SUMMARY OF ALL CYCLE II.5 SHEAR AND BOUNDARY LAYER MEASUREMENTS - AERODYNAMICS

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Abstract

The two measurement systems were used to measure mean velocity and velocity, mass flux and total temperature fluctuations in the turbulent boundary on the fuselage of a KC 135 aircraft. The boundary layer thickness ranged between about 20 and 30 cm for the range of flight Mach numbers from about 0.25 to 0.85 and Reynolds numbers between 3 and 6 $\times 10^6$/m. The adaptation of each system for use in airborne applications is discussed. The data obtained from each system are given and compared with each other and they indicate that the two systems represent viable ones for use in future airborne turbulence experiments.

Introduction

Ground testing and aerodynamic simulation techniques can reproduce flight conditions in only a few applications. The primary parameter that cannot be accurately simulated is Reynolds number. Therefore, the accurate determination of quantities that are known to be sensitive to the Reynolds number may require the instrumentation of the component or the entire aircraft for in-flight experimentation. Several difficulties arise in regard to flight testing, one being the measurement of turbulent flow-field quantities at high speeds. These turbulence quantities (such as streamwise and cross-stream fluctuation intensities, Reynolds shear stress, and spectra) are sensitive to the Reynolds number and are critical to the success of present computational-fluid-dynamics codes that use subgrid closure schemes. Thus, there is a clear need to measure turbulence at realistic Reynolds numbers.

In the last few years, technical advances in hot-wire anemometry and the development of the laser velocimeter have made the measurement of turbulence properties in compressible flows realizable (e.g., Johnson and Rose[1,2]). Wind tunnel experience has proven that hot-wire probes can be constructed to withstand a wind tunnel environment almost indefinitely. Also, the wire response for transonic flows is now sufficiently understood that quantitative measurements of turbulent flow properties can be made for this flow regime.[3] Both the laser velocimeter and hot-wire anemometer are proven techniques for use in high-speed wind tunnels; their potential contribution to flight testing should not be overlooked.[4]

Recently, these two techniques were used in a flight experiment to probe the viscous flow on an aircraft fuselage. This application to in-flight flow-field measurements is the subject of the present paper. The approaches taken to adapt the two techniques for flight measurements are discussed and the performance is evaluated. Problems unique to flight applications are also addressed. Examples of data taken are presented to indicate the overall success of the flight program. The following section briefly describes the experimental setup.

Experimental Apparatus

The flight experiment evolved from a joint program between Air Force Weapons Laboratory, Kirtland AFB, and NASA-Ames Research Center on the propagation of light through the turbulent boundary layers of aircraft. The desire to measure both local turbulence properties and optical distortion effects of the flow field dictated the design of the experiment.

To measure the light propagation properties of the boundary layer, a mirror was mounted away from the fuselage to return the transmitted beam to the airplane. Figure 1 shows the return mirror support system mounted aft and above the wing of an Air Force KC-135. The window for the light propagation studies and the two holders for the hot-wire probes are apparent. The airfoil-shaped upper and lower support struts for the return mirror housed part of the laser velocimeter optics. Windows for the laser velocimeter (not visible in the photograph) were located midchord of the two struts. The return mirror was approximately 1 m from the fuselage; the side support struts were separated from each other by about this same distance.

Two 5-μm tungsten wires (one to measure mass-flux fluctuations and the other to measure total-temperature fluctuations) were mounted on each of the two probe supports in Fig. 1. Mass-flux fluctuations were measured using DISA model 55M01 constant-temperature anemometer systems with high overheat ratio settings. Constant-current anemometer systems operating at very low overheats were used to measure the total temperature. To reduce the possibility of wire breakage during takeoff and landing the probes were turned backward to the flow during these periods.

The 5-μm sensors were spot-welded to nickel electrodes that had been epoxied into a ceramic body. The probe was about 2 cm in length and about 5 mm in diameter. These probe bodies held four electrodes, making two sensors with a length-to-diameter ratio of about 100. All sensors were calibrated before the flights in the Ames 2- by 2-Foot Transonic Wind Tunnel. Although the wind tunnel cannot duplicate flight-length Reynolds numbers, it can duplicate the range of wire Reynolds numbers encountered in flight. Thus, the required individual calibration of wire sensors over the range of flow variables encountered in flight could be realized. The constant-current sensor and anemometer were calibrated by a no-flow oven test, followed by a wind-tunnel test to establish the variation of wire recovery factor with Mach number. The constant-temperature sensor and anemometer were calibrated in the wind-tunnel flow using a ratio of heated-temperature to recovery-temperature of 2.0. The remainder of the calibration procedure, described in detail by Rose and McDaid[4] has been used extensively in previous wind-tunnel tests.[2] The hot-wire anemometer data reduction procedures used were the same as those described by Johnson and Rose.[2] Finally, the two pair of wires could be positioned continuously at any distance from the fuselage surface up to about 35 cm by use of two independent lead screws and crank mechanisms. This allowed the determination of correlation length scales in the transverse direction by fixing the location of one probe and cross-correlating the outputs as the separation between the probes increased.

A laser velocimeter system for determining both the mean and unsteady streamwise velocity component was installed in the aircraft such that flow-field measurements could be made at essentially the same locations as the hot-wire probes. Figure 2 shows a schematic of the velocimeter setup. Because of the expected highly turbulent and/or separated flow produced by the test geometry (discussed below), a Bragg cell was used to ensure an adequate number of fringe crossings for all possible particle trajectories. The beam divider cube was cut so that two parallel beams (one shifted in frequency by 40 MHz) were transmitted to the lower airfoil strut. There, a lens and mirror focused and crossed the beams at the sensing volume in the center of the two airfoil struts. A leadscrew and crank system allowed a continuous positioning of the crossover at any distance from the fuselage up to about 35 cm. The off-axis optics used to collect the forward light scatter signals were housed in the upper airfoil strut. The mirror and lens of the collection optics could also be traversed by a leadscrew and crank system. The photodetector was mounted inside the aircraft cabin. The single particle signals were collected by a zero-crossing-counter-type signal processor. A multichannel analyzer was used to generate histograms of the incoming data, which were then processed by a Hewlett Packard 9830 calculator to produce mean and rms streamwise velocity values. Of course, the laser velocimeter is an absolute system in the sense that no calibrations in a known flow are required once the fringe spacing has been chosen. Details of the data-reduction procedures used in the present study are essentially the same as those described by Johnson and Rose.[2]

As mentioned above, test geometries that produced separated, highly turbulent flows were investigated in flight. Again, the test geometries were dictated by the desire to measure optical distortion effects due to various flow fields. The three distinct flows studied are shown schematically in Fig. 3. The first flow was the fuselage turbulent boundary layer as it arrived at the measurement station approximately 30 m aft of the aircraft nose. Note that for any given aircraft weight, flight Mach, and Reynolds numbers, the boundary layer that was surveyed has an unknown trajectory over the aircraft, and, thus, has an unknown pressure gradient history. The history of the pressure gradient has a significant effect on both the turbulence properties and mean-flow properties of a turbulent boundary layer, no matter what the local pressure gradient is at the measuring station. Therefore, the results, shown later, should be taken as unique measurements rather than measurements characteristic of a flat plate flow (zero pressure gradient). Furthermore, changes in aircraft weight and flight conditions may change the pressure gradient history, making universal correlation of the data with Mach and Reynolds numbers difficult if not impossible.

The second and third test geometries, also shown in Fig. 3, involve porous "fences" or vertical obstructions that are 14.5 cm high and have a 48% porosity. The height of the fences is about one-half the height of the undisturbed boundary layer. They are typical of those used to eliminate resonance conditions over open ports although no open port was used in the present study. The measuring station remained fixed and the fences were attached at two positions. One, denoted the near fence, was about 8 cm upstream of the measuring station, while the midfence was located about 80 cm upstream of the measuring station.

Results and Discussion

A total of 12 flights, each about 5 hr in duration, were made aboard the instrumented KC-135. In the last stage of the test, six additional flights were made with the mirror support system removed to check that it had not affected previous flow measurements. Indeed, sufficient flight time was available to appraise the suitability of the measurement techniques for flight applications. The test envelope covered Mach numbers from 0.25 to 0.85 and altitudes from 0.3 to 10.7 km. The low-altitude tests were conducted over the Gulf of Mexico.

The primary concern in using hot-wire anemometry in flight was wire breakage. However, in more than 100 hr of flight time, wire breakage occurred only once, when ice crystals were encountered below a cirrus layer. Although the ice crystals were not visually apparent, particle arrival rates registered by the laser velocimeter reached 50,000 sec^{-1} before the wires broke. In subsequent flights, flying immediately below cirrus clouds was avoided. Unavoidably, many low-lying clouds were encountered but fortunately did not cause wire breakage. Apparently, the wires were strong enough to shear the water droplets.

The two main concerns of the laser velocimeter were the availability of atmospheric particles for light scattering and the ability to maintain optical alignment. For the test envelope, particle arrival rates ranged from several thousand per second for low altitudes of 0.3 to 1.2 km (1000 to 4000 ft) over the Gulf of Mexico to as low as 5 sec^{-1} for some of the high-altitude flights. However, the rates were sufficiently high under all conditions to obtain mean velocities and turbulence intensities. Regions of high-particle density were not sought out; in fact, they were avoided because of the hot-wire probes. Optical misalignment in flight did not occur. This was especially fortunate, since realignment of the system would have been impossible during flight.

The laser velocimeter was used to obtain mean velocity and turbulence intensity profiles. The hot-wire instrumentation provided mass-flux and total-temperature fluctuation profiles as well as streamwise and cross-stream length scales across the boundary layer. Samples of the data are presented in Figs. 4, 5, 6, and 7. Figure 4 shows the mean and unsteady profiles for each of the three test geometries. The hot-wire data were reduced to values of u' from mass-flux fluctuations using the techniques outlined by Horstman and Rose.[3] The velocimeter values of u are consistent with the expected flows. No reverse mean velocities were observed in the flows downstream of the fences. While the boundary-layer profile (Fig. 4a) is very full, recall that this flow came from an unknown pressure gradient history, including the acceleration over the wing root section. Also evident in Fig. 4a are the relatively low values of $\langle u'\rangle/\bar{u}$ (even considering the high Reynolds number $Re_x \approx 2\times10^8$). It is not possible to determine if these lower values are the result of the very low free-stream turbulence characteristic of the atmosphere or are simply a pressure gradient history effect. The general agreement between the laser velocimeter and hot-wire anemometer results for the rms of $u'(\langle u'\rangle)$ are consistent with comparisons made between the two systems in wind tunnel applications.[3] The comparisons for the fence configurations (Figs. 4b, 4c) are nominally acceptable, considering that the near fence has fluctuations of $\langle u'\rangle/\bar{u}$ in excess of 40% where the hot-wire anemometer cannot yield any more than a qualitative indication of turbulence. However, for the mid-fence, values of $\langle u'\rangle/\bar{u}$ of up to 20% appear to be accurately determined by the hot-wire.

Although the nonintrusive feature of the laser velocimeter appears ideally suited to airborne measurements, the present limitations on signal processing and valid signal occurrence rates make some fluid mechanical functions difficult to obtain. Two such functions are the correlation lengths and spectra, both of which were required in the present study. The continuous analog signals from the hot-wire anemometer, analog correlators, and spectrum analyzers make the determination of these functions straightforward. Figure 5 presents the correlation function $\overline{u_1'u_2'}/\overline{u_1'^2}$ as a function of probe separation in the transverse direction. An exponential curve appeared to best fit all the data. The curve fits can be integrated easily to yield the integral scale in the cross-stream direction L_z. A value of L_z may be determined at each point in the boundary layer to produce a plot such as shown in Fig. 6, which shows L_z variations throughout the layer. The streamwise integral scales L_x shown in Fig. 6 were determined from time autocorrelations and an assumed convection velocity, $u_c \approx 0.8\bar{u}$. Values of L_z/δ and L_x/δ are about 0.1 and 0.3, respectively, consistent with wind-tunnel boundary layers.

Finally, a spectrum obtained in the boundary layer is shown in Fig. 7. Most of the turbulence energy is below 3 kHz and indicates a near Kolmogorov energy decay as would be expected.

Concluding Remarks

The overall in-flight performance of the laser velocimeter and hot-wire anemometer was remarkably good. After wire breakage and other assorted equipment failures on the first flight, the remaining flights were relatively uneventful. Although the rate of valid data signals at high altitudes was low, enough particles were present to make meaningful velocimeter measurements in the aircraft boundary layer. The hot-wire anemometer also appears usable for in-flight aerodynamic research.

References

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[2]Johnson, D. A. and Rose, W. C., "Turbulence Measurements in a Transonic Boundary Layer and Free-Shear Flow Using Laser Velocimetry and Hot-Wire Anemometry Techniques," AIAA Paper 76-399, San Diego, Calif., July 1976.

[3]Horstman, C. C. and Rose, W. C., "Hot-Wire Anemometry in Transonic Flow," *AIAA Journal*, Vol. 15, March 1977, pp. 395-401.

[4]Rose, W. C. and McDaid, E., "Turbulence Measurement in Transonic Flow," *AIAA Journal*, Vol. 15, Sept. 1977, pp. 1368-1370.

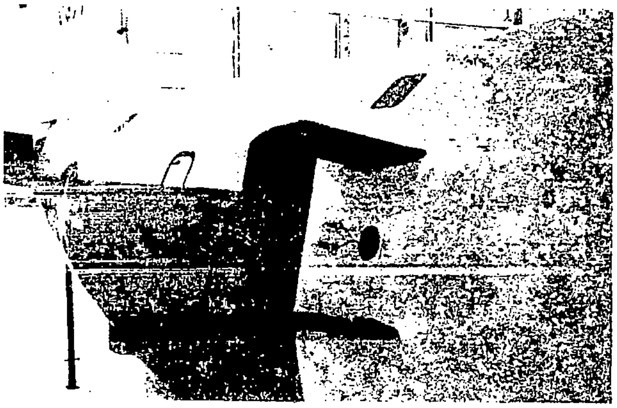


Fig. 1 Photograph of aero-optics experimental installation on aircraft.

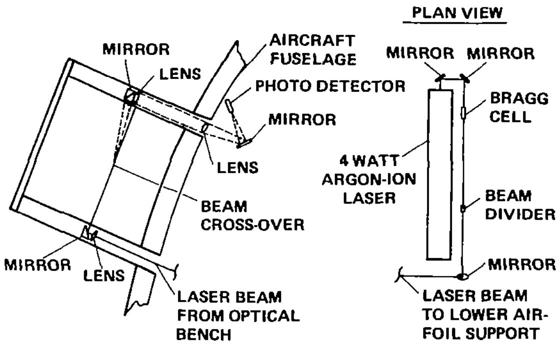


Fig. 2 Schematic of laser velocimeter.

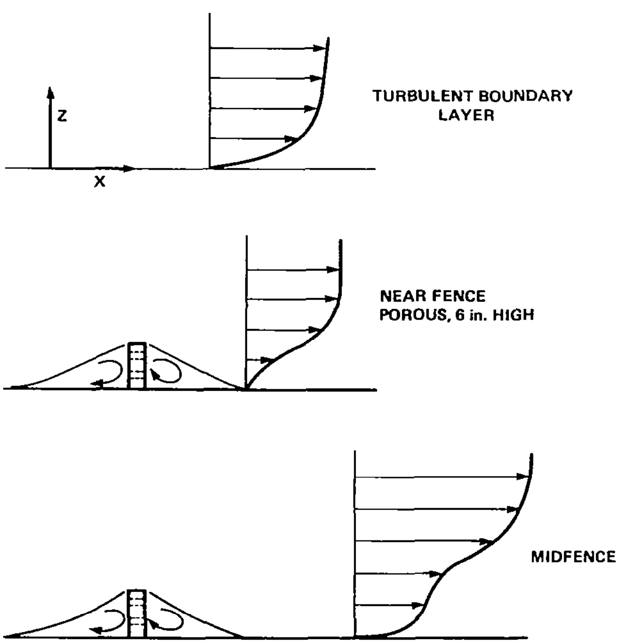
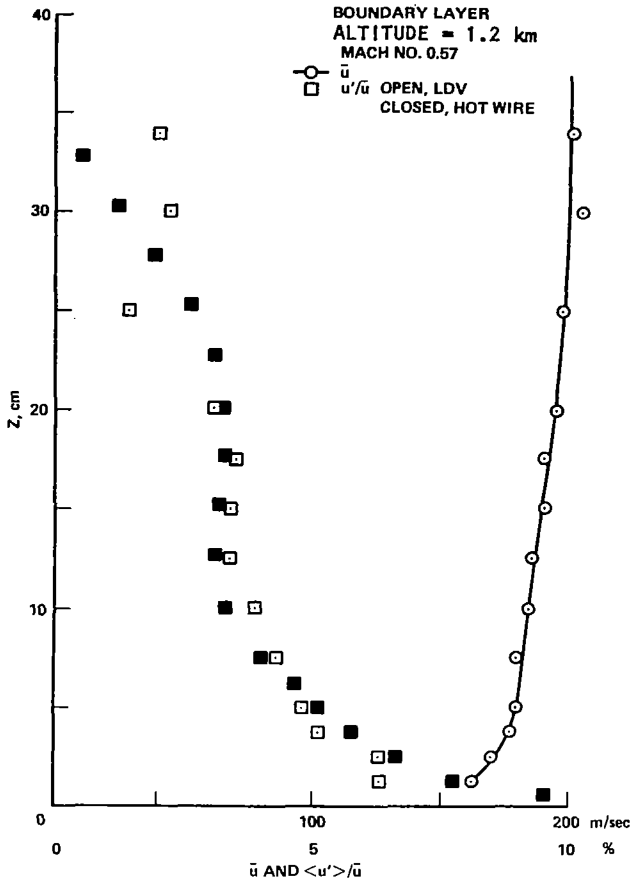
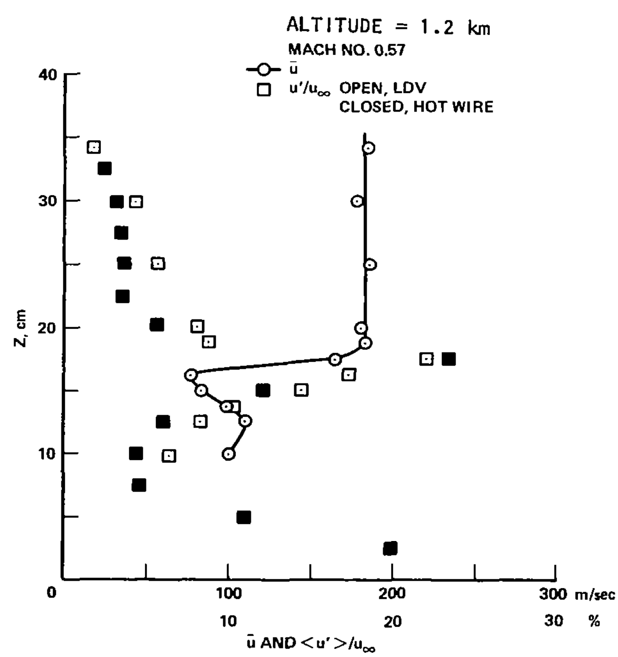


Fig. 3 Sketch of flow conditions studied.

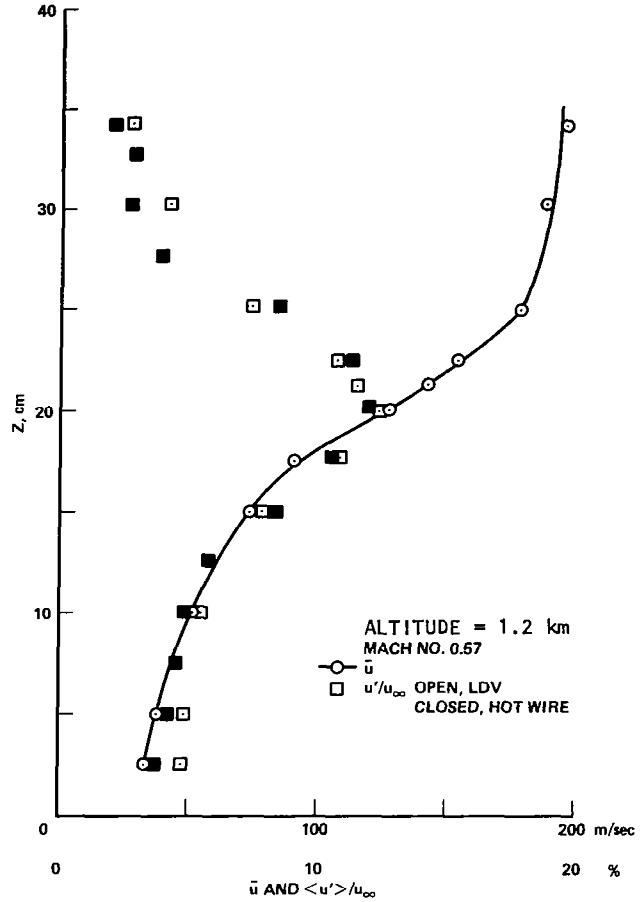


a) Boundary layer.



b) Near fence.

Fig. 4 Measured mean and fluctuating properties typical of those obtained.



c) Midfence.

Fig. 4 Concluded.

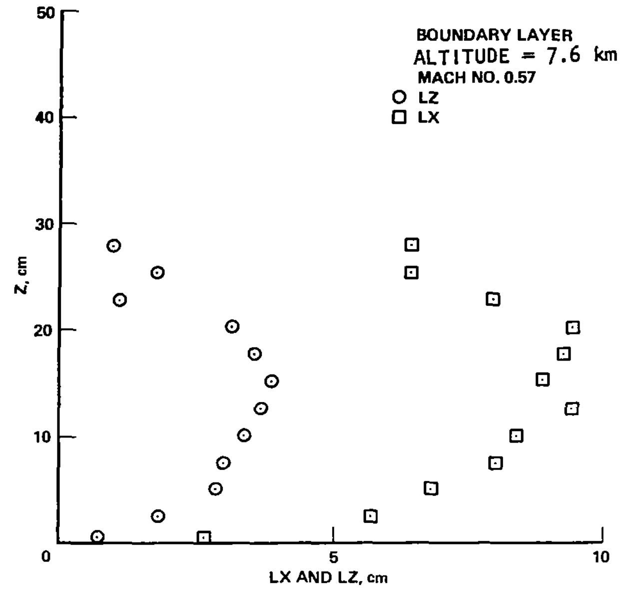


Fig. 6 Typical integral correlation lengths.

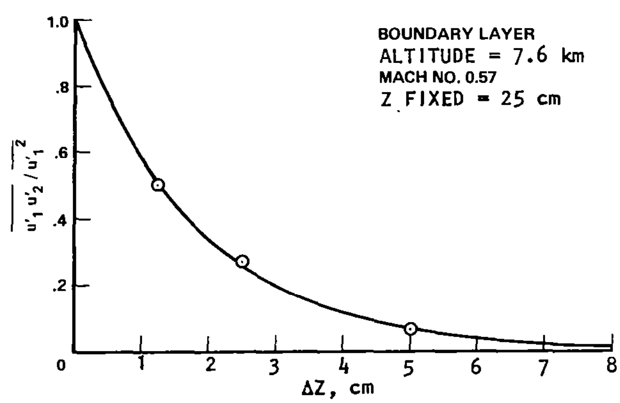


Fig. 5 Typical correlation data and exponential curve fit.

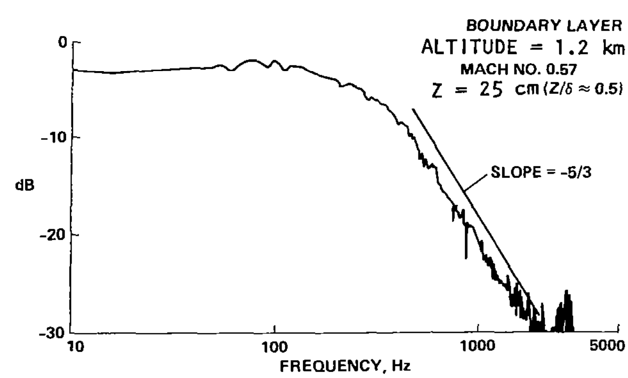


Fig. 7 Typical turbulence spectrum.

KC-135 AERO-OPTICAL TURBULENT BOUNDARY LAYER/SHEAR LAYER MEASUREMENTS

ABSTRACT

Recent Air Force Weapons Laboratory (AFWL) airborne laser propagation experiments have examined aero-optical effects associated with propagating a laser beam through aircraft turbulent boundary layers and shear layers. This series sought to compare observed laser optical performance levels with those inferred from aerodynamic measurements of unsteady densities and correlation lengths within these random flows. Optical instrumentation included a fast shearing interferometer (FSI). A 9cm diameter collimated helium neon laser beam made a double pass through the aircraft random flow via an airfoil mirror located one meter from the fuselage. Typical aircraft turbulent boundary layer thicknesses measured 0.3 meters. Averaging many FSI generated Modulation Transfer Functions (MTFs) and Fourier transforming this average yields the expected far-field intensity degradation associated with an aircraft-mounted laser system. Aerodynamic instrumentation included fine wire probes to measure unsteady temperature and mass flux. A laser doppler velocimeter measured unsteady velocity within the flows. An analysis of these data yielded point measurements of unsteady density and correlation length. Integration of these aerodynamic parameters through the random flow region in turn yields predictions of optical performance via the Gladstone-Dale law. Thirteen flights were flown at altitudes ranging from 0.3 to 11.3 kilometers while Mach numbers varied from 0.25 to 0.85. In addition to fundamental boundary layers, two identical fence configurations were examined located 8 cm and 80 cm upstream of the optical axis. An array of

thermocouples was attached to the inside skin of the aircraft to examine heat transport to the boundary layer. Correlations between the turbulent boundary layer (TBL) aerodynamically-inferred Strehl ratio I/I_0 and its MTF optical counterpart (correlated to large aperture) were very good. Single pass phase variances varied from 0.08 to 0.18 waves ($\lambda = 0.63\mu$) for aircraft dynamic pressures ranging from 0.04 to 0.19 standard atmospheres. Measured TBL correlation lengths varied from 1.0 to 3.5cm for the 30cm thick disturbances. The correlation functions themselves appear to have an exponential character. Corresponding unsteady densities ranged up to 0.5 percent. The aircraft skin acts like an adiabatic wall. A strong shear layer forms at the top edge of aerodynamic fences. Measured correlation lengths are generally smaller (1.0 to 1.5cm) than for TBLs though the strength of the disturbance is significantly greater (0.5 to 3 percent). The majority of the optical path degradation is produced within this thin shear layer. Measured phase variances for the fences varied from 0.08 to 0.26 waves ($\lambda = 0.63\mu$). Degradation is generally independent of fence location over the range of experimental parameters. The existing data base allows scaling these data with wavelength and aperture as well as aircraft parameters such as boundary layer thickness and dynamic pressure. These results suggest that as shorter wavelength laser weapon systems evolve ($\lambda \lesssim 2\mu$), aircraft random flow fields will contribute significantly to the system error budget. Near term adaptive optics is unable to cope with these flows due to their small spatial and large bandwidth requirements. Several passive aerodynamic techniques include using windows with laser turrets to preclude shear layer formation and moving the turret as far forward as possible to minimize boundary layer thickness. Active aerodynamic techniques cry out for creativity, and include suctioning or diverting TBLs and shear layers.