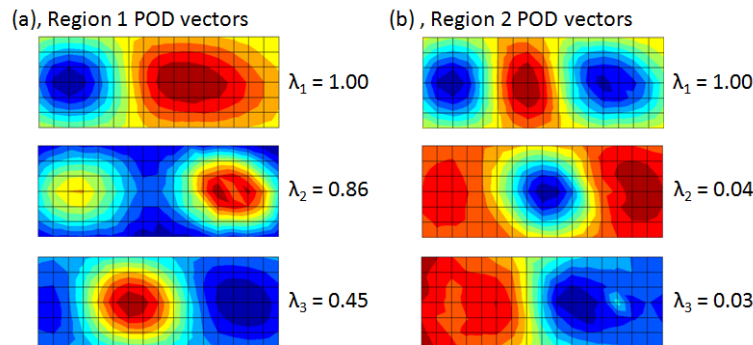


Figure 26: Resulting POD vectors and associated proper orthogonal values for Region 1 and 2 full-field displacement time regions in Figure 25.

In order to explore (encourage) the possibility of bi-stable, cross-well behavior it was decided next to settle on a steady total-temperature/pressure flow testing condition and then modulate the cavity pressure. Reducing the cavity pressure by 2.1 kPa led to the very large amplitude response displayed in Figure 27. Note that the near continuous large, peak-to-peak displacements and strains are 2 mm (0.63 mm RMS) and approaching 10,000 $\mu\epsilon$ (1470 $\mu\epsilon$ RMS) respectively. The PSD of the panel transverse response also displays some interesting characteristics – namely what appears to be strong periodic response with harmonics.

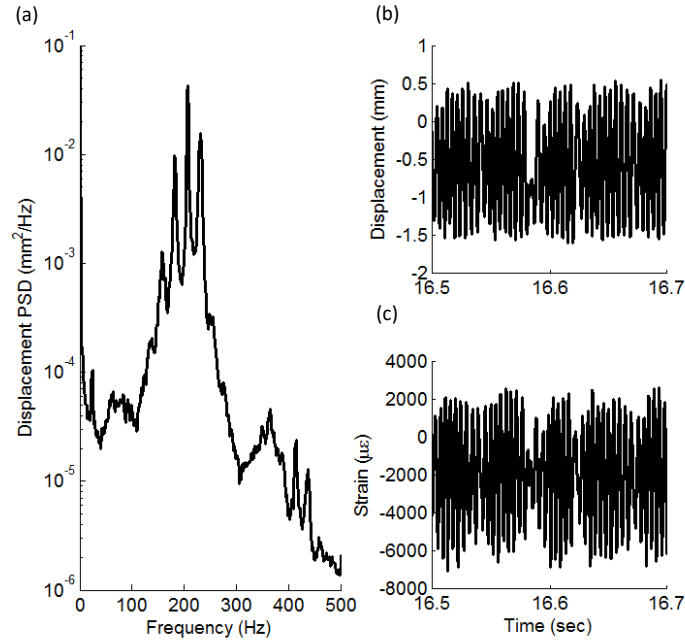


Figure 27: Large amplitude panel response to heated supersonic flow when cavity pressure was reduced by 2.1 kPa (0.6 psi, nominal = 7.8-psi).

Next, the cavity pressure was further reduced resulting in an increase in the peak-to-peak displacements and strain (Figure 28). The strain was calculated using the 3D DIC in the region highlighted in Figure 24. Again, two distinct regions are called-out in the time-histories (Figure 28a/b), and the dominant POD vectors for those respective regions shown in Figure 28c. The character of the dominant deformed shape changes between the regions, from a first to second-symmetric panel mode. The response is clearly intermittent, transitioning between a large amplitude response and a very small amplitude response. This likely indicates the presence of co-existing solutions [32], a troublesome situation for an aerospace structural designer faced with the difficulty of confirming the conservatism in a structural design operating in a nonlinear regime. The strain time-history shown in Figure 28b, displays peak-to-peak values range approaching 15,000 µε (3300 µε RMS).

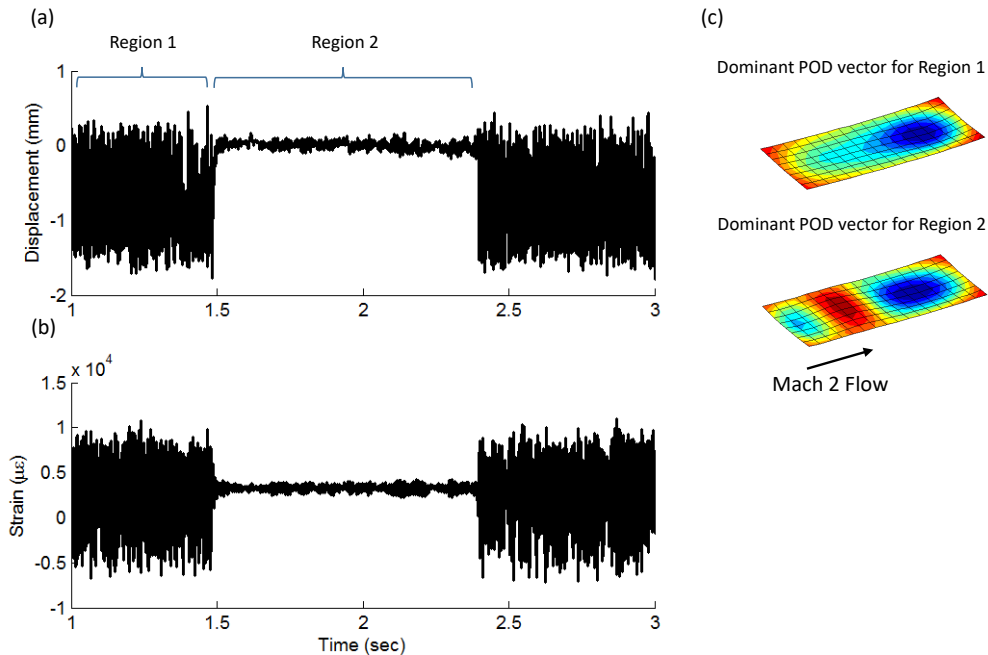


Figure 28: Heated flow (12.7 cm/5-in panel) strain displacement and strain response at the panel 1/4-point ($x/L=0.25$). Pressure in cavity reduced by 3 kPa (0.44-psi, nominal = 7.9-psi)

It was this excessive dynamic strain that led to the panel failure shown in Figure 29. The crack, highlighted in the accompanying window, occurred at the trailing edge of the panel. The time-history displayed in Figure 28 is repeated in Figure 30a, for a location near the region where the crack occurred. In fact, it was determined (using the 3D DIC images) that the panel crack occurred sometime during the consecutive heated flow runs shown in Figure 30a-c, and likely during the extreme, near continuous large amplitude test of Figure 30b.

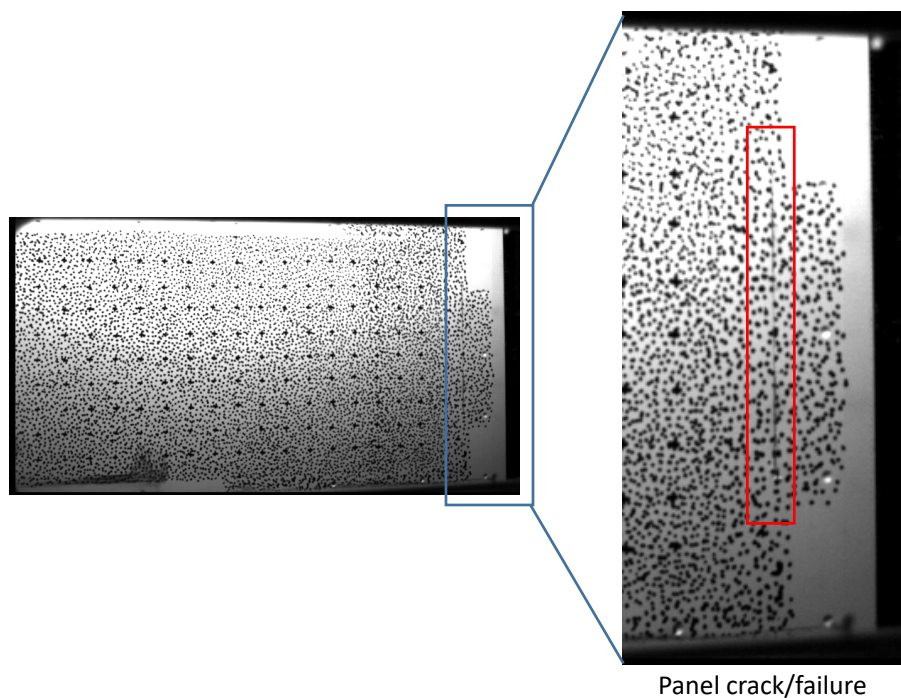


Figure 29: Top surface of thin panel (12.7 cm/5-in panel) with span-wise through-crack in the region of the trailing edge.

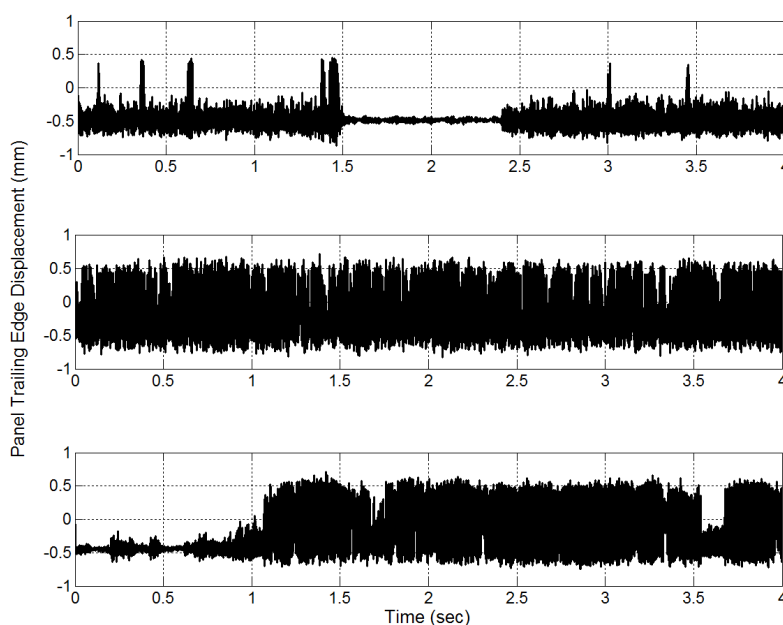


Figure 30: Displacement time-history near the 12.7 cm (5-in) panel trailing edge for three consecutive heated flow test conditions.

Notes: (a) intermittent large displacement, (b) continuous large displacement, and (c) intermittent and now damaged panel.

Finally, it was desired during this most recent experimental campaign, to compare the fast reacting PSP and discrete Kulite pressure sensors. For this case, a near-coincident PSP/Kulite point on the rigid specimen was selected, and then the spectral response of the Kulite used to identify the appropriate spatial scaling for the fast reacting PSP. The results of that comparison can be seen in Figure 31 for the nondimensionalized pressure and frequency. It was found that a 9x9 pixel array from the PSP camera images provided a reasonable comparison with the Kulite data.

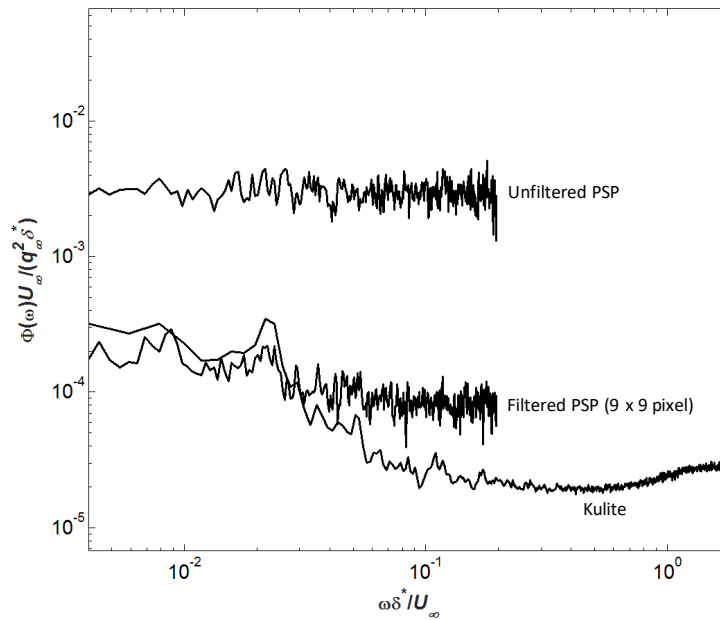


Figure 31: A comparison of Kulite and PSP dynamic pressures with/without spatial averaging ($U_\infty = 490 \text{ m/s}$, $q_\infty = 123 \text{ kPa}$, $\delta^* = 8.6 \text{ mm}$).

The result of that exercise in spatial averaging was next used as the basis for presenting the fast reacting PSP for a representative SBLI result (shock-impingement near the rigid and thin-panel leading edge), superposed onto non-dimensionalized pressure spectra from legacy NASA experimental data [16]. That legacy data comparison is shown in Figure 32a for three different locations on the rigid panel. The relative location along the panel length, for each of those three color-coded points is noted in Figure 32b. The points selected represent three different values of dynamic/fluctuating pressure, with the peak (point 3) at the shock impingement location. The effect of moving closer to the shock impingement location, i.e., from points 1-to-3, can be seen in the increasing pressure spectra. Interestingly, the amplitude of the spectra never reaches the earlier published value for shock impingement, but instead is close to the spectra for separated flow conditions. Figure 32c compares the RC-19 rigid and flexible non-dimensionalized pressure spectra for the same wind-tunnel conditions (point 4 in Figure 32b). The panel dynamics (1st and 3rd panel vibration modes) can clearly be seen in spectra. Again, this pressure spectra came from the spatially averaged fast reacting PSP.

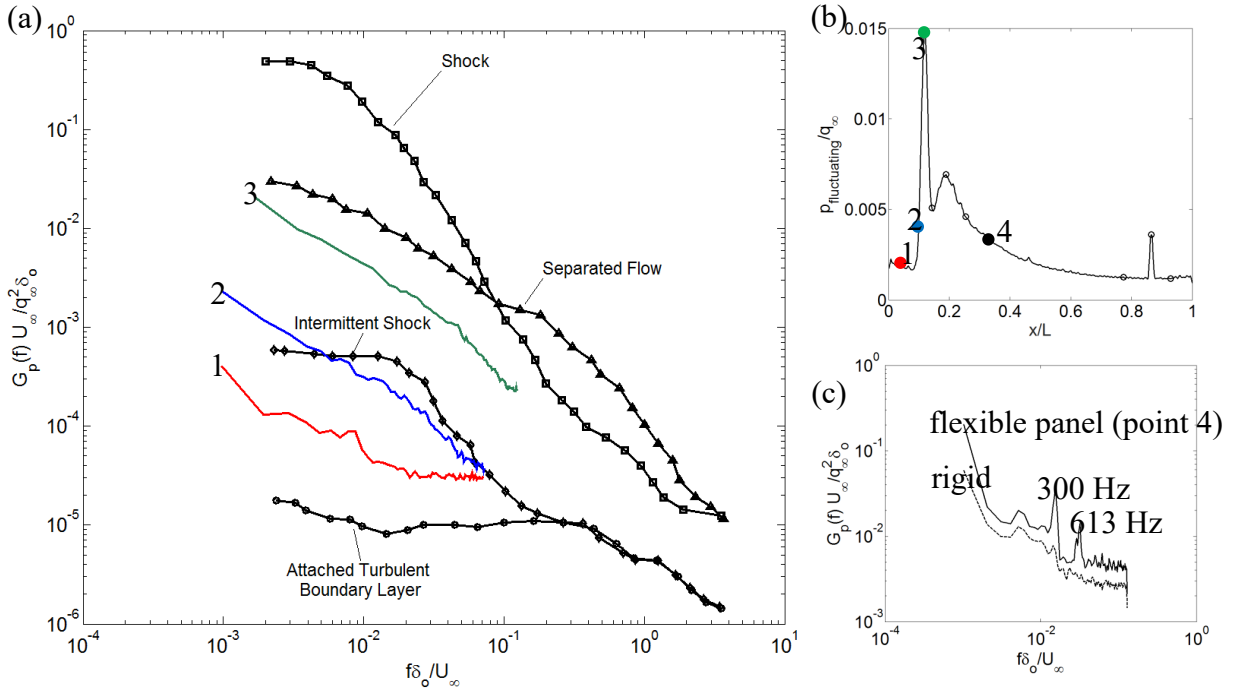


Figure 32: RC-19 (a) measured pressure spectra; (b) PSP measurement location and the variance of measured pressure; and a comparison (c) between the rigid control and thin-panel specimens (Points 1 & 2: $U_\infty = 490 \text{ m/s}$, $q_\infty = 123 \text{ kPa}$, $\delta^* = 8.6 \text{ mm}$, Points 3 & 4: $U_\infty = 287 \text{ m/s}$, $q_\infty = 123 \text{ kPa}$, $\delta^* = 8.6 \text{ mm}$).

7 CONCLUSION

The RC-19 experimental campaign was quite the audacious undertaking, the purpose of which was to observe and measure the response of a thin-panel (aero-structure inspired) to supersonic, turbulent flow, shock boundary-layer interactions with and without the effects of heating. Large deformation, geometric nonlinear response was also desired. In parallel with the discovery/observation aspects of the experiment, innovative full-field measurements were being developed and refined throughout the experimental campaign. Finally, it was hoped from the earliest wind-tunnel entry, that testing conditions leading to specimen failure could be identified. Panel failure (high-cycle fatigue crack near the trailing edge) was finally observed in the last series of experiments in 2017. The present article represents highlights of the total experimental effort, emphasizing several notable achievements. For example, the dynamic response of the panel (1st mode of vibration) was captured in a Kulite 51 mm downstream of the thin-panel trailing edge. Another key achievement was the demonstrated ability to film the thin-panel dynamic response through the supersonic flow and impinging shock-wave. For the present tunnel test conditions, the measurement noise was manageable (and quantified). In the most recent year of testing, exciting results were obtained for the heated flow conditions. Both transient and steady operating conditions were recorded, with large deformation (possibly cross-well bi-stable), high-strain dynamic response leading to Panel failure. Given the sudden change in panel response to cavity pressure modulation, it is strongly suspected that the observed response was likely limit cycle oscillation. Finally, near-coincident fast reacting PSP and more traditional Kulite sensors were compared and the Kulites used to determine the appropriate spatial averaging for the PSP. Once that appropriate averaging was determined, the present rigid and thin-panel PSP data was compared with legacy NASA wind tunnel experiments. It was noted that the thin-panel dynamics were captured in the PSP measurements. Future plans include extending the lessons learned in the RC-19 supersonic tunnel to a more severe aerothermal environment with a structural specimen exhibiting bit more complexity. A more severe environment will push every aspect of the design, observation and data collection. The extensive series of thin-panel RC-19 experiments were quite the audacious undertaking. The results have been more than promising and the authors are quite confident that quantitative advances will continue to be made.

LIST OF SYMBOLS, ABBREVIATIONS, AND ACRONYMS

ACRONYM	DESCRIPTION
3-D	three dimensional
6-DOF	six degrees of freedom
AFRL	Air Force Research Laboratory
alpha	angle of attack
AoA	angle of attack
b	lateral-directional reference area (wing span = 66 ft)
c	longitudinal reference area (MAC = 66.2 ft)
c.g.	center-of-gravity
CAP	control anticipation parameter
cat	category
C _D	drag coefficient
CESTA-CD	Control Effectors for Supersonic Tailless Aircraft – Concept Demonstration
CFD	computational fluid dynamics
CFL	upwind differencing scheme
CG	center-of-gravity
C _L	lift coefficient
C _I	rolling moment coefficient
C _m	pitching moment coefficient
CMBA	continuous moldline bump array
CMT	continuous moldline technology
α	angle of attack
β	sideslip angle
ω_{DR}	dutch roll frequency
°	degrees

END NOTES

[¹] M. Mignolet, A. Przekop, S. Rizzi, S. Spottswood, "Non-Intrusive Finite Element Based Reduced Order Modeling of Nonlinear Geometric Structures," *Journal of Sound and Vibration*, Vol. 332, May 2013, pp. 2437-2460.

[²] A. Gogulapati, R. Deshmukh, J. McNamara, V. Vyas, X.Q. Wang, M. Mignolet, T. Bebernis, S.M. Spottswood and T. Eason, "Response of a Panel to Shock Impingement; Modeling and Comparison with Experiments - Part 2," 56th AIAA/ASME/ASCE/AHS/ASC Structures, Structural Dynamics and Materials Conference, January 5-9, Kissimmee, FL, 2015.

[³] M.H. Morton, "Certification of the F-22 Advanced Tactical Fighter for High Cycle and Sonic Fatigue," 48th AIAA/ASME/ASCE/AHS/ASC Structures, Structural Dynamics, and Materials Conference, 23-26 April 2007, Honolulu, Hawaii, AIAA Paper 2007-1766.

[⁴] S.L. Liguore and T.H. Beier, "Recognition and Correction of Sonic Fatigue Damage in Fighter Aircraft," Paper presented at the RTO AVT Specialists' Meeting on Life Management Techniques for Ageing Air Vehicles, Manchester UK, 8-11 October 2001, RTO-MP-079(II).

[⁵] A. Frendi, "Coupling Between a Supersonic Turbulent Boundary Layer and a Flexible Structure," *AIAA Journal*, Vol. 35, No. 1, January 1997.

[⁶] R.W. Gordon and J.J. Hollkamp, "Reduced-Order Models for Acoustic Response Prediction," AFRL-RB-WP-TR-2011-3040, July 2011.

[⁷] S.M. Spottswood and M.P. Mignolet, "Experimental Nonlinear Response of Tapered Ceramic Matrix Composite Plates to Base Excitation," *AIAA Journal*, Vol. 40, No. 8 (2002), pp. 1682-1687.

[⁸] E.A. Thornton, "Thermal Structures: Four Decades of Progress," *Journal of Aircraft*, Vol. 29, No. 3, May – June 1992.

[⁹] T.L. Lewis and N.J. McLeod, "Flight Measurements of Boundary-Layer Noise on the X-15," NASA TN D-3364, National Aeronautics and Space Administration, Washington D.C., March 1966.

[¹⁰] J.D. Watts, "Flight Experience with Shock Impingement and Interference Heating on the X-15-2 Research Airplane," NASA TM X-1669, National Aeronautics and Space Administration, Washington D.C., October 1968.

-
- [¹¹] E.A. Thornton, J.T. Oden, W.W. Tworzydło and S.K. Youn, "Thermo-Viscoplastic Analysis of Hypersonic Structures Subjected to Severe Aerodynamic Heating," 30th AIAA, ASME, ASCE, AHS, and ASC, Structures, Structural Dynamics and Materials Conference, Apr. 3-5, 1989, Mobile, AL, United States, AIAA Paper 89-1226.
- [¹²] N.T. Clemens and V. Narayanaswamy, "Low-Frequency Unsteadiness of Shock Wave/Turbulent Boundary Layer Interactions," Annual Review of Fluid Mechanics, Vol. 46, 2014, pp. 469-492.
- [¹³] D.S. Dolling, "Fifty years of shock wave/boundary layer interaction: what next?" AIAA Journal, Vol. 39, 2001, pp. 1517-1531.
- [¹⁴] P. Pozefsky, "Identifying Sonic Fatigue Prone Structures on a Hypersonic Transatmospheric Vehicle," AIAA 12th Aeroacoustics Conference, April 10-12, 1989, San Antonio, Texas.
- [¹⁵] R.D. Blevins, I. Holehouse, K.R. Wentz, "Thermoacoustic Loads and Fatigue of Hypersonic Vehicle Skin Panels," Journal of Aircraft, Vol. 30, No. 6, Nov.-Dec. 1993.
- [¹⁶] C.F. Coe and W.J. Chyu, "Pressure-Fluctuation Inputs and Response of Panels Underlying Attached and Separated Supersonic Turbulent Boundary Layer," Technical Paper presented at AGARD Symposium on Acoustic Fatigue, Toulouse, France, September 26-27, 1972, NASA TM X-62,189.
- [¹⁷] D.A. Kontinos and G. Palmer, "Numerical Simulation of Metallic Thermal Protection System Panel Bowing," Journal of Spacecraft and Rockets, Vol. 36, No. 6, November-December 1999. pp.842-849.
- [¹⁸] C.E. Glass, L.R. Hunt, "Aerothermal Tests of Spherical Dome Protuberances on a Flat Plate at a Mach Number of 6.5," NASA TP-2631, 1986.
- [¹⁹] C.E. Glass, L.R. Hunt, "Aerothermal tests of quilted dome models on a flat plate at a Mach number of 6.5," NASA-TP-2804, 1988.
- [²⁰] A.J. Culler and J.J. McNamara, "Impact of Fluid-Thermal-Structural Coupling on Response Prediction of Hypersonic Skin Panels," AIAA Journal, Vol. 49, No. 11, November 2011, pp.2393 - 2406.
- [²¹] L. Maestrello and T.L.J. Linden, "Measurements of the Response of a Panel Excited by shock Boundary-Layer Interaction," Journal of Sound and Vibration, Vol. 16 (3), 1971, 385-391.
- [²²] S. Willems, A. Gulhan, and B. Esser, "Shock Induced Fluid-Structure Interaction on a Flexible Wall in Supersonic Turbulent Flow," Progress in Flight Physics, Vol. 5, 2013, 285-308.
- [²³] D. Daub, S. Willems, and A. Gulhan, "Experimental Results on Shock-Wave/Boundary-Layer Interaction Induced by a Movable Wedge," 8th European Symposium on Aerothermodynamics (89557), 2-6 March 2015, Lisbon, Portugal.
- [²⁴] D. Daub, S. Willems and A. Gulhan, "Experimental results on unsteady shock-wave/boundary-layer interaction induced by an impinging shock," CEAS Space Journal, Vol. 8, 2016, 3-12.

[²⁵] M.R. Gruber and A.S. Nejad, "Development of a Large-Scale Supersonic Combustion Research Facility," 32nd AIAA Aerospace Sciences Meeting & Exhibit, 10-13 January 1994, Reno, Nevada, AIAA Paper 94-0544.

[²⁶] E.H. Dowell and H.M. Voss, "The Effect of a Cavity on Panel Vibration," AIAA Journal, Vol. 1 (2), 1963, 476-477.

[²⁷] A.J. Pretlove, "Free Vibrations of a Rectangular Panel Backed by a Closed Rectangular Cavity," Journal of Sound and Vibration, Vol. 2 (3), 1965, 197-205.

[²⁸] R. Brincker, L. Zhang, P. Andersen, "Modal identification of output-only systems using frequency domain decomposition," Smart Materials and Structures (2001), 10, 441-445.

[²⁹] R. Perez, G. Bartram, T. Beberniss, R. Wiebe, S.M. Spottswood, "Calibration of aero-structural reduced order models using full-field experimental measurements," Mechanical Systems and Signal Processing (2017), 86 Part B, pp. 49-65.

[³⁰] T.J. Beberniss, S.M. Spottswood, D.A. Ehrhardt, R. Perez, "Dynamic response of a Thin Panel Subjected to a Shock Wave Impingement and Thermal Buckling," 33rd AIAA Aerodynamic Measurement Technology and Ground Testing Conference, 5-9 June 2017, Denver, Colorado, AIAA Paper 17-.

[³¹] R. Wiebe, S.M. Spottswood, "On the dimension of complex responses in nonlinear structural vibrations," Journal of Sound and Vibration (2016), 373, pp. 192-204.

[³²] R. Wiebe, S.M. Spottswood, "Co-existing responses and stochastic resonance in post-buckled structures: A combined numerical and experimental study," Journal of Sound and Vibration, Vol. 333, September 2014, pp. 4682-4694.