

The AEDC 10-Deg Cone Wind Tunnel/Flight Laminar-Turbulent Correlation Program

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The AEDC 10-Deg Cone Program stands as a historical tunnel/flight correlation program in which a simple standard calibration body (a sharp, smooth, slender 10-deg included-angle cone) was used for correlation of wind tunnel flow quality and Reynolds number effects in 23 major wind tunnels. The investigation, initiated by the USAF at AEDC, was conducted in the 1970's and included participation by NASA, the Navy, the United Kingdom, France, the Netherlands, and Industry. The flight test was performed at NASA/DFRC (now AFRC) at Edwards AFB on a NASA F-15 A aircraft to Mach 2.0. Testing in the wind tunnels was from Mach 0.2 to 5.5. This program produced an extensive database.

Nomenclature

a	=	angle of attack, deg
β	=	angle of sideslip, deg
C_p	=	pressure coefficient
ΔC_p	=	non-dimensional pressure oscillations, p'_s/q_∞ (where p'_s is composite total rms 200 Hz-20 kHz, %)
F	=	non-dimensional frequency (Strouhal number/unit Reynolds number)
H	=	flight altitude, ft
M	=	Mach number (edge e or free-stream ∞)
p'_s	=	static pressure oscillation (measured on the Cone surface)
p'_t	=	total pressure oscillation (measured by free-stream impact probe, total rms 0 to 20 kHz)
$p_{t\infty}$	=	free-stream total impact pressure, psf
q_∞	=	free-stream dynamic pressure, psf
Q_w	=	local heat flux at the Cone surface wall, BTU/ft ² -sec
Re/ft_∞	=	unit Reynolds number (based on free-stream velocity and dynamic viscosity, U_∞/ν_∞)
Re_t	=	onset of transition Reynolds number ($Re/ft_\infty \cdot x_t$)
Re_T	=	onset of transition Reynolds number ($Re/ft_\infty \cdot x_T$)
$T_{t\infty}$	=	free-stream total temperature, deg R
T_{aw}	=	adiabatic wall recovery temperature on the Cone surface, deg R
T_w	=	Cone surface wall temperature, deg R
x	=	surface length distance from the Cone apex
x_p	=	location of peak rms oscillation amplitude in transition (measured from the Cone apex)
x_t	=	onset of transition location (measured from the Cone apex)
x_T	=	end of transition location (measured from the Cone apex)
1	=	location of forward microphone on the Cone surface ($x_1 = 18.0$ in., $\phi = 180$ deg ray)
2	=	location of aft microphone on the Cone surface ($x_2 = 26.0$ in., $\phi = 225$ deg ray)

I. Introduction

Nearly 50 years ago, a sharp slender 10-deg included-angle cone was fabricated in the Shop at the Arnold Engineering Development Center (now Complex, AEDC) in Tennessee. A nearly identical sharp 10-deg cone had been fabricated earlier. Numerous cones were fabricated at AEDC of various half-angles and tip bluntness for testing, these two specifically as standard calibration bodies for boundary layer laminar-turbulent transition.

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The latter cone was tested in AEDC's transonic and supersonic wind tunnels, NASA's tunnels at Langley (LaRC) and Ames (ARC) Research Centers and Navy and Industry wind tunnels in the US. The earlier cone was tested in four AEDC von Karman Facility (vKF), two NASA Jet Propulsion Laboratory (JPL), one NASA LaRC, and one NASA Lewis (LeRC, now Glenn) supersonic/hypersonic wind tunnels, and in the AEDC 16-Ft Supersonic Propulsion Wind Tunnel by Pate. Both cones had sharp tips (~ 0.004 in. dia.), a mirror smooth surface finish ($\sim 10 \mu$), and 3.0-ft length.

The latter cone, intended for testing in transonic aerodynamic and propulsion and supersonic tunnels, became known simply as the AEDC 10-Deg Cone, a standard calibration body for determining Reynolds number effects of unsteadiness (flow quality) in the test section flow¹⁻⁸. Its utility was extended to subsonic wind tunnels as well.

The earlier cone produced the Pate and Schueler correlation⁹ that wind tunnel test section wall boundary layer noise had a dominant effect on laminar-turbulent boundary layer transition on a cone in nine conventional wind tunnels at Mach numbers from 3 to 8. Both cones utilized a sliding pitot probe in contact with the cones' surface (a Preston tube) to determine the end of transition location x_T and, additionally, onset of transition location, defined as x_t . The AEDC 10-Deg Cone and Pate's Cone were both flight-tested on the nose of a NASA Dryden Flight Research Center F-15A aircraft at Edwards AFB¹⁻³, Pate's Cone as a Facsimile Cone with different supporting sets of instrumentation.

The AEDC 10-Deg Cone Program provided a large, comprehensive data collection on the flow quality (in terms of flow unsteadiness) in 23 major participating wind tunnels⁴, where all kinds of aerodynamic models have been tested. Part of the plan was a flight test in the atmosphere (AEDC 10-Deg Cone in free flight) at varied Mach numbers and unit Reynolds numbers for a tunnel-flight correlation. Pate and Schueler included hollow-cylinder testing as well.

This is a history session paper about the AEDC 10-Deg Cone Tunnel/Flight Correlation Program – its primary results and test techniques, with lessons learned*. **The first lesson learned was to have the benefit of oversight by the National Transition Study Group. Such a Program with international investment and sponsorship needed oversight into plans, methods, and results on a regular-reporting and continuous- interaction basis. The author (Principal Investigator, PI) presented plans and progress to the Study Group under the auspices of the AIAA Fluid Mechanics Panel twice a year for over four years at AIAA national conferences and at NASA/DFRC.**

II. Test Techniques/Methodology

An important consideration in undertaking a large correlation program among wind tunnels and tunnel-to-flight is constancy in the test article configuration, test techniques, and methodology, with assurance of rigor and consistency in measurement system calibrations. To ensure that this took place, the Principal Investigator (PI) traveled with the Cone for testing at the participating locations. This included working with Co-investigator partners at the various sites plus the regular reporting to the Transition Study Group, assuring the investment of all parties. **This was the second lesson learned and employed from the onset of the Program, PI travel with the Cone and having Co-investigators at participating sites to ensure constancy in techniques/methods/verification of the data.**

A. Flight Tests

The flight tests are discussed first as they provided the reference transition and free-stream environment data:

- 1) There were extensive calibrations performed in multiple wind tunnels before flight of parametric sensitivities and of the measurement systems used. There was a Precursor flight test to establish methodology.
- 2) There was the full rigor of certification of the Cones' hardware, instrumentation, and controls for Flight Certification in NASA/DFRC's test chambers – vibration, thermal/vacuum, and functional.
- 3) There was rigorous precursor calibration of aircraft parameters using the AEDC 10-Deg Facsimile Cone.

B. Wind Tunnel Tests

There were repeat entries of the AEDC 10-Deg Cone made in a number of tunnels for parametric studies:

- 1) Configuration of test article ancillary systems and positioning in wind tunnel test sections.
- 2) Calibration of transition sensitivities to wind tunnel flow incidence angle and thermal stabilization.
- 3) Calibration of the fixed Mach meter-altitude-alpha-beta probe Cone measuring system before flight.
- 4) Calibration of transition detection methods – traversing pitot probe versus fixed surface microphones.
- 5) Overlap tests of the AEDC 10-Deg Cone with the (Facsimile) Cone tests in the same vKF tunnel.

* Lessons learned are related herein as first-hand personal recollections of the author, then Program Manager and Principal Investigator for the AEDC 10-Deg Cone Program.

C. Instrumentation

The AEDC 10-Deg Cone is shown in flight on the F-15 A aircraft in Fig 1. It had a rack-and-pinion traverse mechanism for moving the sliding pitot probe in the boundary layer along the Cone surface at 38 in./min. A typical measurement trace from the traversing pitot probe (in continuous contact with the Cone surface) is shown in Fig. 2.



Fig. 1. AEDC 10-Deg Cone In-Flight

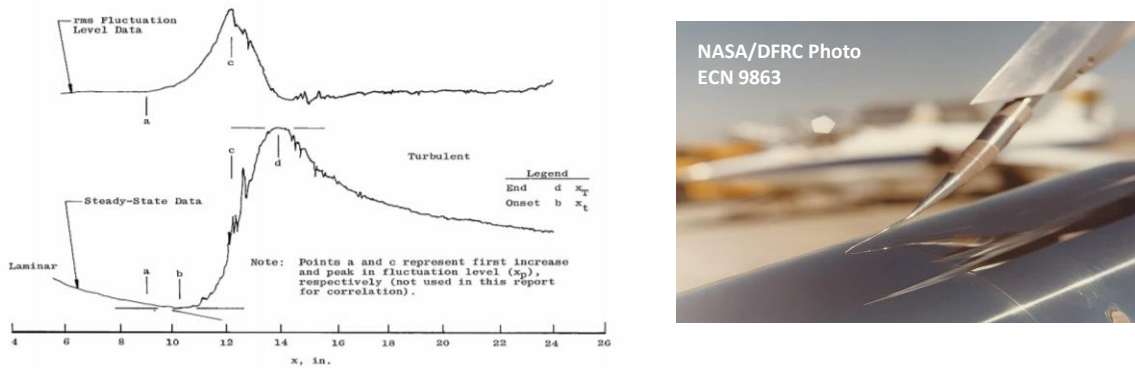


Fig. 2. Definition of Boundary-Layer Transition Location

The traversing probe was a 0.020-in.-inner-diameter hypodermic needle flattened at the tip to an opening height of 0.006 in. A close-coupled 0.094-in.-diam differential semiconductor strain-gage-type pressure transducer was located inside the probe head. The pitot probe signal was split into a low-pass filtered portion (steady-state) and a dynamic portion, which had relatively low frequency response (1 – 50 Hz) and detected the passage of turbulent spots in the transition region. The probe traversed fore and aft along the surface of the Cone on the top-center ray ($\phi = 0$) a distance from the Cone apex of 16.3 in. to 35.3 in. ($x/L = 0.37$ to 0.79 with the clamshell Cone extension in place).

The end of transition location x_T was chosen for correlation. Onset of transition x_t and the location of peak root-mean-square filtered oscillations x_p are noted also. The turbulent spots increased in amplitude and rate of occurrence giving the oscillations peak slightly past mid-way between the departure from the laminar trend at x_t and the fully-developed turbulent trend x_T . Onset of transition was more clearly defined in-flight than in the wind tunnels, and the ratio x_t/x_T was a near-constant $0.85 \pm 5\%$ in-flight, but varied tunnel-to-tunnel. Both the steady and the dynamic Preston tube data were digitized and recorded on a digital data acquisition system at 200 samples/sec and in analog output on an 11 x 17-in. strip chart real-time, along with telemetered free-stream steady-state flow conditions data.

A real-time strip chart trace of an ill-defined traverse profile was cause to reject that data and repeat the test point. Such occasions were rare and occurred in-flight on rare encounters with atmospheric free-stream turbulence. Other occasions occurred in wind tunnels when there was an indication at the time that the Cone was not in thermal equilibrium with the flow-stream at the start of traverse or upon real-time monitoring the surface microphone outputs.

Pressure oscillations on the Cone surface were measured using Bruel and Kjaer 1/4-in. condenser microphones and, in some cases, Kulite 0.094-in. strain-gage-type pressure transducers in wind tunnels and in-flight. The free-stream flow unsteadiness was measured in-flight using an impact pitot total pressure Kulite 0.094-in. transducer on the leading edge of the traverse mechanism (above the Cone, see Fig. 3). Surface microphone data and Kulite transducer data were recorded always with the traverse mechanism fully retracted to its aft-most position. An Endevco tri-axial accelerometer internal to the Cone near its base was used to measure orthogonal-coordinate vibrations of the Cone in wind tunnels and in-flight. These dynamic data were all recorded on a 14-track magnetic tape and analyzed using a Fourier wave analyzer with frequency-response up to 20 kHz, ensemble averaged 36 sec with resolution of 24.4 Hz.

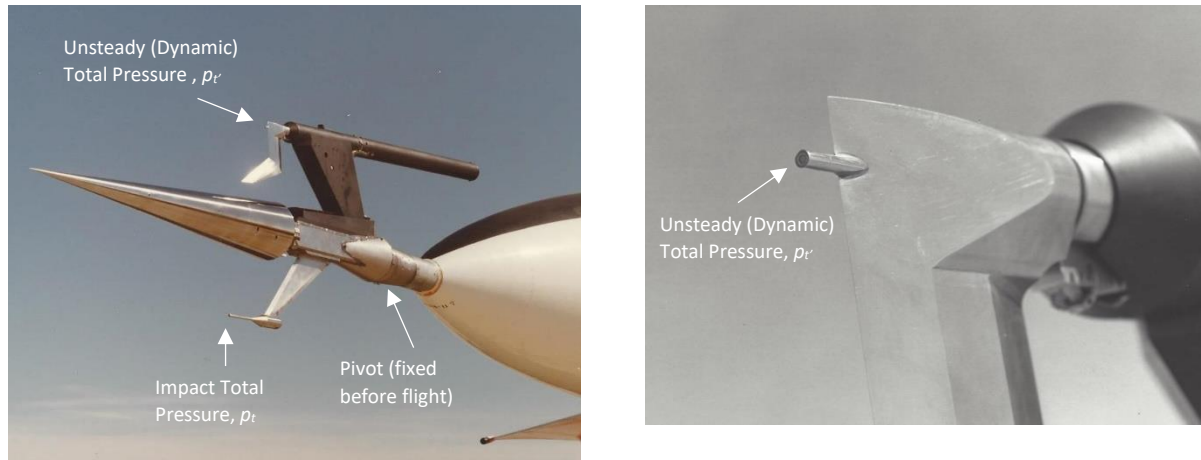


Fig.3. Cone Dynamic Total Pressure, Fixed Probe, and Fixed Pivot

The measurement of Mach number M_∞ , altitude H , and α and β was by means of an AEDC-fabricated five-hole hemispherical head cylindrical fixed probe. The fixed flow-sensing probe contained an impact pressure orifice for p_t and a ring of static pressure orifices 4.7 diameters back on the cylindrical portion of the probe, p_s averaged, for deriving the measurement of airspeed and altitude. The probe required an in-flight calibration to correct for the influence of the aircraft's forward flow field. The Cone was installed seven feet in front of the aircraft's fuselage nose, Cone apex to aircraft nose apex (in place of the normal vane-type α - β nose boom). The data used for the flight position error curves were obtained from the following two methods: (1) Pacer flights (for subsonic Mach numbers) and (2) radar tracking (for subsonic and supersonic Mach numbers). True free-stream Mach number M_∞ , and indirectly, the pressure altitude and ambient, total, and dynamic pressures, were determined by using these position error curves. Boundary layer edge and free-stream M_e and M_∞ , p_e and p_∞ , q_e and q_∞ , Re_e and Re/ft_∞ were predicted using Cone flow Euler computations. Only the two Rosemount total temperature T_t probes on the aircraft fuselage were used of the normal ship's air data measurement system. The second T_t probe (beneath the fuselage) gave more accurate data than the first (on the side of the fuselage) as determined from radiosonde balloon data before flight each day of test.

Two pairs of orifices located 40 deg from the stagnation point of the hemispherical probe head in the pitch and yaw planes were used to measure angle of attack α and angle of sideslip β . The wind tunnel time for calibrations for α and β was provided in the NASA Ames Unitary 11- by 11-Ft Transonic and 9- by 7-Ft Supersonic Wind Tunnels. The Cone was tested at roll angles of 90 and 180 deg as well as at 0 deg using the traversing pitot probe to detect movement of transition location while using the tunnel's pitch mechanism to sweep through pitch angles from 2.0 to -2.0 deg. This technique of using transition movement on the Cone's surface to check the mechanical alignment of the Cone in the wind tunnel, to look for any asymmetries in test section flow, and to calibrate the fixed probe in the proximity flow field beneath the Cone proved satisfactory and was employed in a number of the wind tunnels as well.

The hemisphere-cylinder fixed probe sensors were calibrated probe-alone before its installation on the Cone structure. There were β angle sweeps performed in-flight as the pilot induced rudder trim to vary sideslip from 2.0 to

– 2.0 deg at 0.25-deg increments and holding flight Mach number and altitude and making traverses at each setting, confirming the precision control of the aircraft and the effect on transition that had been measured in the wind tunnels.

The probe-alone calibrations were performed in the AEDC 1-FT Transonic Wind Tunnel. Precision alignment of the fixed probe to the AEDC 10-Deg Cone mounting structure was made in the Propulsion Wind Tunnel (PWT) Shop. The fixed probe was doweled with a pin to hold its position. **The third lesson learned was that the Principal Investigator, present for the fixed probe mounting, might have allowed more time for the doweling operation to be more exacting, as an error was made in the fixed probe's β positioning. The probe manufacture and installation were performed by a highly-skilled AEDC Master Machinist. The resulting β correction factor was larger than desired, but had no deleterious effect, as was proved in β excursion Re_T data in-flight being nearly perfectly symmetrical about indicated 0 deg β .** The α alignment was good. The β position error was confirmed and accounted for in AEDC and NASA/Ames and Langley wind tunnel Cone tests in four wind tunnels performed before flight: in the AEDC 4-Ft Transonic Tunnel, AEDC 40-In. Supersonic Tunnel, NASA/Langley 16-Ft Transonic Dynamics Tunnel, 4-Ft Supersonic Unitary Plan Wind Tunnel, and then in the Cone's Precursor flight calibration tests against the aircraft's flight accelerometer inertial measurements.

The AEDC 10-Deg Cone is shown in the AEDC vKF 40-In. Supersonic Wind Tunnel A in Fig. 4.

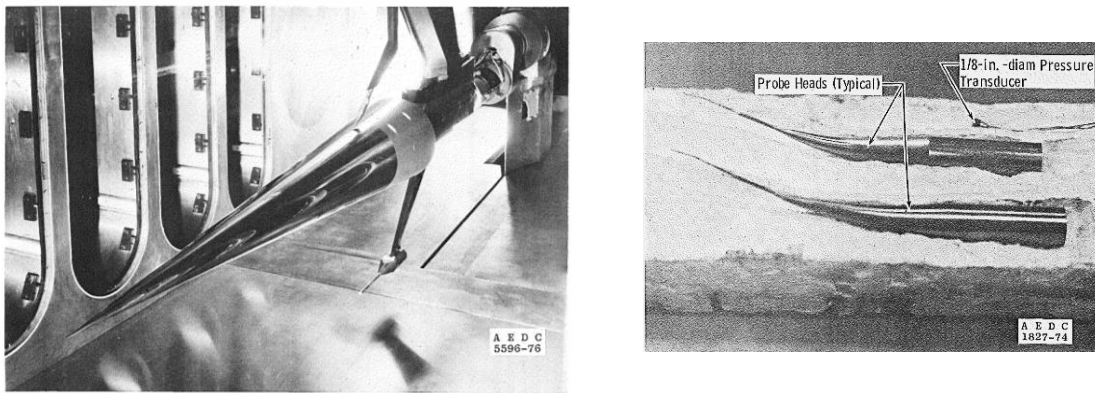


Fig. 4. AEDC 10-Deg Cone in the 40-In. von Karman Facility Tunnel A (M_∞ 1.5 to 5.5)

After computations made from the aircraft's equations of motion confirming the fixed probe's β sensors error, the decision was made to shift the β read-out that fixed amount to compensate for the physical installation error.

The fixed probe's α error determination was based on readings of α on the fixed probe versus the wind tunnel pitch mechanism sting angle of 0 deg pitch and on analytical Euler computations of the Cone flow field at varied Mach number. A Mach number-dependent calibration for the fixed probe's α measurement was developed in the wind tunnels and utilized in-flight, likewise confirmed by the flight accelerometer inertial measurements. These were done in the Precursor flight phase. The physical α positioning correction for misalignment was 0 deg. The conical flow-field proximity correction was a maximum of nearly – 1.4 deg at Mach 1.1 and followed conical flow Euler computation predictions up to and exceeding Mach 2.0. Scatter in α versus M_∞ calibrations in the wind tunnels was ± 0.1 deg, and the NASA Ames data set gave the closest match to the Euler theory, within 0.1 deg above Mach 0.5.

The fixed probe α and β calibrations were therefore assessed to be ± 0.1 deg (with position corrections), and the tests for incidence angle effects on transition on the Cone were conducted with control to as fine as 0.25-deg increments. An 8.5 -in. Cone extension was added to the Cone(s) before flight, and tested in several wind tunnels. Four mutually perpendicular orifices on the Cone extension at $x/L = 0.940$ were manifolded to a precision 13-bit altitude transducer to provide a reference static pressure p_s for the traversing pitot probe, the semiconductor strain gage microphones, and the Facsimile Cone static pressure array. The reference Cone length L was 44.5 in.

The total impact pressure p_t' Kulite transducer was calibrated statically before flight and using the voltage substitution method to calibrate the recorder channel. The condenser microphones (pre-amplifiers inside the Cone limited to 125°F max) on the Cone's surface were calibrated in-place using a pistonphone producing a sine wave at 1.0 kHz of 140 dB ± 0.5 dB sound pressure level (Ref. 0.0002 dynes/cm²). That is 4.16 psf rms. Amplifier gain settings were used in the wind tunnels up an order of magnitude to 160 dB (41.6 psf rms) and down an order of magnitude to 0.416 psf rms. The Kulite microphones had temperature compensation and were accurate from -60 to 200°F.

The pistonphone required a precision fixture apparatus to seal the pistonphone cavity centered over the microphone sensing diaphragm correctly each time and produce an undistorted sine wave shape of the right amplitude. Shock and expansion waves formed over the microphone edges if there was excessive protrusion and vortex formation or if there was excessive recess with respect to the Cone's surface, introducing p_s' errors. The ¼-in. condenser microphones' diaphragm resonance sensitivity to vibrations was the basis for decision to limit the recording bandwidth to 20 kHz.

A modal survey of the aircraft was performed with the AEDC 10-Deg Cone installed as configured for flight. The Cone/aircraft fuselage bending natural frequencies were found to be 5.6 Hz in the lateral direction and 7.0 Hz in the vertical (pitch) direction during the Ground vibration test. The only significant in-flight vibrations were found by the Cone's accelerometers to lie below 200 Hz. The Cone microphone data were band-pass filtered from 200 Hz to 20 kHz. The free-stream total impact probe measurement was recorded from 0 to 20 kHz.

The AEDC 10-Deg Cone is shown in the AEDC 16-Ft Transonic Propulsion Wind Tunnel is Fig. 5 during microphone calibration. The microphone installation details were as shown in Fig 6 for tunnels and flight.

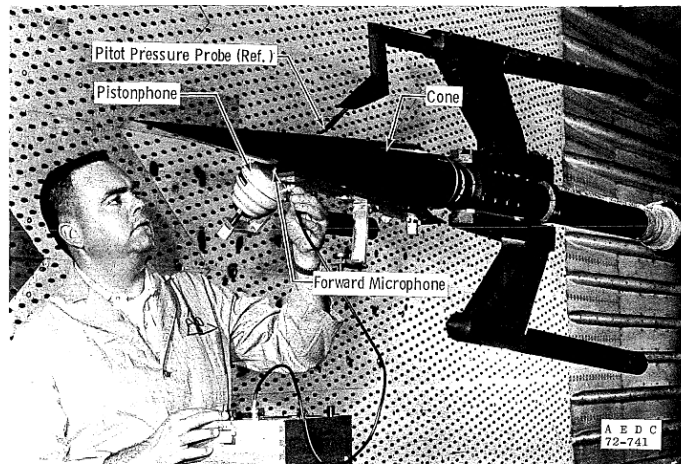


Fig. 5. The AEDC 10-Deg Cone in the AEDC 16-Ft Transonic Wind Tunnel

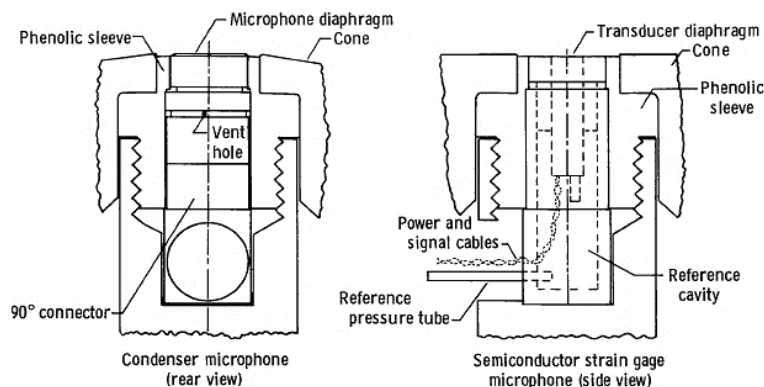


Fig. 6. Flush-Mount Installation of Microphones and Transducers on the Cone Surface

The ancillary structure downstream of the Cone used in earlier wind tunnel tests included a dummy strut (symmetrical with the traversing pitot probe strut) to ensure flow symmetry at subsonic Mach numbers. In planning for the flight testing, wind tunnel testing with the fixed probe in repeat entries in tunnels where the dummy strut had been used before proved that the dummy strut was not essential. Subsonic/transonic flow asymmetry without it would be small (almost negligible), and ability to have the fixed probe air data measurement system was essential as the Cone installation would render the ship's Mach meter-altimeter- α - β air data system in error. At supersonic Mach numbers, the Cone flow field would be conical flow all the way to the shoulder expansion at its base, $x/L = 44.5$ in.

The Cone body after 36.0 inches was a cylinder of slightly smaller diameter terminating in a blunt base. It was decided for the flight tests that a clamshell Cone extension of 8.5 in. length would be added to lessen the favorable pressure gradient on the Cone's aft portion of pitot probe travel (to 35.3 in.). In the wind tunnels at subsonic Mach numbers, the pressure gradient effect of the flow expansion aft of the Cone would be small as transition was located forward of the aft microphone ($x_2 = 26.0$ in.) at most Re/ft test points except for M_∞ approaching 2.0 in-flight.

D. Cone Flow Field as-mounted on the Aircraft

The Cone flow field as mounted on the aircraft was assessed using Cone/aircraft Euler computations, which formed the basis for decision to put the Cone seven feet in front of the aircraft's nose, to meet testing requirements in-flight at subsonic Mach numbers. An additional consideration was that a fixed pivot for the nose boom sting mount for the Cone would be needed, pre-set before flight at an angle compensating for the aircraft's trim angle of attack, which varies with q_∞ at Mach number and altitude. The flight test technique would be to set the pivot angle for a series of flight test points planned for a given day and to fly those points at constant trim angle of attack (constant q_∞), another pivot angle for test points another day at another trim angle of attack, so that the Cone would be at 0 deg angle of attack with the airflow. Small α error was accounted for by the pilot as the aircraft burned off fuel.

The Precursor flight test phase was planned with a surface static pressure array p_{se} of measurements on the 0-deg ray of the Facsimile Cone (same as the pitot probe traverse on the transition Cone) and a circumferential static pressure array p_{se} around the Facsimile Cone (90, 180, and 270 deg ϕ). The Cone surface pressure data acquired was compared to the Euler predictions⁸, and the agreement was judged to be satisfactory. The slightly-favorable p_{se} axial pressure gradient for the Cone-alone was lessened to near zero pressure gradient in-flight at subsonic flight Mach numbers. The supersonic p_{se} pressures measured at Mach numbers flown agreed with conical flow predictions from J. C. Adams. Subsonic and supersonic pressure data measured on the Facsimile Cone surface are shown in Fig 7.

The Precursor flight tests included use of Gardon heat transfer gages on the Facsimile Cone wall in place of the pressure transducers.

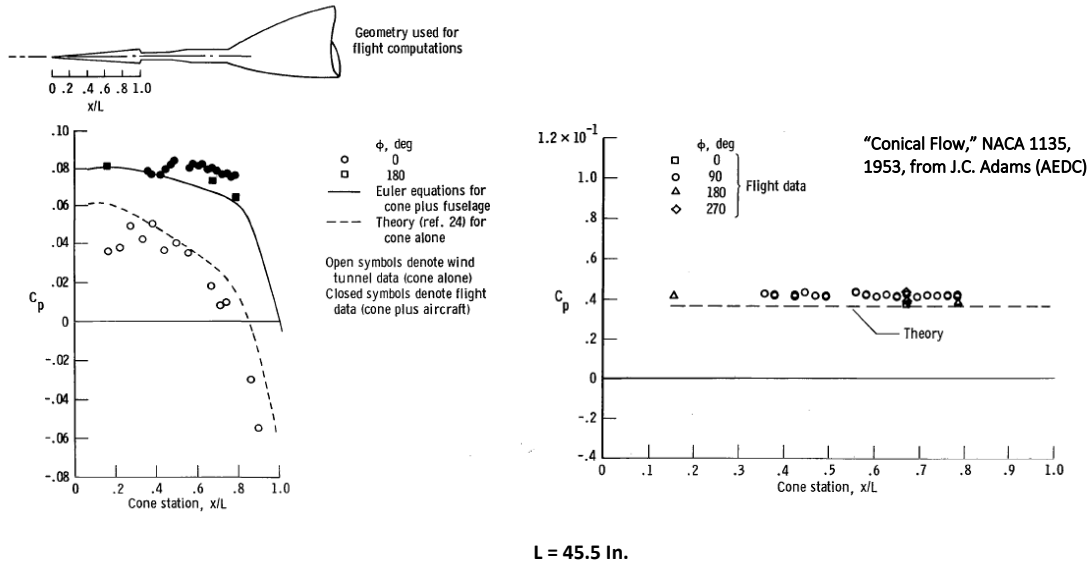


Fig.7. Precursor Flight Test Pressure Distribution on the Cone at Mach 0.6 and at Mach 1.7

The thermal sensors included both Gardon heat flux gages and thermocouples. The heat flux gages were 0.2 millimeter (0.008 inch) thick and 6.35 millimeters (0.25 inch) in diameter, specially fabricated in the vKF Instrument Lab by C. T. Kidd. Procedures were established for the flight test based on the Precursor Facsimile Cone data acquired.

During the flight tests it was possible to control the Transition Cone's temperature within ± 6 % of the adiabatic wall temperature, T_{aw} , for about 90 % of the test points using these procedures. This deviation in temperature had a large influence on transition location, however, as shown in Fig. 8. The Re_T data have been nondimensionalized by the adiabatic wall temperature, Re_{T0} denoting data at zero deg Cone incidence angle. The Re_T sensitivity to heat transfer appears to have been essentially independent of Mach number and strongly dependent upon T_w/T_{aw} . The Precursor flight data shows heat transfer influence on transition to be approx. to the -7^{th} power law, delayed transition occurring

when the boundary layer was cooled ($T_w/T_{aw} < 1.0$), earlier transition occurring when the boundary layer was heated ($T_w/T_{aw} > 1.0$). Heat transfer Q_w was 0.3 BTU/ft²-sec at $T_w/T_{aw} = 0.97$, -0.4 at $T_w/T_{aw} = 1.08$.

Each Cone transition data point in-flight was taken after waiting for the Cone to settle near $T_w/T_{aw} = 1.0$ (thermal equilibrium), watching the Cone surface and body thermocouples in real time T_w approach T_{aw} . The flight test technique employed was to pre-heat the Transition Cone to 220-240 °F before flight at the end of the runway using a hot-air blower cart, then launch immediately as the Cone began to cool, and to lay out a succession of target flight data point conditions based on radiosonde balloon data taken each day before flight. The test points were aligned on a logical sequence for the Cone's cooling over time at each trim angle of attack, and available fuel and test time.

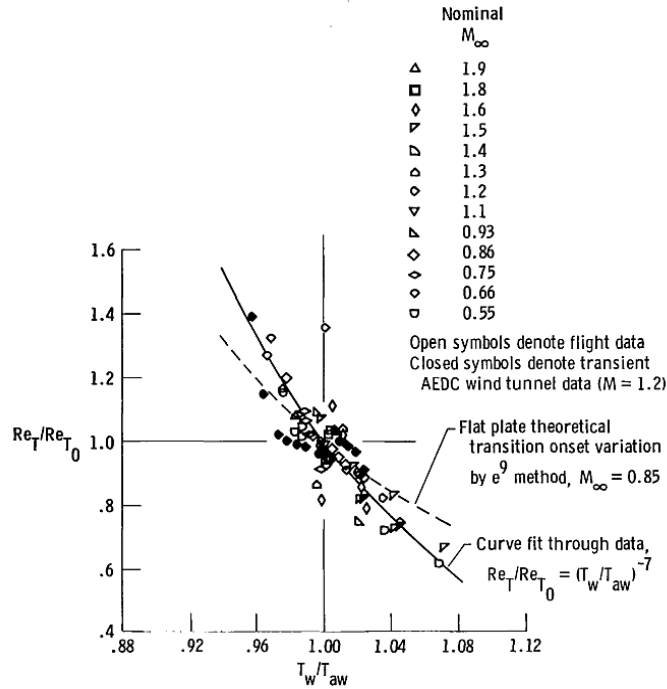


Fig. 8. Effect of Heat Transfer measured on the Facsimile Cone in Precursor Flights

Reference heat transfer effect on transition data shown were taken in a test of the AEDC 10-Deg Transition Cone in AEDC's 4-FT Transonic Wind Tunnel. It was possible to make a relatively rapid change in the wind tunnel free-stream total temperature T_t at constant p_t and Mach number and measure the change in transition location on the Cone as it was heated and cooled as in-flight (bias being toward heating the boundary layer, $T_w/T_{aw} \geq 1.0$, in-flight).

Generally, in all the wind tunnel testing, total temperature T_t was held near constant based on the facility circuit cooler settings while varying total pressure p_t to obtain variable density across the tunnel Mach number range. The Cone wall and body temperatures were monitored in real time to wait for the Cone to reach thermal equilibrium with the free-stream at each test point. There were a few occasions where it was noticed that the Cone was not in thermal equilibrium and test points were repeated with the attempt to obtain equilibrium.

The strong effects on transition with the Cone at non-zero incidence angle or not in thermal equilibrium were known early in the wind tunnel testing before flight. Therefore, steps in test techniques were taken and concurred with by the oversight group to minimize those effects and make the correlation at as close as possible at zero incidence angle and $T_w/T_{aw} = 1.0$. **This fourth lesson learned was considered essential to the Program's success and guided test conduct throughout the entire wind tunnel/flight Program, to correlate Re_T specifically at zero incidence and adiabatic wall conditions, exercising the tightest practical controls over these variables.**

Calculation of T_{aw} utilized the turbulent boundary layer adiabatic wall recovery factor $r = 0.88$ as the Cone's control thermocouples were located aft of the transition location in most all cases. Surface heat transfer in high supersonic-hypersonic laminar-turbulent transition is so high as to provide a good means of detecting transition location. However, in the flight tests to Mach 2.0 of this Program, the heat transfer was deliberately controlled to be small at data points, and the Gardon gages did not show enough sensitivity to be utilized for transition detection.

E. Transonic and Supersonic Wind Tunnel Configuration Factors

The AEDC 10-Deg Cone is shown installed in the NASA Langley 16-Ft Transonic Tunnel, NASA Ames 11-FT Transonic Tunnel, Naval Ship Research and Development 7- X 10-Ft Transonic Tunnel, and RAE Farnborough 8- X 6-Ft Transonic Tunnel in Fig. 9. Transonic wind tunnels have plenum chambers surrounding the test section with ventilated walls and flow re-entry flaps at the diffuser entrance to be able to operate slightly above Mach 1.0 and cancel the strength of the shock waves that impinge back on a model at low supersonic Mach numbers.

The transonic wind tunnels are distinguished in the AEDC 10-Deg Cone Program by test section wall type, although drive compressor fans, coolers, corner turning vanes, dryers, stilling chamber screens and honeycomb, and nozzle throat contraction ratio are important factors as well.

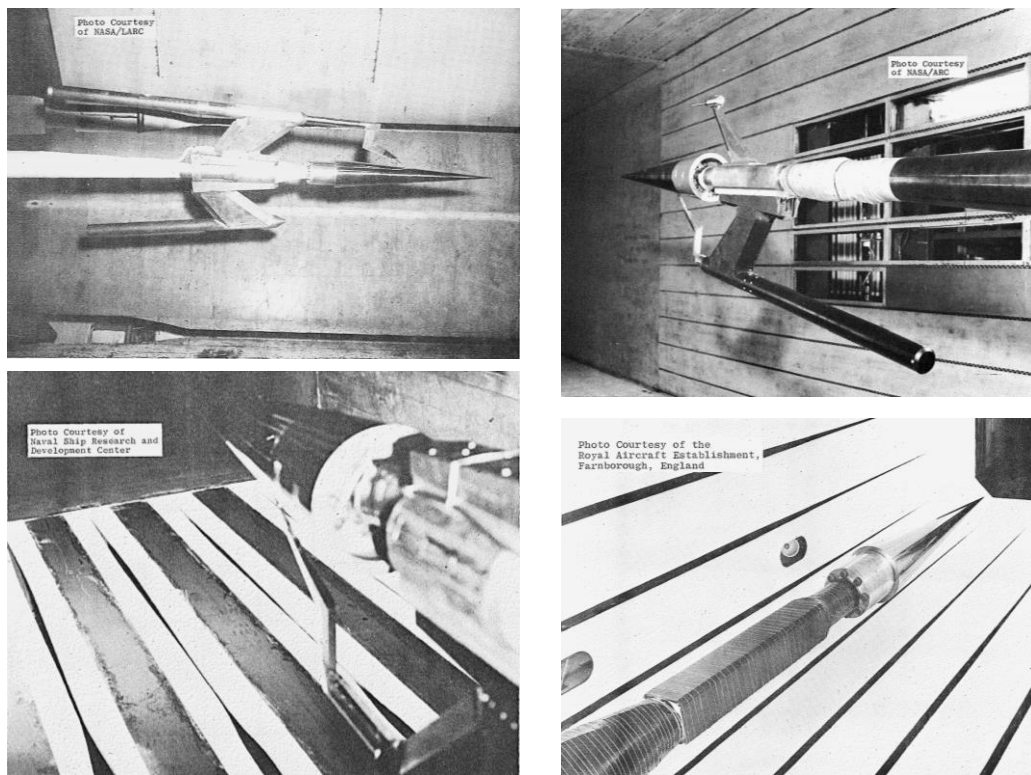


Fig. 9 AEDC 10-Deg Cone installed in Four Major Transonic Wind Tunnels

The Navy's Transonic Tunnel (for testing aircraft models) and RAE Transonic Tunnel in Farnborough have slotted walls of nearly similar shape as those of the Langley 16-Ft (shown here), 16-Ft Transonic Dynamics (Flutter) Tunnel, and 8-Ft Transonic Pressure Tunnel. These are distinguished from the 6-% perforated-walls with 60-deg inclined elliptical holes of the AEDC 16-Ft and 4-Ft Transonic Tunnels and the NASA Ames 11-Ft Transonic Tunnel, which has constant-width slots with Z-shaped baffles. A perforated-wall configuration, used in the Calspan 8-Ft and ARA, Ltd. 9- X 8-Ft Tunnels, is a 22.5 % perforation pattern of normal circular holes.

The transonic wind tunnels, like the supersonic wind tunnels, have a second throat at the diffuser entrance and near-constant Mach number along the test section length. Subsonic wind tunnels (with solid walls) like the NASA Ames 12-Ft Pressure Tunnel and RAE Bedford 8-Ft Tunnel operate only below Mach 1.0, or in the case of the large RAE tunnel in Bedford, in a supersonic mode above Mach 1.4 as well.

F. Program Anomalies

All the tests in Europe were performed without the traversing pitot probe. The test in the RAE Farnborough Tunnel utilized a fixed pitot probe attached to the Cone's surface at a single particular x location on a ray opposite from the Cone's forward microphone, $x_1 = 18.0$ in. The Cone tests in Europe all used the microphones to detect transition location and the Farnborough test a comparison with the fixed pitot probe as well.

Wind tunnel dryer effectiveness is important. Calibrations performed in many of the wind tunnels for the effect of humidity, varying dew point temperature and introducing visible fogging, generally have shown little or no effect on test results. However, during repeat testing of the AEDC 10-Deg against Pate's vKF Cone (the Facsimile Cone) there was consistently-low Re/ft at low free-stream supersonic Mach numbers and low density (high altitude) settings when visible fogging was present in vKF Tunnel A. The Tunnel control settings did not agree with the fixed flow probe Cone data for Mach number, total pressure, and total temperature and derived M_∞ and Re/ft_∞ .

The Tunnel A wedge probe, which dropped down into the flow stream to take flow conditions data and retracted back flush with the wall, was experiencing strong condensation shocks due to a continuous very strong rain storm that went on for days and nights over that particular period of testing. For very low sub-atmospheric pressure testing, in-leakage into the circuit at the time overcame the dryer capacity and produced excessive humidity in the test section. The Mach number and Re/ft data were corrected after test, and the agreement in Tunnel control set conditions and the Cone's fixed flow probe came into close agreement. And the Cone boundary layer Re_T data came into good agreement with Pate's data from years before. These were very heavy rain storms that produced several inches of rain in three days during this test, and this was a testing anomaly recognized and fixed.

The flight testing had virtually no humidity influence effects in the arid atmosphere above Edwards AFB, Rogers Dry Lake, and the supersonic corridor. A general rule was to invoke test pilot's prerogative to avoid flight in clouds or in regions of atmospheric turbulence. There were virtually no clouds of consequence and no inclement weather.

There were two other Program anomalies over the eight years of testing⁴. Both affected the tests in Europe. One was that upon initial opening of the AEDC 10-Deg Cone's shipping box (a special felt-lined mahogany box), the Cone had preceded the Principal Investigator's arrival, there were transverse scratches present approx. 4 in. from the apex where the Cone tip protective cover was lightly attached to the Cone surface with cloth tape. It appeared as if a sharp metallic object might have been used to pry the tape loose and had scratched the Cone across the 0-deg ray.

This was at the start of the European test campaign. Application of jeweler's rouge and hand polishing could not remove the blemish in a reasonable period of time, and it was decided to use the Cone's two surface microphones for tests in the seven tunnels in England, France, and the Netherlands. A localized roughness element can induce laminar-turbulent transition at the spot, and there is an expected transition location 17-deg wedge that can spread behind it, if the local roughness height has a critical ratio with the boundary layer thickness δ . But the spreading would not be around the Cone's circumference enough to affect laminar-turbulent transition at either the forward or the aft microphone locations. The method of operation turned out to have been completely effective in all seven Tunnel entries in Europe. The blemish was later polished out, taking the necessary time upon the Cone's return to the US.

The second of the two anomalies occurred after the test in the RAE 8-Ft Tunnel had been completed and inspections were being made late into the evening of the Cone tip in the ARA, Ltd. Tunnel 8- X 7-Ft Tunnel in Bedford. The Principal Investigator was performing the inspection, standing on a stepladder, and momentarily lost balance and lurched backward (without falling), but raking a metallic instrument across the Cone's tip. The Cone's tip was bent slightly. The Principal Investigator then took two jeweler's flat blocks and carefully straightened the tip by hand. It turns out that bending of the tip was only slight, and had no discernible deleterious effect on the correlation results obtained in Europe. The Cone's tip was straightened then, but not exactly to within the ± 0.01 deg manufacturing tolerance of the Cone. It was straightened afterwards, when the Cone was received back at AEDC.

These two anomalies to the Cone's configuration were minor – the **fifth Lesson Learned, that written Program instructions from the Principal Investigator to all organizations, including Travel, coming into proximity so as to touch the Cone should be much more stringent and specific to assure the Cone's physical integrity.**

Photographs before and after completing this work are shown in Fig. 10. There had been a local violation of the ± 0.01 deg conical surface design requirement. It was restored to within ± 0.01 deg for the balance of the Test Program.

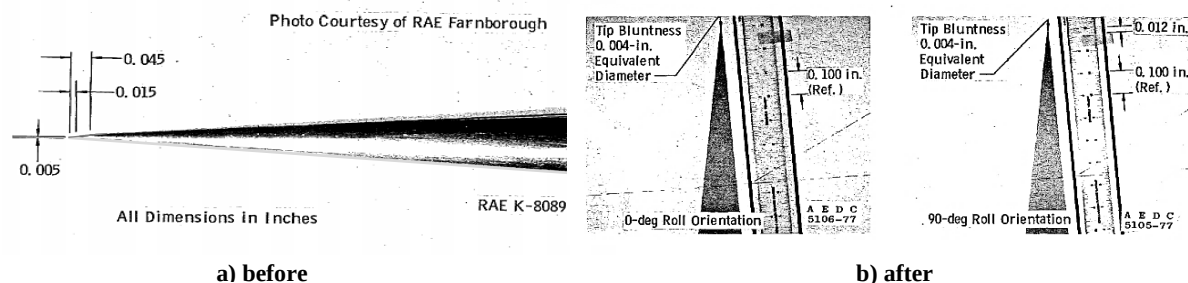


Fig. 10. Restoration of the Bent Nose Anomaly as Seen on the Optical Comparator

Based on the tunnel and flight data acquired thereafter, the deviation in the Cone's configuration was deemed acceptable for those tests affected, and the decision to proceed that was made was borne out to have been a good decision. There was no loss of sharpness of the Cone tip, surface finish was maintained near the tip, and the surface was restored to within $\pm 0.01^\circ$.

Restoration of the Cone – complete polishing out of the blemish with jeweler's rouge and final straightening of the tip using a jeweler's block – were accomplished making use of the optical comparator (precision magnifier) at AEDC. Not so much as a fingerprint had been allowed the entire eight years on the Cone's mirror-finish surface or on the Cone microphone diaphragms. Not a single insect strike was seen on the Cone in the wind tunnel or flight tests.

G. Flight Test Data Point Matrix As-Flown

The Mach number-altitude flight test matrix is shown in Fig 11. The test pilot's control was to Cone zero-incidence and adiabatic wall temperature upon arriving at a given test point for a given trim angle of attack. There was therefore a variation in trend for nominal q_∞ versus Mach number and altitude. Some variation was expected as the flight tests occurred over a years' time with repeat points sometimes months apart showing seasonal variation (adiabatic wall temperature T_{aw} for the Cone based on static temperature at altitude measured by the balloon that day).

The nose boom pivot angles were from -4.5 to -0.75° relative to the aircraft's centerline. The variation in Re/ft_∞ from approximately 1.5 to 4.5×10^6 is the same as was tested in the wind tunnels.

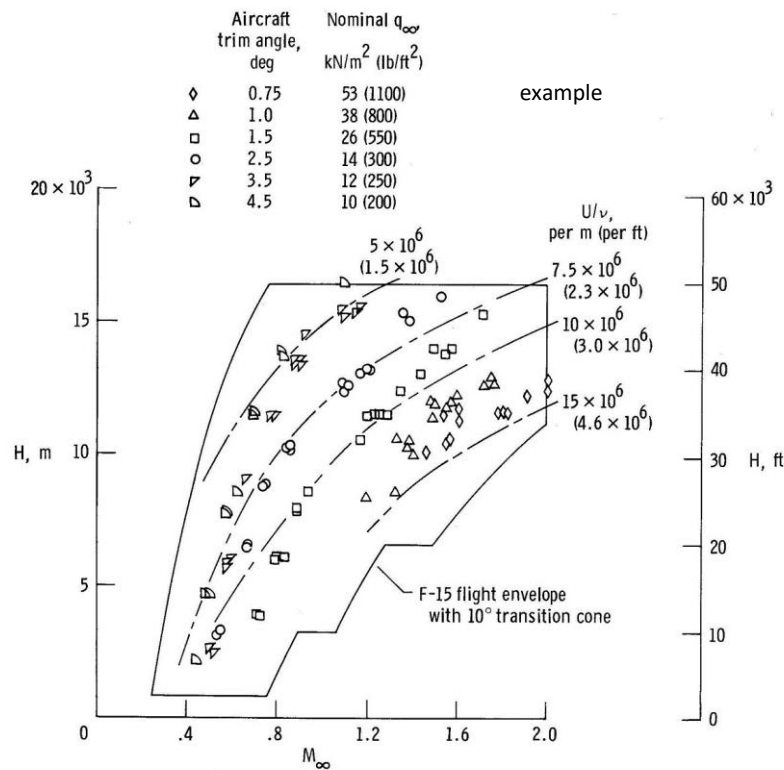


Fig. 11. Flight Test Data Matrix

The Precursor flight tests with the Facsimile Cone were flown in envelope expansion with the mounted Cone configuration to the corners of the flight envelope and across the range of Re/ft to verify the operating procedures. The pilot's ceiling per operating rules was 50,000 ft. Laminar-turbulent transition onset x_t was off the Cone at Mach 2.0 and 37,000 ft. The median Re/ft extending through the supersonic-hypersonic range in the wind tunnels was 3.0×10^6 .

The adiabatic wall recovery temperature T_{aw} on the Cone at the Mach 2.0 test points was as high as 640 °R (120 °F). A single data point could be acquired at Mach 2.0 on a given day, the total flight time being ~11 minutes duration, going to afterburner to accelerate once in the supersonic corridor to steady flight speed and altitude on conditions, winding up back over the runway to circle for landing on the remaining fuel.

The atmospheric static temperature T_∞ follows a lapse extending to 30,000 ft, sometimes higher, and a decrease in the rate of the lapse at higher altitudes. There was therefore an appreciable variation in the target Cone T_{aw} determined by the radiosonde balloon data due to normal changes in the atmosphere that occur day-to-day.

III. Correlation Results

In the low-disturbance environment of flight in the atmosphere, 10^{-10} down to 10^{-12} in mean square static pressure per analysis bandwidth (24.4 Hz) when normalized by q_∞^2 , there was an amplified band of frequencies seen on the Cone microphones when the boundary layer was laminar¹. The center frequency (most-amplified frequency F) is indicated in normalized power spectrum $G(f)$ plots such as shown here in Fig. 12 at Mach number 1.35 as much as two orders of magnitude above the free-stream disturbance p'^2 level but which at lower frequencies p'^2 peaks to nearly 10^{-8} . Plus, there are turbulent spots present, giving rise to low-frequency disturbances seen on the aft microphone at x_2 . Low-speed experiments by Schubauer and Skramstad¹⁰ and by Saric¹¹ and others had measured transition, setting the incompressible reference. Leslie Mack developed a compressible linear stability theory¹² for predicting transition Reynolds number under subsonic, transonic, and supersonic conditions.

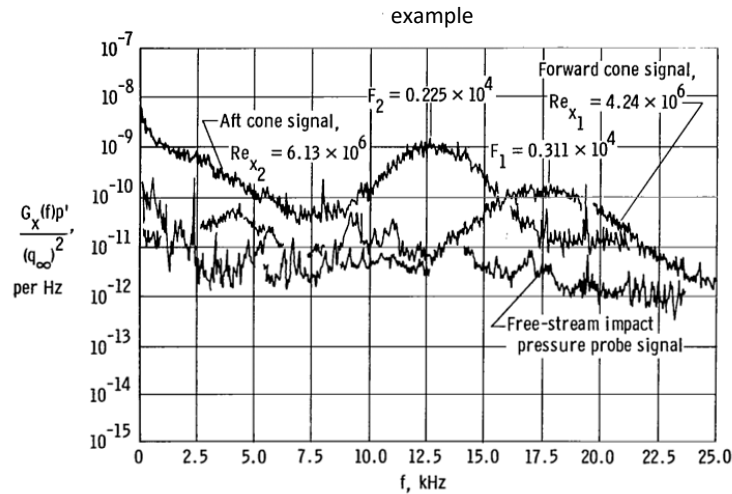


Fig. 12. Typical Free-stream Impact and Cone Surface Spectral Data ($M_e = 1.35$)

Les Mack was asked by the Principal Investigator to come up from Caltech to Edwards AFB to review these data. The variation in F with Mach number, and secondarily with q_∞ and Re/ft_{x_2} , was a function of the length Re_x of the respective Cone microphone x_1 and x_2 . Mack confirmed that the flight microphone data generally matched his linear stability theory (e^N method, where N is the growth amplitude exponent of disturbances in the laminar boundary for transition) with the presence on the Cone of Tollmien-Schlichting waves (the first one-dimensional mode of instability) in plots like Fig. 12 as an example, shown to him across the 0.5-to-2.0 Mach number range of the flight test.

Mack made the pronouncement from his inspection of the data. Other researchers since have made use of these data and have extended both computational methods and measurements down to very low speed and up to the hypersonic regime. Mack's linear stability theory had predicted cross-flow (oblique) modes and higher-frequency modes to become dominant at high Mach numbers above ≈ 1.3 into the supersonic and hypersonic regimes. Experimenters made measurements to much higher ranges of frequency than 20 kHz and used hot-wire and hot-film anemometers and pitot probe impact total unsteady pressure measurements to better define velocity fluctuations and pressure fluctuations to characterize test environments and surface fluctuations growth of the higher modes at higher Mach numbers. Mack's oblique (cross-flow) and axial second mode of instability were at higher frequencies.

This was the **sixth Lesson Learned and applied by the Principal Investigator, to ask Les Mack to make the independent review and pronouncement about the transition data in the flight environment and gain that validation.** Les Mack and Jim Kendall both came to DFRC immediately upon request and reviewed the data.

The wind tunnel test section environments, as indicated by the integrated full measured bandwidth pressure fluctuations on the Cone microphones under laminar boundary layer flow⁴, were an order of magnitude or sometimes greater in amplitude, except in one wind tunnel, the NASA Langley 4-Ft Supersonic Pressure Tunnel, where ΔC_p was as low as 0.08 %, at Mach 2.0 at Re/ft of 3.0×10^6 . (ΔC_p is the integration of the square root of the time-averaged mean square pressure $(p'^2)^{1/2}$ fluctuation across the entire recording spectrum to 20 kHz, normalized by q_∞ , x 100, %).

End of transition Reynolds number Re_T data measured on the Cone in selected wind tunnels and flight is shown in Fig. 13; Re_T increases with Mach number and has a secondary variation with q_∞ , or Re/ft_∞ . The flight data are compared here to data from several of the wind tunnels having lowest ΔC_p . Wind tunnel data at lowest subsonic Mach numbers shown were taken at q_∞ lower than the 200-300 psf of the flight data at lowest Mach numbers.

The several wind tunnels in Fig. 13 having Cone Re_T nearly as high as in the flight test had low disturbance levels (ΔC_p) operating at subsonic Mach numbers, but nearly all too high to confirm Tollmien-Schlichting waves present. These were the NASA Langley 8-Ft Transonic Pressure Tunnel, 16-Ft Transonic Tunnel, and Transonic Dynamics Tunnel, the Naval Ship 7- X 10-Ft Tunnel, the NASA Ames 12-Ft Pressure Tunnel, the RAE Bedford 8- X 8-Ft Tunnel, the NASA Langley 4-Ft Supersonic Pressure Tunnel, the AEDC 40-In. vKF Tunnel A, and the NASA Ames 9- X 7-Ft Tunnel. Among these, facility provisions were present to control free-stream turbulence (velocity oscillations) and acoustics. There were the two exceptions – the NASA LaRC 4-Ft SPT and 16 TDT.

Only the NASA Langley 4-Ft Tunnel at $M_\infty = 1.6 - 2.0$ ($M_e = 1.5 - 1.8$) and, marginally, 16-Ft Transonic Dynamics Tunnel, had low enough background unsteadiness spectra $G_x(f)/q_\infty \approx 10^{-9}$ or less to make out laminar boundary-layer wave growth and that at power spectrum levels at amplified frequencies well below the mean pressure squared at low frequencies in those tunnels. There was marginally-low ΔC_p at high frequencies in the NASA/Langley 16-Ft Transonic Dynamics Tunnel that appeared to indicate Tollmien-Schlichting wave growth, only after the benefit of having the Cone flight data in hand. This was at $M_\infty = 0.85$ and not evident at any other Mach number in that test. All the other wind tunnels had unsteadiness p_s' (as measured by the Cone surface microphones) and transition (affected by what is broadly termed here as by-pass mechanisms), Tollmien-Schlichting waves not observed in the mean-squared spectra.

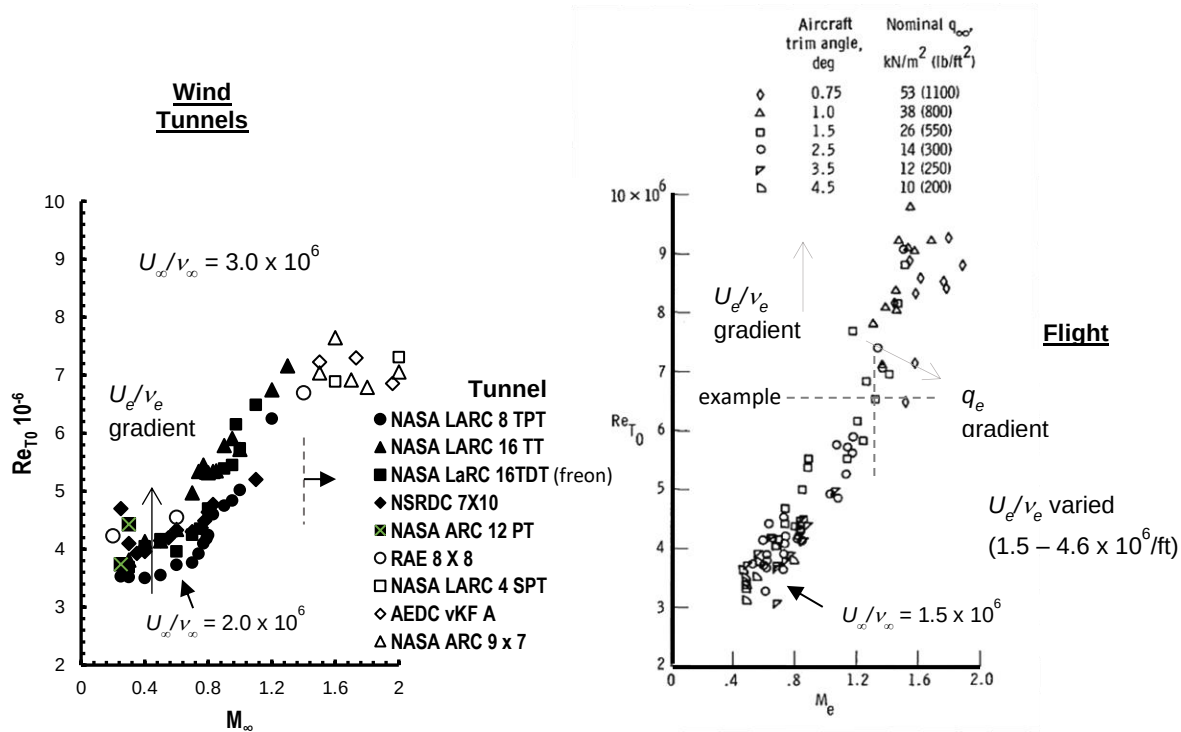


Fig. 13. Transition Reynolds Numbers in Several Wind Tunnels and Flight

Although the wind tunnel disturbance ΔC_p levels were considerably higher overall, the Re_T data in these wind tunnels up to $M_\infty \approx 1.6$ were reasonably close to the flight Re_T data (corrected to zero incidence and adiabatic wall). Operations in these nine cited wind tunnels showed data scatter tunnel-to-tunnel about the same as flight at varied q_∞ (altitude at a given Mach number). (A notation is plotting herein as a function of M_e here instead of M_∞ .)

The transonic tunnels have suction applied to the ventilated walls (closed symbols). The solid-wall subsonic and supersonic wind tunnels are shown with open symbols. Transonic wind tunnels with wall suction slots and some convergence added to the flexible nozzle upstream can reach a maximum of 1.3-1.6 M_∞ . Subsonic wind tunnels can operate only below Mach 1.0. Supersonic wind tunnels (with a converging-diverging nozzle and a second throat downstream) operate at approx. 1.4 lowest Mach number. Divergence in Re_T from flight begins at $M_e \approx 1.4$.

The most-amplified frequencies (Tollmien-Schlichting waves) in flight varied linearly as a function of Mach number at constant q_∞ for each microphone on the Cone surface and varied inversely with the microphone x locations. Typical data correlating the most-amplified frequencies f are shown in Figs. 14 and 15 where in Fig. 15 frequencies have been normalized to F by Strouhal number/unit Reynolds number using the edge velocity U_e and viscosity ν_e .

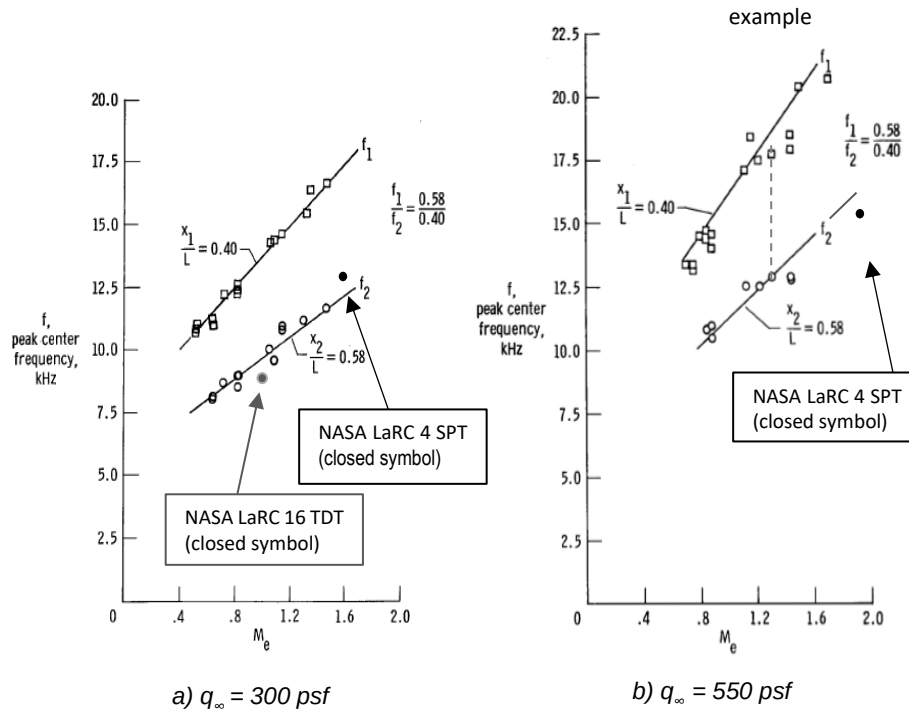


Figure 14. Most-Amplified Unstable Frequency In-Flight

Mack's normalization of the most-amplified frequency in the unstable frequency band¹² (Tollmien-Schlichting waves) is the Strouhal number S divided by the unit Reynolds number, where ν_e is the dynamic viscosity μ_e/ρ_e .

$$F = (2\pi f/U_e) \cdot (\nu_e/U_e) \quad \text{for } M_e = 0.5 \text{ to } 2.0 \text{ (flight)} \quad (1)$$

The non-dimensional frequency F is proportion to $(Re_x)^{1/2}$ on the Cone surface (x length from the Cone apex to each microphone) and secondarily to M_e , or M_∞ .

According to the theory (1969), the first mode, with wave angle $\psi = 0$ deg (one-dimensional), has reduced amplification by some three orders of magnitude in going from low subsonic to Mach 2. Then from approx. Mach 2 to approx. Mach 4, an oblique first mode becomes the most amplified at $\psi = 60$ deg. Above approx. Mach 4, the high-frequency second instability mode, at $\psi = 0$ deg (one-dimensional), becomes the most amplified for transition. This

was much the same as the measured data trend of the transition data on the AEDC 10-Deg Cone. Mack represented the expression for the amplification rate of the oblique first mode most-amplified frequency at $\psi = 60$ deg as

$$A_o/A_r \sim (M_1/1.3)^2 \quad \text{for } M_e > 1.3 \quad (2)$$

where the subscript 1 here refers to the first (one-dimensional instability) mode, o is the amplification of the oblique mode, r refers to the reference amplification, and $M_e = 1.3$ is the reference Mach number.

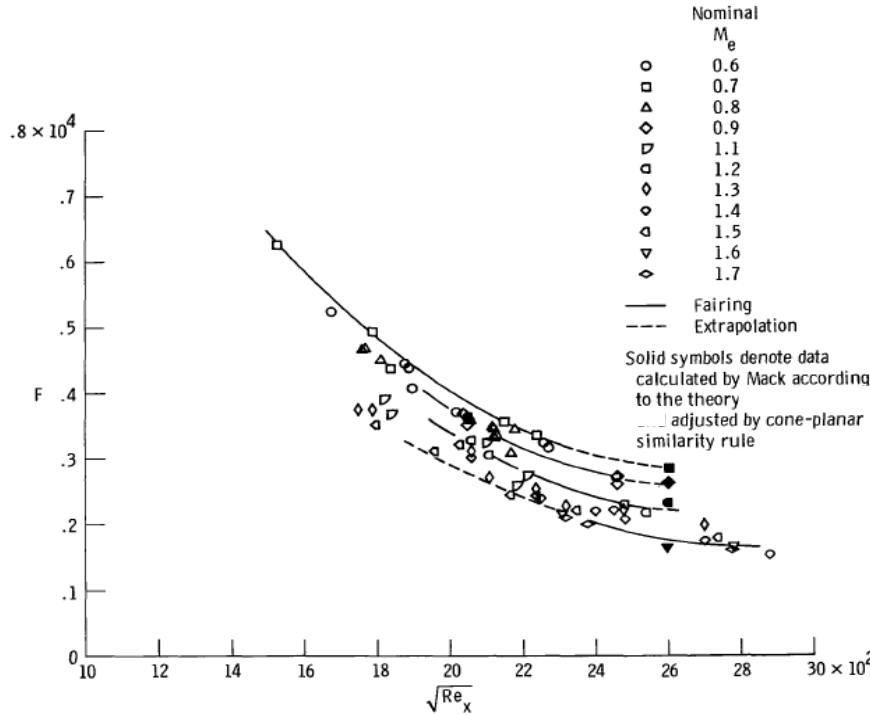


Fig.15. Non-dimensional Most-Amplified Unstable Frequency on the Cone

The disturbance levels recorded by the free-stream impact probe and on the Cone surface in-flight are shown in Fig. 16. They are the square root of the mean square averaged overall integrated spectrum to 20 kHz levels normalized by q_∞ . Static pressure fluctuations on the Cone surface agree well with the total pressure impact fluctuations in the free-stream except for the influence of laminar unstable waves growth and, in some cases, the presence of turbulent spots. The condenser microphones and the semiconductor strain gage Kulites gave very comparable readings; but the Kulites were able to function at higher temperatures and were utilized at the higher Mach numbers. The preponderance of the wind tunnel data was with the condenser microphones, some with the Kulites.

The Cone microphones measured higher fluctuation levels after transition to turbulence, reflecting the disturbances in the local turbulent boundary layer passing over the microphones at $x_1 = 18$ in. and $x_2 = 26$ in. Some examples of this are the data at $M_\infty = 2.0$ in the NASA Langley 4-Ft Supersonic Pressure Tunnel and at $M_\infty = 1.4$ in the RAE 8- X 8-Ft Tunnel at Bedford in Fig.16. Other examples are the data from the NASA Ames 9- X 7-Ft Supersonic Tunnel in at Mach numbers from 1.5 to 2.5 at 3.0×10^6 Re/ft and the NASA Langley Supersonic Unitary Plan Wind Tunnel up to Mach 4.6. The lowest p_s' disturbance amplitude data on the Cone surface almost track the impact p_t' disturbance data in flight, although there are locally disturbances in the Cone boundary layer higher than the free-stream background. Some of those reflect previously-mentioned Tollmien-Schlichting wave growth and turbulent spots.

Taking $q_\infty = \frac{1}{2}(\gamma) p_{s\infty} M_\infty^2$ to be the logical driving energy measure, the normalized disturbance data have an inverse trend with M_∞ and include a minimum threshold at $M_e \geq 1.3$. A threshold of $\sim 0.02\%$ to Mach 2.0 is very low free-stream disturbance level. Shown in Fig. 17 is data from the Cone microphones in the NASA LaRC 4 SPT as tunnel total pressure p_t was raised from 0.5×10^6 to 4.0×10^6 Re/ft at $M_\infty = 2.0$ and laminar ΔC_p was as low as 0.07 %.

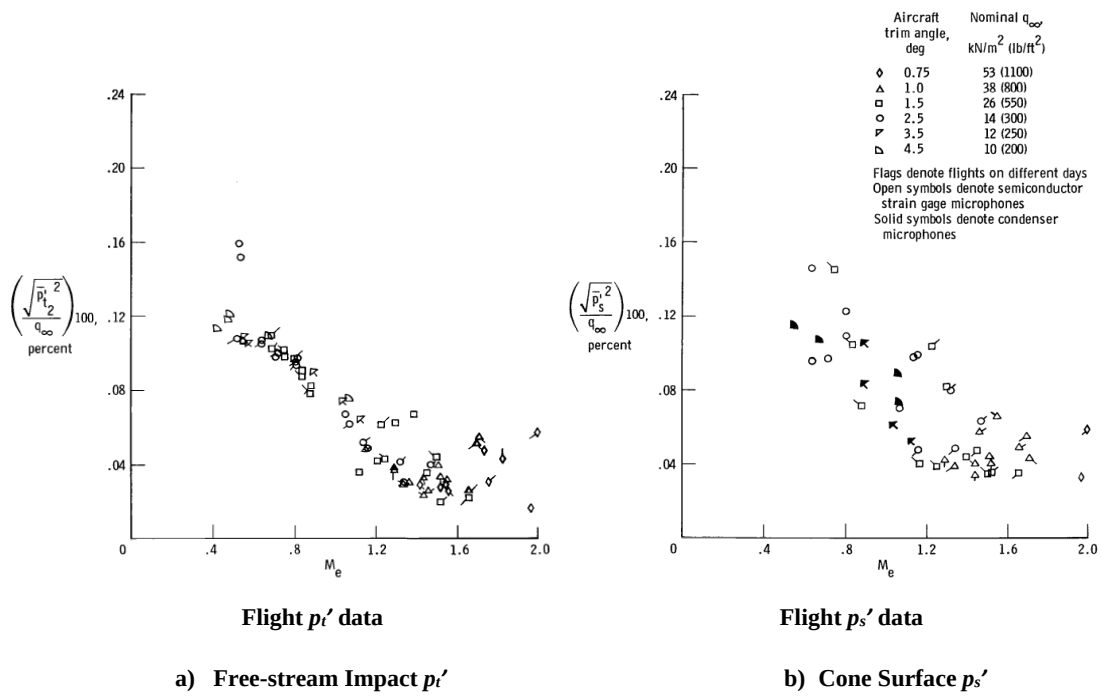
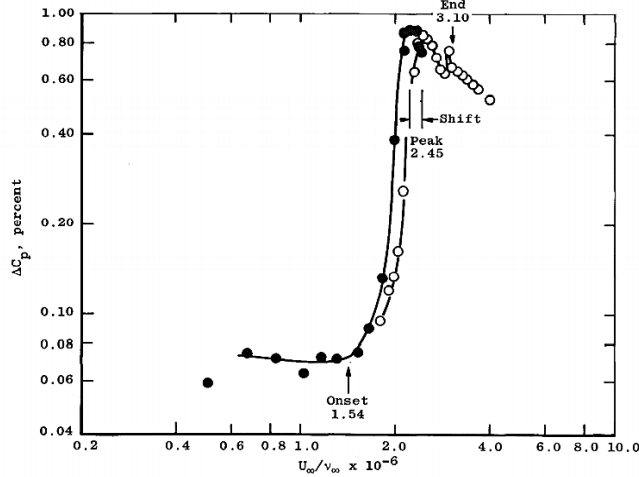


Fig. 16. Measured Unsteadiness In-Flight (Normalized)

Reference Wind Tunnel data (NASA LaRC 4 SPT) on the right at $M_\infty = 2.0$



Reference Wind Tunnel data (RAE 8 X 8) on the left at $M_\infty = 1.4$

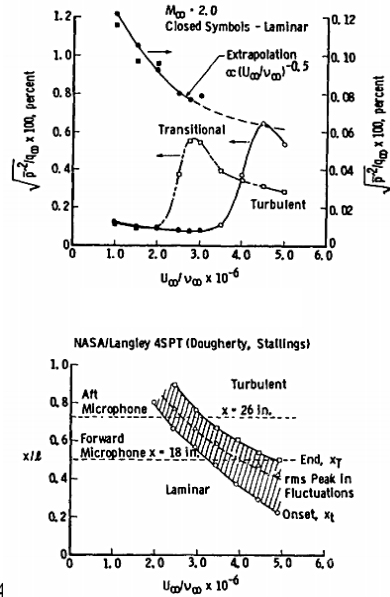


Figure 17. Movement of Transition across the Microphones on the Cone

The data in Fig. 17 on the left and right were taken while holding tunnel total temperature T_t constant. On the left, Re/ft is increased from 0.5×10^6 to 4.0×10^6 in the RAE 8 X 8 at Bedford at $M_\infty = 2.0$. During this instance, it was noticed that the Cone thermocouple temperature data was changing (not fully in thermal equilibrium), and the excursion was stopped and started back again, retracing the passes of the point x_p (location of peak disturbances due

to turbulent spots) passing over the microphone. There was a slight shift in transition location after waiting for the Cone to come into thermal equilibrium (solid symbols before, open symbols after). Laminar disturbance levels, ΔC_p , before transition were approx. 0.07 % in the RAE tunnel and 0.08 % in the LaRC 4 SPT.

When the Cone boundary layer is locally turbulent over the microphone's location on the Cone surface, there is a large increase in ΔC_p rms where the peak in fluctuations occurs midway through the transition zone due to turbulent spots passing over the microphone is still higher. Re_T was 6.7×10^6 in the RAE Tunnel and about the same in the Langley tunnel at $3.0 \times 10^6 Re/ft$ and increased to a maximum of 7.3×10^6 at approx. $4.9 \times 10^6 Re/ft$.

As the boundary layer transitions to fully turbulent, the pressure oscillation spectrum becomes fully random (without spectral peaks). If there should happen to be larger external disturbances incident on the Cone, spectral content of those disturbances (both random and narrow-band) will be recorded over and above the Cone boundary layer noise seen by the microphone.

The NASA Ames 9- x 7-Ft Supersonic Tunnel data at $3.0 \times 10^6 Re/ft$ show Re_T decreasing from 6.9×10^6 at $M_\infty = 1.6$ to 6.2×10^6 at $M_\infty = 2.5$. The corresponding measured laminar disturbance levels, ΔC_p , were 0.14% down to 0.12 %, a little higher for about the same measured Re_T . In the NASA Langley 4-Ft Supersonic Unitary Plan Wind Tunnel^{4,13} Re_T was 6.5×10^6 at $M_\infty = 1.6$ and 6.0×10^6 at $M_\infty = 2.0$, with ΔC_p 0.11% at these Mach numbers. In the Langley UPWT higher test section, Re_T was 4.8×10^6 at $M_\infty = 2.86$ and 6.5×10^6 at $M_\infty = 4.6$ with ΔC_p measured under laminar boundary layer flow at approx. 0.22 % at 2.0 - $3.0 \times 10^6 Re/ft$.

The data in the RAE Bedford 3- X 4-Ft High Speed Supersonic Tunnel showed ΔC_p approx. 0.2 % and comparable Re_T at $M_\infty = 2.5$ to 4.0. The Cone data in these tunnels correlated well showing a decrease from the maximum in Re_T near Mach 1.5 to a minimum near Mach 3.0, increasing in the supersonic to hypersonic range nearly the same as the data taken in the AEDC vKF 40-In. Tunnel A. Data shown for the AEDC 16-Ft Supersonic Tunnel are Pate's data⁹.

Data are shown from three AEDC wind tunnels – vKF Tunnel A, the 16-Ft Transonic Propulsion Wind Tunnel, and the 4-Ft Aerodynamic Wind Tunnel and several NASA Langley, British, and French Tunnels in Fig. 18.

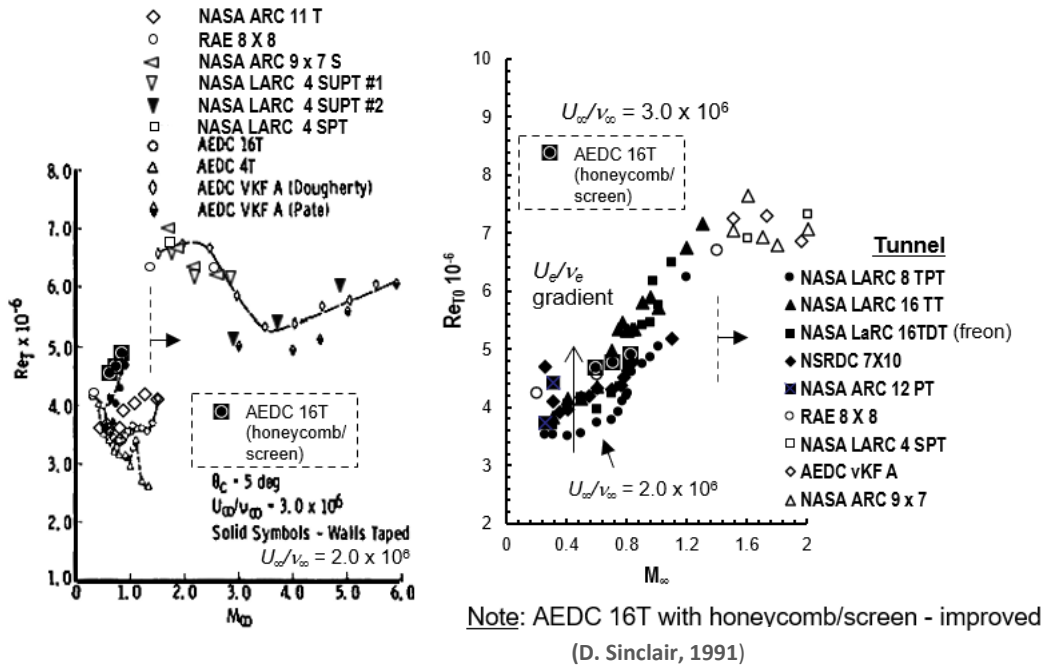


Fig. 18. Re_T Comparisons in Several Wind Tunnels to Mach 6

It was recognized that there were large ΔC_p levels in the AEDC 16T and 4T tunnels, with near discrete peaks over 2% at high subsonic M_∞ (non-dimensionalized spectral peaks $G(f)$ of $> 10^{-3}$ approaching 10^{-2}). The Re_T levels measured on the Cone confirmed an effect down to the lowest Re_T measured in the 4-Ft Transonic Tunnel at approx. 2.5×10^6 . An experiment was performed with the Cone in both of the transonic tunnels as this largest contributing noise source measured by the Cone microphones was noise from the 60-deg inclined holes in the perforated test section walls. The experiment was to tape the walls closed temporarily to check for the effect of the wall noise at $M_\infty \leq 0.9$ ⁴. Discrete

narrow-band tones from the holes in the walls disappeared, and Cone Re_T with the walls taped approached the same trend as the lower-noise slotted-wall transonic tunnels and solid-wall subsonic tunnels at NASA/LaRC and RAE. The inset subsonic data marked AEDC 16T (honeycomb/screen) were taken years later in an AEDC Transition Cone 1991 entry after the tunnel was modified with addition of a honeycomb-screen combination to reduce the test section turbulence level and reduce flow angularity, measurements of resultant turbulence level/flow angularity included.

The noise levels measured in the NASA Ames 11-Ft Transonic Wind Tunnel were high⁴, similar to the levels in the AEDC transonic tunnels, with the Z-Baffled longitudinal slots for test section wall ventilation in that NASA tunnel. The experiment to tape the walls closed temporarily had much the same result as in the AEDC transonic tunnels, small increase in Re_T toward the trend of the lower- disturbance tunnels at subsonic Mach numbers. The ARC 11-Ft tunnel has also undergone upgrade improvements since the Cone tests in the 1970s.

The transonic tunnel ventilated-wall noise effect was reduced, but test section wall discrete noise is not the only flow unsteadiness source reaching the test section and some measured unsteadiness, particularly at lower frequencies of approx. 3 kHz or less, appeared to have little to small effect on measured Re_T . The lower-noise slotted test section wall-disturbances (low frequency) peaked at high subsonic Mach numbers to ΔC_p levels over 1.0-1.5% with apparently little effect. Test section-generated noise tended to have a resonance effect near $M_\infty = 0.7$ -0.75 in rectangular-shaped test sections.

The two AEDC tunnels, Ames 11-Ft Tunnel, and two Industry Tunnels had ΔC_p levels peaking to as large as 2-3%, in the case of the AEDC 4-Ft Tunnel to a maximum of 3.6%. The high noise spawned a parallel research study into ventilated-wall aerodynamic noise characterization and possible means to suppress it. Studies conducted in the 6-In. Acoustic Research Tunnel at AEDC on wall noise and on turbulence suppression in the stilling chamber proved that various wall tones could be eliminated with minor modifications and turbulence suppressed using honeycomb-screen combinations. The NASA/Ames 12-Ft Pressure Tunnel was the low-turbulence reference in this correlation.

Wind tunnel compressor fan blade-passing discrete tones (proportional to fan speed and the number of blades) were identified incident on the Cone in the signatures of a number of the tunnels, but none were shown to have major consequence on Re_T . Repeat test points at the same operating conditions and repeat Cone entries in the same tunnels years apart proved the wind tunnels to have highly-repeatable Re_T and ΔC_p and p_t'/p_t so as to ascertain these characteristics as revealing unique and repeatable signatures for each particular tunnel.

Transition Cone data were taken over as much of each wind tunnel's operating envelope of air-flow Mach number M_∞ and density ρ_∞ as possible covering as wide a range in Re/ft . It was found that ΔC_p as measured on the Cone at the wind tunnel centerline was also correlated to microphone p_s' measurements made on the wind tunnel walls or on free-stream probes at subsonic Mach numbers. Surface-mounted microphones were always underneath a local wall boundary layer (usually turbulent) and thus measured turbulent boundary layer noise locally plus noise from other sources propagated to the microphone location. That noise propagated to the microphone from other sources in wind tunnels was always above the spectrum level of the local Cone surface boundary layer.

The European Co-investigators additionally reported on the European wind tunnel data¹⁴⁻¹⁷, and data were further summarized at an AGARD symposium¹⁸ together with details of the flight test correlation and calibration techniques used¹⁹⁻²¹. The need to calibrate Cone testing for the effect of heat transfer had been stressed by Eli Reshotko²⁰⁻²¹.

Work on the prediction of turbulent boundary-layer noise by M.V. Lowson²² proved to be an important reference in that a steady-flow, attached, turbulent boundary layer has a predictable spectrum and that integration of the total composite noise across the spectrum gives a predictable ΔC_p . This spectrum provided a reference minimum-expected Cone turbulent noise level that could be related to a Reynolds number and boundary layer thickness δ at the microphone location. Lowson's prediction for attached boundary-layer noise on a smooth wall is

$$\Delta C_p = p_s' \text{ rms } / q_\infty = \sqrt{p_s'^2} / q_\infty = 0.006 / (1 + 0.14 M_\infty^2) \times 100\% \quad (3)$$

where, in this Program, M and q were both evaluated at free-stream, ∞ , conditions and p_s' is the square root of the measured surface pressure squared, time-averaged after recording over the full measurement frequency bandwidth.

Cone surface microphone measurements made showed good correlation with Lowson's expression for the overall root mean square and for the mean square power spectrum $p_s'^2$ in wind tunnels at subsonic Mach numbers above approx. 0.3 and supersonic Mach numbers above approx. 3.0 in the Re/ft range approx. 2.0 to 4.0 x 10⁶. Below approx. Mach 0.5, Re/ft for a Cone measurement was usually at lower than 2.0 x 10⁶, and q_∞ was correspondingly low⁴.

There was an interesting experiment performed in the RAE 8- X 6-Ft Transonic Tunnel at Farnborough affixing a pitot probe to the Cone surface (fixed at 18 in. Cone location 180-deg opposite from the forward microphone, at x_1). This fixed pitot probe gave comparison data between pitot probe and microphone data as Re/ft was varied and the

transition zone passed over the $x_1 = 18$ in. location (zero incidence and adiabatic wall conditions). Data acquired at $M_\infty = 0.3$ and 1.19 are shown in Fig. 19⁴. The better interpretation of the end-of-transition Reynolds number Re_T is judged to be that from the pitot probe, 4.3×10^6 at Mach 0.3 and 5.25×10^6 at Mach 1.2.

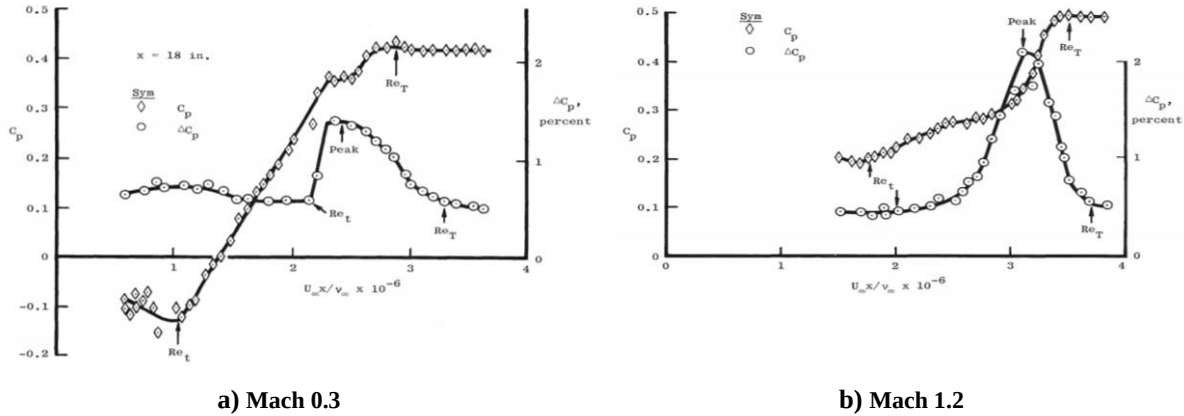


Fig. 19. Transition Location and Disturbance Level on the AEDC 10-Deg Cone in the RAE 8 X 6 T

An interesting detail from this fixed pitot probe comparison experiment was the sublimation pattern using naphthalene and acenaphthene as the indicators for the turbulent wake behind the fixed pitot probe (as a protuberance boundary layer trip on the Cone surface) as shown in Fig. 20. Clearly, the forward microphone opposite the fixed probe was determined to be in laminar flow at $x_1 = 18$ in. Expectations for a single roughness element protuberance on the Cone surface are that a 17-deg turbulent wedge would form behind the protuberance. Considering the curvature of the conical surface, the sublimation wake study confirmed what was expected, transition to turbulence in a 17-deg wedge at that Re/ft .

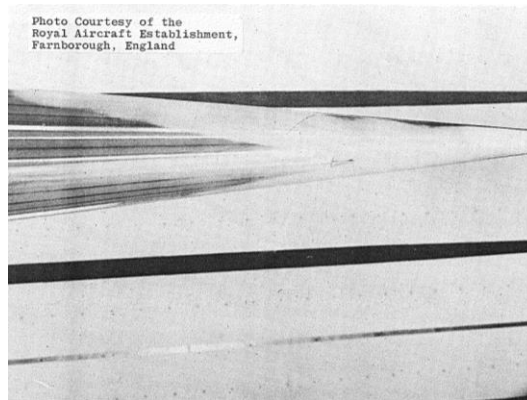


Fig. 20. Sublimation Pattern on the AEDC 10-Deg Cone with the Fixed Pitot Probe attached

The addition of more instrumentation to try to characterize transition or flow disturbances more fully in the wind tunnels was attempted in a number of the tests and reported separately. Use of hot wire anemometer and total pressure fluctuation probes and hot-film anemometer gages on surfaces has been perfected to a far greater extent than existed at the time of the AEDC 10-Deg Cone tests and is a welcome addition.

A lesson learned was that more definition to discern total pressure fluctuations p'_t and u' , v' , and w' velocity fluctuations (turbulence intensity) from incident pressure fluctuations in the disturbance spectra would be beneficial and could always be acquired at a future time following the Cone-entry correlation tests. Many

investigators have done so, at around the time of the AEDC Cone Test Program and since. This is documented here as the **Seventh Lesson Learned** in the correlation program, as good research invariably leads to additional research to answer questions that inevitably arise, and that research has been greatly expanded.

The NASA Ames 12-Ft Pressure Tunnel had turbulence suppression screens in the stilling (settling) chamber, large nozzle contraction ratio entering the test section. The Cone Re_T data in this tunnel was about the same as in other low-disturbance tunnels⁴. It was known to be a low-turbulence intensity wind tunnel standard. Turbulence measured by NASA in the Ames 12-Ft Pressure Tunnel²⁸ was as low as 0.040 % ($u'/u \times 100$, %, turbulence intensity) in the test section at Mach 0.3 up to approx. 0.2% to $M_\infty \leq 0.8$, where u' is the axial component of the rms velocity fluctuations.

The NASA Langley 8-FT Pressure Tunnel has been used for important Laminar Flow Control (LFC) testing, and NASA determined the turbulence intensity to be not very much greater than in the Ames 12-Ft Tunnel²³, and Re_T on the Cone was only a bit lower⁴. At $M_\infty = 0.2$, the fluctuating pressures in both tunnels are considered to be low ($p_t'/p_t \approx 0.01\%$). The Cone flight test free-stream impact probe data, normalized by $p_{t\infty}$, are shown in Fig. 21. Extrapolation down to the $M_\infty = 0.2$ conditions in these wind tunnels confirms these two tunnels indeed to have very low turbulence intensity expressed as p_t'/p_t for e.g., testing at aircraft take-off and landing speeds. Turbulence level in terms of p_t'/p_t total pressure in AEDC 16T was reduced nearly an order of magnitude at subsonic Mach numbers, closely in line with the NASA ARC 12 PT after the addition of the honeycomb-screen improvement³⁰.

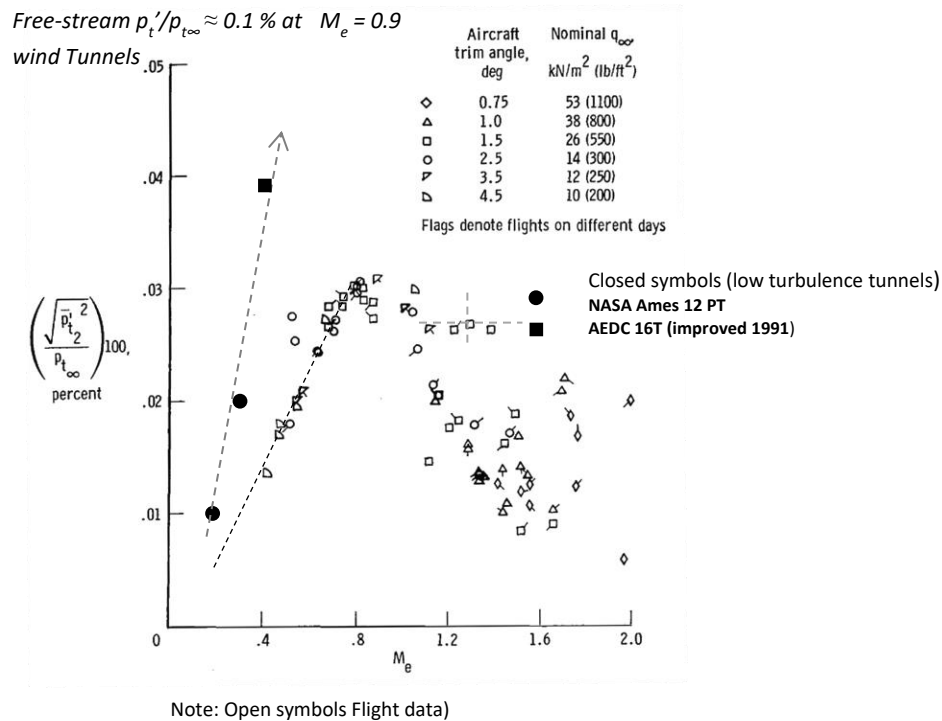


Fig. 21. Comparison of NASA Ames 12-Ft Pressure Tunnel Turbulence at Low Speeds with Flight

Premature transition (reduced Re_T) in wind tunnels found in the AEDC 10-Deg Cone Correlation Program had an accompanying set of flow disturbance data (unsteadiness) that indicated repeated signature characteristics of the wind tunnels⁴; but separating out specific source contribution effects on the Cone boundary layer transition was not as clear as was possible in tunnels operating at Mach 3 and above. The Pate-Schueler correlation dealt with a single noise phenomenon in those tunnels, radiated tunnel wall boundary noise and the shear stress in the tunnel boundary layer source. “Quiet” tunnels – Mach 2.0 to 6.0 or greater – have been put in operation (a Mach 8 “quiet” tunnel under construction presently) to deal with high supersonic- hypersonic issues such as laminar-turbulent transition heating.

NASA DFRC went on to investigate bug (insect) impacts on the wing leading edge in aircraft flights making landings and take-offs in a number of major airports around the US^{24,25} and “bug washer” devices that could remove their effect on leading edge premature roughness-element-induced transition.

NASA DFRC and ARC performed a tunnel-flight experiment using the AEDC 10-Deg Facsimile Cone and its distributed pressure instrumentation to $x = 35$ in. on the F-15A aircraft at 11 deg angle of attack with respect to the aircraft centerline²⁶⁻²⁸. This experiment in-flight, at resultant incident angle 2.2 deg in banked, coordinated turns, and in the NASA Ames 6- X 6-Ft Supersonic Wind Tunnel at M_∞ 1.5 and 1.8 (tunnel and flight) provided a good correlation wind tunnel-to-flight of Cone flow separation, measured separation data against computational fluid dynamic prediction on the Cone surface and wind tunnel measurement versus flight at the same flight conditions.

A transition strip composed of nominally 0.014-in. diam. glass beads was affixed to the Facsimile Cone surface at $x = 4.9$ in. from the Cone apex in those tests. The rms mass flux levels m'/m on a single hot wire in the 6 X 6-Ft wind tunnel have been measured at 1.5% and rms pressure fluctuations ΔC_p 1.1% of q_∞ at these Mach numbers compared to as low as 0.04% measured by the total impact Kulite pressure transducer in-flight. The unit Reynolds number in the wind tunnel and flight, Re/ft , was between 3 and 4×10^6 . Re_T was approx. $1.3 - 1.6 \times 10^6$ with the fixed-transition strip on the Cone, guaranteeing a fully-developed turbulent boundary layer for flow separation around the Cone.

The flight control of the F-15 A aircraft proved it to have been a very stable test bed for the Cone tests at zero and at high incidence. Early in the Program, a USAF RF-4C aircraft based at Wright-Patterson AFB had been slated for the Cone flight tests. A 0.05-scale model of the aircraft and Cone had been tested in the AEDC 16-Ft Transonic Tunnel (16T)²⁹, and an advanced flight test of the RF-4C had been made with $\alpha - \beta$ nose boom vanes to check how close incidence angle could be controlled in the flight test airspace. It was judged adequate to meet Program requirements (≤ 0.25 deg). Natural transition location on the Cone had proved to be very sensitive to small incidence angle^{1,4}. When the decision was made between the USAF and NASA to use the F-15 A test bed aircraft at DFRC instead, NASA's decision to proceed was based on no need to retest in the wind tunnel with an F-15 plus Cone scale model (proved to be a sound decision).

It was addressing of specific physical characteristics of the participating wind tunnels by the invested parties that yielded ensuing benefits of flow quality improvements; e.g., possibly better correlation of effects such as on drag coefficient measured on the many test models in the tunnels, including cases of subsonic flow, transonic cruise supersonic flow, flow separation, and buffet.

Among improvements cited are those in AEDC 16-Ft, and also improvements in the 4-Ft Transonic Tunnel, with the addition of honeycomb and screens in the stilling/settling chamber along with careful maintenance of all the turning vanes in the corner just ahead of the stilling chamber. The Transition Cone had been put back into the AEDC 16-Ft Tunnel for retest by D. Sinclair³⁰ after the improvements that greatly reduced the turbulence, and the results, shown in Fig. 21, were closer in line with other low-turbulence tunnels at subsonic/transonic speeds. The NASA Ames 11-X 11-Ft Transonic Wind Tunnel has likewise added a honeycomb-screen combination in the settling chamber ahead of the nozzle.

Dennis Mabey³¹ of RAE also cites focused studies and improvements that were realized in RAE wind tunnels based on their years of research programs.

Significant improvements are not always realized by addressing just one aspect of flow quality, but addressing more aspects at the same time; e.g., flow angularity, turbulence, acoustics, Mach number uniformity, thermal and humidity management, and transonic shock cancellation. The most dramatic improvement in the two AEDC transonic tunnels was, perhaps, in reduction of turbulence and associated flow angularity. Along with physical improvements has been a dramatic improvement in instrumentation and controls and making measurements in this digital age of technology. Flow angularity across the test section was reduced in these tunnels down to less than ± 0.1 deg, turbulence intensity down to levels among the best of conventional production wind tunnels ($\sim 0.1\%$ at M_∞ 0.8-0.9).

The by-pass mechanisms that can influence boundary layer transition location are many and were by no means sorted out in the Cone Program. The Reynolds number range from approx. 1.5×10^6 to 4.5×10^6 /ft was that which was possible on the 3-ft-long Cone traverse measurement length. There was good comparison with freon as the working fluid versus air (different Froude number $Fr = U_\infty/(gL_{ref})^{1/2}$ in the NASA Langley 16-Ft Transonic Dynamics Tunnel)⁴, where g is the gravitational constant and L_{ref} a reference length. Cone data in freon matched that in air.

After rich histories over decades, the NASA Langley 16-Ft Transonic Tunnel, 8-Ft Transonic Pressure Tunnel, and 4-Ft Supersonic Pressure Tunnel have all been demolished. The NASA Ames 14-Ft Transonic Tunnel has also been demolished. Investments were made in high Reynolds number simulation for model testing in wind tunnels.

The extrapolation to high Reynolds numbers using cryogenic nitrogen as the working fluid at below -250°F and high working pressures was successful in the design of high Reynolds number wind tunnels. Capable of 30×10^6 chord Reynolds number for aircraft at subsonic cruise, the low-disturbance NASA Langley National Transonic Facility (NTF) and European Transonic Windtunnel (ETW) in Cologne, Germany, are engaged in Natural Laminar Flow (NLF) studies and means of laminar flow control. Laminar Flow Control (LFC) and NLF technology applications for extending the laminar flow region on subsonic wing and empennage surfaces allow decrease in friction drag and a potential for savings in fuel consumption and extending aircraft range.

NASA DFRC conducted the F-16 XLR Wing Glove LFC flight experiment³² to Mach 2.0 using temperature-compensated hot film anemometers and obtained transition Reynolds numbers as high as of 22.7×10^6 . NASA AFRC conducted tests of a supersonic Natural Laminar Flow airfoil³³ to Mach 1.8 or greater mounted on an F-15 B aircraft achieving laminar extension regions to over 11×10^6 Reynolds number.

IV. Conclusion

Credit is due to the USAF for initiating the AEDC 10-Deg Cone Program and to NASA and the other participating agencies who contributed so much to its utilization and historical significance and acted upon the data acquired to realize improvement benefits for the user community who test in those tunnels. This includes NASA's experiment later with flow separation done with the Facsimile Cone in-flight and in the NASA Ames 6-X 6-Ft Supersonic Tunnel and AEDC's Re_T re-test with the Transition Cone in the AEDC 16-Ft Transonic Tunnel showing the measure of wind tunnel facility flow quality improvement investments made. Credit is given to the AEDC Pate and Schueler correlation program in supersonic and hypersonic wind tunnels that preceded the AEDC 10-Deg Cone Program and to the many years of transition research that has been done at AEDC and the benefits realized.

So much of what was learned had to do with physical design features for wind tunnels and test techniques used which in turn had benefit on design and operation of new wind tunnels since, including high Reynolds number tunnels for investigating Natural Laminar Flow and Laminar Flow Control. Knowledge gained about wind tunnel facilities has been implemented in new facilities designed and built since and in current operation.

Lessons learned from the AEDC Cone Wind/Tunnel Laminar-Turbulent Correlation Program have been provided as they had generally to do with attention to detail in conducting experiments and control over the variables that can affect the outcome, as recollected by the Program Principal Investigator. Lessons learned and acted upon in the course of the Program are included. The original Cone data collected were tabulated and reported as measured, then correlated, for all who use the original data to be able to correlate it themselves in any manner they might utilize it. The body of data collected reflects the contributions of many dedicated individuals from many participating organizations to make it the comprehensive data set that it is.

Acknowledgments

David F. Fisher, Co-Investigator for the Flight Test, and the NASA Dryden (now Armstrong) Flight Research Center F-15 A Test Team are acknowledged, and special appreciation is given to Test Pilots Bill Dana and Einar Enevoldsen. Acknowledgment is extended to D.S. McRae and D.J. Peake of NASA Ames Research Center who extended investigation with the AEDC Facsimile Cone (Pate's Cone) to the follow-on flow separation tunnel/flight correlation tests at 2.2-deg incidence angle and Mach 1.5 and 1.8 on the F-15 A aircraft and in the NASA Ames 6 x 6-Ft Supersonic Wind Tunnel.

The National Transition Study Group is acknowledged for their insight and oversight for the AEDC 10-Deg Cone tunnel/flight correlation: Eli Reshotko, Chairman, Mark Morkovin, Phil Klebanoff, Leith Potter, Leslie Mack, Jim Kendall, and Ken Stetson.

Members of the AEDC Team are acknowledged: Robert O. Dietz, Carlos Tirres, Alex Money, Marion L. Laster, Keith Kushman, Jack D. Whitfield, Leon Ring, Bruce Estabrooks, Sam R. Pate, Clarence (Chuck) Schueler, Oliver P. Credle, John A. Benek, Michael D. High, David L. Whitfield, Darryl Sinclair, Wallace Luchuk, Travis Binion, Elton Thompson, John C. Adams, Carl T. Kidd, Michael Mills, and Henry T. Jones and the many Test Facility skilled craftsmen who built, maintained, and instrumented the AEDC 10-Deg Cone and the Facsimile vKF Cone.

University of Tennessee Space Institute Professor Jain-Ming (Jimmy) Wu and AFIT student Robert Lock are acknowledged, and thanks are extended to Col. D. H. Daley at the USAF European Office of Aerospace Research.

Appreciation was extended to Co-investigators at NASA Ames and Langley Research Centers Frank Steinle, Robert L. Stallings, Jerome T. Foughner, William Igoe, W. D. Harvey, and Calvin Stainback, R. de Kuyper at CALSPAN Corp., D. Davidson at the Naval Ship Research and Development Center and to Co-Investigators in Europe Dennis Mabey of RAE and R. Jordan of ARA. Ltd, United Kingdom, Xavier Vaucheret, Office National D'Etudes et de Recherches Aerospatiales, France, and Rudolph Ross, National Luchten Ruimtevaartlaboratorium (NLR), the Netherlands. A special thanks is given to Phillippe Poisson-Quinton, ONERA.

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