Survey of Validation Data Base for Shockwave Boundary-Layer Interactions in Supersonic Inlets

A. Hamed*
University of Cincinnati, Cincinnati, Ohio 45221
and
J. S. Shang†
Wright Research and Development Center, Wright Patterson Air Force Base, Ohio 45433

The performance of supersonic inlets is strongly affected by the boundary-layer development over its internal surfaces. Boundary-layer bleed is used to suppress separation and to provide the desired inlet performance. The gain in pressure recovery and stability is accompanied, however, with a loss in mass flow and an increase in drag that must be minimized by optimizing the amount of bleed and bleed configuration. The purpose of this work is to review and assess the data base for shock boundary-layer interaction that is pertinent to the flow prediction in supersonic inlets. The first part of the review concerns mixed compression supersonic inlets and their bleed system performance at design and off-design conditions. Based on the assessment of this data, specific areas related to shock wave/boundary-layer interaction bleed, for flow-separation control, are identified. The last part of the review addresses this phenomena in various two- and three-dimensional flow configurations. The effect of bleed in the interaction zone is especially emphasized.

Introduction

ONE of the problems that the designer of a supersonic inlet system faces is how to provide for the highest possible pressure recovery, at the mass flows required by the engine, within the vehicle imposed integration constraints, along with the required stability. When the flight Mach number is above 2.0, the recovery losses associated with the normal shock upstream of the inlet lip become excessive, and a started inlet requires variable geometry. The terminal shock wave in started inlets is stabilized in the diffusing section upstream of the compressor, or fan face, in a conventional installation. The closer to the inlet throat the terminal shock is allowed to move, the higher the inlet recovery. However, along with this benefit comes the increased risk of unstarting the inlet. When the inlet unstarts and the terminal shock wave is expelled, the mass flow rate and pressure recovery are sharply reduced, the drag increases, and a significant transient load is applied to the inlet structure. A large drop in pressure recovery may cause the engine to stall and flame out.[1] Williams described the complex starting process in mixed compression supersonic inlets in Ref. 2.

Unstart can result from an internal disturbance, such as a reduction in the inlet airflow requirement, or from an external disturbance, such as a gust. Bowditch et al.[3] reviewed the test data for 60–40, 40–60, and 20–80 mixed compression inlets to demonstrate the effect of external-internal supersonic area contraction ratio on the unstart limits and distortion. They also discussed the throat bypass system that improves the stability, and the vortex generators that control distortion in the supersonic inlet's internal contraction. Krohn[4] discussed the means for suppressing buzz in supersonic inlets at subcritical operation, and demonstrated experimentally that instabilities triggered by the shock-induced boundary-layer separation on the spike surfaces could be avoided by boundary-layer bleed. Hiller et al.[5] used the results obtained from the different inlets tested during the preceding 10 years to demonstrate that the total pressure recovery, boundary-layer bleed, and cowl drag have the greatest influence on aircraft range.[2-7]

Boundary-layer bleed has been used in supersonic inlets to suppress separation and to improve performance.[2-7] Using the data from different axisymmetric and two-dimensional supersonic inlets designed for flight Mach numbers between 2.5 and 3.5, Bowditch[8] showed that a linear relation exists between the inlet-to-capture-flow ratio and the wetted-to-throat-area ratio. The same data was correlated by Hiller et al.[5] in terms of the percentage of internal compression, which is often used to classify supersonic inlets. The results indicated a linear increase in the amount of bleed, with internal compression. Two-dimensional inlets required 70% more bleed at the same internal compression.

Seyberg and Koncsek[9] discussed the bleed inlet design procedures, the selection of bleed geometry, and the performance characteristics of scoops, slots, and bleed holds. The determination of inlet contours and bleed systems in inlet design procedures are usually based on independent[10-14] or coupled[15] inviscid flow and boundary-layer analyses. The shock/boundary-layer interaction regions are usually handled in a separate analysis,[16] requiring empirical or semiempirical relations.[17] These procedures cannot model the boundary layer in the region of contained normal shocks.[14,16] When the shock was strong enough to cause local separation,[14,16] the computations exhibited discrepancies with experimental measurements in started axisymmetric mixed compressor inlets.[18-20] Large flow separation regions might exist in supersonic inlets at high-pitch or yaw angles.[21] In addition, the strong expelled shock during subcritical operations could cause boundary-layer separation, which alternately choke or unchoke the inlet throat.[22] Algorithms developed for the numerical solution of the unsteady Navier-Stokes equations in supersonic inlets have also been used to predict steady-state[23,24] and oscillatory flows.[25] These procedures are capable, in principle, of modeling strong shock boundary-layer interaction and flow separation. How-

Presented as Paper 89-2939 at the AIAA/ASME/SAE/ASEE 25th Joint Propulsion Conference, Monterey, CA, July 10–12, 1989; received July 18, 1989; revision received April 4, 1990; accepted for publication July 6, 1990.

*Department of Aerospace Engineering and Engineering Mechanics.

†Flight Dynamics Laboratory.

ever, comparisons between steady-state experimental results in supersonic inlets and boundary-layer bleed[26] revealed discrepancies in the prediction of shock location and velocity profiles.[23,24]

The purpose of this work is to review and assess the data base for shock wave boundary-layer interactions that is pertinent to the flow prediction in supersonic inlets. The data base, therefore, will include normal and oblique shock interactions with turbulent boundary layers, but not the shock laminar boundary-layer interaction literature reviewed by Adamson and Messiter.[27] The sources used were determined by the availability of publications.

Supersonic Inlet Performance

The experimental measurements in many of the supersonic inlet studies consisted mainly of the total pressure recovery at the engine face and the static pressure distribution over the inner surfaces. Smeltzer and Sorensen[10,11] tested a mixed-compression axisymmetric inlet model with a traveling centerbody, cowl and centerbody bleed, and bypass, as well as secondary air systems upstream of the engine face and vortex generators downstream of the throat. They measured the effect of altering the bleed and vortex generator configuration on the inlet performance, at the design inlet Mach number of 2.65. Experimental results were also presented at several inlet Mach numbers ranging between 0.8 and 1.4. Wong and Anderson[28,29] conducted an experimental investigation of a large-scale two-dimensional mixed-compression inlet model, with a design Mach number of 3.0, to test its performance at off-design transonic Mach numbers between 0.6 and 1.28. Variable features for off-design operation included an adjustable ramp height system and translating cowl. During transonic operation, boundary-layer bleed was provided only on the ramp and side-wall surfaces, but cowl bleed was eliminated to preclude the possibility of reverse flow through bleed holes. The inlet performance was found to be insensitive to variations in angle of attack and yaw between 0 and 4°. The vortex generators that proved advantageous at design speed were not significantly effective at transonic speeds. Gubbison et al.[30] conducted a cold pipe experimental study to evaluate the effect of bleed flow rates and bleed location on the performance of an axisymmetric mixed-compression inlet with translating centerbody. The inlet was designed for Mach 2.5, with 40% of the supersonic area contraction occurring externally and 60% internally. The best overall performance was obtained with a 2.5% supercritical bleed located just downstream of the last cowl and centerbody shock reflection points in the supersonic diffuser. As the bleed was moved aft, both the overall pressure recovery and subcritical operating range increased. The pressure recovery increased and the distortion decreased with bleed mass flow at critical operation. Large increases in total supercritical bleed flow, however, significantly reduced the subcritical operating range of the inlet. A mission analysis indicated that the downstream bleed leads to a significant improvement in payload over other locations at all operating conditions.

Wong and Hall[31] reported the results of an experimental investigation in which blowing was used to suppress shock-induced separation in a Mach two-scale supersonic inlet. The blowing was applied through a single row of aft-facing discrete jets in the 7° compression ramp to energize the boundary layer upstream of the strong terminal shock. The results indicated that control over a wide range of operating mass flow ratios was possible with a blowing that is 2.5% of the inlet critical flow. The schlieren pictures showed that this was achieved by reducing separation to a very small bubble. In a subsequent study, Wong[32] investigated the effect of increasing the blowing temperature and hole spacing on the inlet performance. Both were found to reduce effectiveness, especially at low inlet mass flows. The hot blowing also produced a temperature distortion at the interface. The cold blowing provided the best near-ideal performance for terminal shock strengths from Mach 1.6 to 2.0.

Fukuda et al.[33,34] tested an axisymmetric inlet at $M = 2.5$, with bleed at the first shock wave boundary-layer interaction on the cowl and the first and second interactions on the centerbody. The measured boundary-layer profiles indicated that across-the-shock bleed should be avoided, since the transformed form factor H_{tr} decreased with bleed mass flow and then reached a plateau, but δ^* and θ continued to increase. Hingst and Johnson[18] presented the results of their experimental measurements in an axisymmetric inlet, with 60% internal supersonic area contraction designed for Mach 2.5, with boundary-layer bleed through rows of normal holes on the cowl and centerbody. The tests were conducted at two Reynolds numbers of 8.2×10^6 and 2.7×10^6 per meter. The measured boundary-layer profiles indicated that the laminar boundary layer separated at the first centerbody shock interaction, but reattached at the measuring location downstream of the shock at the lower Reynolds number. No separation was observed at the higher Reynolds number. Bleed in the aft portion of the centerbody produced reverse flow through the forward holes, resulting in separated boundary layer.

Bleed System Performance

The basis for the analytical development of a bleed system is the incorporation of mass removal in the regions of boundary-layer separation. In order to prevent boundary-layer separation, various criteria are used, such as a limit on the value of the shape factor or the exponent in the boundary-layer velocity profile. Sorensen and Smeltzer[35,36] maintained an exponent of three or greater in the supersonic diffuser and of seven or greater in the throat. Syberg and Koncsek[13] used the criteria that a shape factor of 1.3 corresponds to a "full" velocity profile, of one-seventh power low, whereas a shape factor between 1.8 and 2 indicates a highly distorted profile close to separation. Fukuda et al.[33] based their evaluation of the bleed system performance on the shape factor, which was determined from the boundary-layer profile measurements downstream of the shock boundary-layer interaction-bleed region.

Several factors affect the bleed system performance and, consequently, the boundary-layer development in supersonic inlets, such as the bleed hole geometry, plenum chamber conditions, and shock impact location relative to bleed. Slanting the bleed holes leads to higher bleed pressure recovery, thereby minimizing the momentum drag of the overboard bleed flow. Slant angles of 20° are used in the forward bleed regions, but are increased to 40° in the middle and 90° in the throat region[10,11,35] to provide maximum normal shock stability.[9] The effects of hole size on bleed system performance have been investigated by Fukuda et al.,[33] who reported no significant effect of the hole size. On the other hand, Wong[37] reported significant hole size effects, and recommended hole sizes below the boundary-layer thickness for better bleed performance. The plenum pressure also affects the bleed mass flow, since it determines whether or not the flow in the bleed holes is choked. At present, these effects are accounted for empirically in bleed system designs.[15] Flow recirculation through the bleed holes, which might lead to separation, can be prevented if the plenum chamber is compartmented,[35] which also allows higher pressure recovery in each plenum. Syberg and Koncsek[13] found that the measured overall bleed flow rates were about 20% below the predicted values. Similar accuracy of bleed estimates were reported by Sorensen and Smeltzer[35] with the forward, mid- , and overall cowl bleed 20% below prediction, with better agreement for the centerbody bleed.

Reported boundary-layer measurements in the experimental supersonic inlet studies are not sufficiently detailed to describe the complex shock and expansion wave system that can result from shock/boundary-layer interactions, which are further modified by mass removal.[38] Turbulence measurements have not been reported in any of the inlet studies, although the turbulence structure is strongly influenced by the shock interactions.[39] In addition, the boundary-layer measurements were mostly obtained at design conditions, and, therefore, did not include strong shock boundary-layer interactions leading to separation, which can play a major role in off-design conditions. It is clear from the preceding discussion that an additional data base of shock boundary-layer interaction measurements is required for validating supersonic inlet flow-field predictions. A number of shock boundary-layer experimental studies are reviewed and evaluated (Tables 1 and 2) based on how their experimental results might help in understanding the phenomena described in the preceding discussion.

Effect of Suction on Shock Boundary-Layer Interactions

A number of experimental studies were conducted to investigate the effect of bleed on shock boundary-layer interaction in different geometric configurations. The test arrangements included oblique,[40–45] glancing,[46,47] and conical[48] shocks interacting with boundary layers over flat plates,[40–44,46,47] in compression corners,[45] inside cylinders,[48] and on airfoils.[49] The bleed through holes,[42,46,48] slots,[45,46,51] or porous wall segments[40,41,43,44,51] were implemented upstream,[40,42,48,49] downstream,[42,47] and across[40–46,48,49] the shock (see Tables 1 and 2).

When suction was applied through a porous wall, Strike and Rippy[40] reported that the interaction length decreased and the pressure ratio increased. It was determined that less suction is required to control separation, when applied upstream of the shock. Lee and Leblanc[41] reported that only 60% of the plain wall pressure rise could be recovered with weak suction ($F = -0.0056$) through a perforated wall. For the strong suction ($F = -0.008$), the maximum pressure was within 3% of its ideal value. The measurements indicated a thinner boundary layer with a fuller profile near the wall, in the case of strong suction. The weak suction produced the opposite effect at the first two stations. This was attributed to the surface roughness, which has a stronger influence upstream of the shock, due to the penetration of the surface irregularity outside the laminar sublayer. The static pressure gradient normal to the wall was close to zero, except in the interaction zone effected by the incoming and reflecting shock and expansion waves. Hingst and Tanji[42] determined that bleed eliminated flow separation and produced fuller velocity profiles in the interaction zones. Bleed also shortened the interaction length, and increased the pressure gradient in the interaction zone. Benhachmi et al.[43,44] demonstrated that suction (10% of the boundary-layer mass flow) eliminated the separation, and reduced the three-dimensional effects, as confirmed by off-axis measurements. The effect of bleed was manifested as an increase in both pitot pressure and slope at the wall, with a downstream shift of the shock impingement location. The profiles downstream of the bleed region were much fuller than those upstream. In spite of the wide scatter in the bleed mass flow measurements on the porous plate, the results[44] confirmed previous observations[42] that the bleed distribution follows the trend of the wall pressure.

Tanner and Gai[45] applied bleed through the slot generated by lifting the wedge from the tunnel wall. The data indicated that the optimum bleed height was near four-thirds the boundary-layer thickness before the interaction. Barnhart et al.[46] obtained flow measurements in turbulent boundary layers crossing a glancing side-wall shock wave. The data without suction showed trends toward separation at the $M = 2.5$ and $\alpha = 8$. The upstream influence of the interaction was reduced by 50%, and the flow became quasi-two-dimensional at 30% boundary-layer suction. Toby and Bogdonoff[47] presented the results of an exploratory study of corner bleed on a 90° fin that generated three-dimensional shock wave turbulent boundary-layer interaction at $M = 2.95$. The surface flow visualization and the measured surface pressure distribution

Table 1 Shock wave boundary layer interactions over flat plates and compression corners

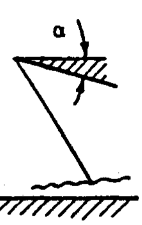
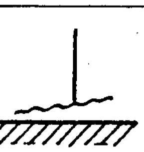
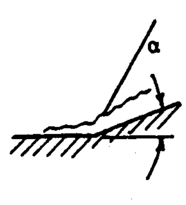
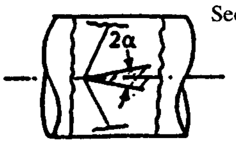
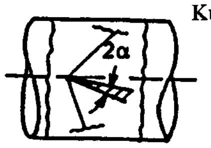
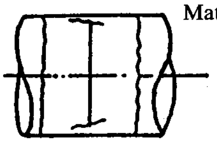
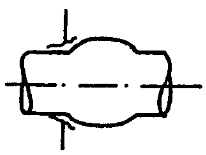
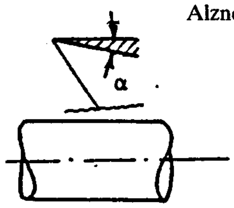
Flow Geometry (2-D)	Reference (No.)	M	$Re \times 10^{-6}$	α, deg	Bleed	Surface measure-ments	Flow measurements Mean	Flow measurements Turb.
	Strike & Rippy[40]	3.0	0.5–10/ft	6.6–11.6	US, AS, porous	P	No	No
	Lee & Leblanc[41]	1.4		3	AS perforated	P	LDV	No
	Hingst & Tanji[42]	2.0	20/m	6 & 6.5	US, DS, AS holes	$\dot{m}_{bl}$	p_{pitot}	No
	Benhachmi[43]	2.5 & 3.0	16.6–18.5/m	0–8	AS porous	$\dot{m}_{bl}$,p	p_{pitot}	No
	Kuehn[50]	1.2–4.4	1–10 (δ)	8–11	No	P	p_{pitot}	No
	Squire & Smith[53]	2.5	2.7/m	2–7	US, holes, blowing	P	p_{pitot}	No
	LeBalleur & Delery[59]	1.92	6.7 (δ)	8.2	No	P	LDV	No
	Voisinet[57]	2.95	0.1–3	—	Porous blowing	P, T, τ	p_{pitot}, hotwire	No
	Delery[68]	1.3–1.45		—	No		LDV	Yes
	Sajben et al.[81]	1.34	.0146 (θ)	—	No		LDV	Yes
	Tanner & Gai[45]	1.95	3.98/ft	8–16	Yes	P	No	No
	Kuehn[50]	1.2–4.4	1–10/ft	12.5–30	No	P	p_{pitot}	No
	Law[60]	2.96	0.1–1 (δ)	15–25	No	P, oil flow	No	No
	Roshko & Thomke[61]	2–5	4.8–20.9/ft	5–27	No	P	No	No
	Ardonceau[65,66]	2.25	1.1/m	8–18	At 18	P	Hotwire & LDV	Yes
	Muck & Smits[83]	2.87	0.776 (δ)	0–16	No	No	Hot wire	Yes
	Hayakawa et al.[84]	2.85	73/m	8	No	No	Hot wire	Yes
	Hayakawa et al.[85]	2.87	7.76 (θ)	8–24	At 20 & 24	No	Hot wire	Yes
	Debieve et al.[87]	2.32	0.0042 (θ)	6	No	P	Hot wire	Yes
	Settles et al.[89]	2.85	73/m	8.24	No	P	Hot wire	No
	Dolling & Murphy[91]	2.9 & 2.95	0.78–1.4 (δ)	24	No	P	Trans-ducers	No

Table 2 Shock wave boundary layer interactions over cylindrical surfaces

Flow Geometry (3-D)	Reference (No.)	M	$Re \times 10^{-6}$	α, deg	Bleed	Surface measure-ments	Flow measurements: Mean	Flow measurements: Turb.
	Seebough & Childs[48]	2.82, 3.78	5.6/ft	10–15	US, AS, holes	P	p_{pitot}	No
	Kussoy et al.[71]	2.2	12–110/m	—	No	p, τ Flow Direction	p_{pitot} Hot film	Yes
	Mateer et al.[82]	0.48–1.32	2.25–8.5 (δ)	—	No	p, τ Flow Direction	p_{pitot}, P Hotwire	No
	Buchalo & Johnson[70]	0.425–0.925	6–20	—	No	P	Turb. Rey, stress Mean Vel. Field	—
	Alzner & Zakkay[56]	6	200	10	Injection	P, T	p_{pitot}	No

indicated a change in the streamwise extent of the interaction, with no significant change in the flow structure, as the apex of the fin was lifted from the surface into the supersonic flow within the boundary layer.

For axisymmetric flow in the interaction region of the boundary layer inside a 2.03 in. diam duct, Seebough and Childs[48] determined the approximate separation and reattachment points by plotting the Mach number variation vs the axial coordinate at fixed distances from the wall, and extrapolating for the intersections with $M = 0$. A comparison of the estimated flow deflection angles for incipient separation with the two-dimensional results of Kuehn[50] revealed that separation occurred at lower angles for conical shocks. It was possible to remove observable effects of separation for the 15° cone at $M = 3.78$ by bleeding 3.1% of the upstream boundary-layer mass flow through one row of 0.064 in. diam bleed holes. Bleeding reduced the length of the interaction zone (the extent of the surface static pressure rise) by 40%. Contrary to the conclusions of Strike and Rippy,[40] suction within the interaction region was found to be more effective in suppressing the effects of separation. Four rows of bleed holes were required to obtain equal suction flow rate, to suppress separation, when the perforation was located upstream of the pressure rise.[51]

Thiede et al.[49] conducted an experimental study to improve the off-design performance of the VFW VA-2 supercritical airfoil through boundary-layer suction and ventilation. Suction was found to delay shock-induced separation to greater shock strengths, and to stabilize the shock in a rearward position. The improvement in off-design lift was slightly less for passive control, but it offered potential for a large drag reduction, up to 40%, due to the smearing of the shock wave pressure gradient. This strong passive effect was caused by the secondary flow through the plenum underneath the shock, leading to forward blowing and rearward suction. The net result was a partial pressure compensation in the double slot, and perforated strip configuration without any suction.

Smits and Wood concluded in their review[52] that experimental investigations of the effect of suction are influenced by surface roughness and departures from two-dimensionality. They also indicated that skin friction measurements in most of the studies are inadequate.

Effect of Roughness and Blowing on Shock Boundary-Layer Interaction

The reason for including these effects in the present review is that sometimes, when bleed holes or porous wall segments extend across the shock impingement location, zero suction and even blowing may result, due to flow recirculation inside the bleed chamber. The unintended mass exchange can occur upstream of the shock, due to the large surface pressure difference. Both blowing and roughness produce somewhat similar influence on the boundary-layer velocity profile; this influence is quite different from the effect of bleed.

Squire and Smith[53] reported that the interaction length increased and the start of the interaction moved forward with blowing, upstream of the interaction between the boundary layer on a flat plate and an oblique shock generated by a ramp. The peak of the pressure distribution tended to decrease and move forward with increased blowing rates, and the downstream pressure approached the inviscid levels. The velocity profiles after the blowing, but before the shock impact, were found to become thicker and less full. The ratio of the thickness of the subsonic layer to the boundary-layer thickness was found to increase with blowing, and the interaction length was found to correlate with the thickness of the laminar sublayer at the end of the interaction. Some three-dimensional effects were observed from the surface flow visualization. Oil flow patterns indicated that separation occurred just above $\alpha = 5°$ for all blowing rates, suggesting that the shock strength for incipient separation is insensitive to blowing.

Yanta et al.[54] obtained boundary-layer measurements on 8 deg sphere cones, with mass addition at Mach number of 2.5 and Reynolds number of 8.86×10^6/m. The longitudinal and transverse velocity profiles, as well as the turbulence intensity and Reynolds stress distributions, were deduced from the two-dimensional LDV measurements. The results were presented for four different injected air flow rates corresponding to $F = \rho_w v_w / \rho_\infty U_\infty$ of 0, 0.2, 0.5, and 1.0. The longitudinal velocity profiles were found to become less full with increased blowing, and, at $F = 1.0$, the boundary layer appeared to be blown off the model. Linear regression methods indicated that the value of n in the power low for each profile ranged between 5 for no blowing and 1 for the highest blowing rate. The longitudinal turbulence intensities increased and the maximum moved away from the wall with increased blowing rates, as in the case of incompressible flow.[55] The Reynolds stress was found to increase significantly with increased mass injection in the boundary layer's midsection, even though it was found to decrease in the inner part near the wall.

Alzner and Zakkay[56] investigated the effect of injecting air and hydrogen on the surface pressure and heat transfer in shock wave boundary-layer interactions with flow separation at Mach 6. The shock generated by an axisymmetric ring with a wedge angle of 10° impinged on the boundary layer over a centerbody, and the tangential injection was applied through a slot. A range of injection mass flow rates corresponding to $\rho_j U_j / \rho_e U_e$ between 0.0205 and 2.74 were investigated at three positions relative to the interactions. These positions corresponded to location upstream of the separated flow region, near the detachment point and near the attachment point. In all tests, the basic surface pressure distribution prevailed and, in comparison to the case of no injection, the length of the separation region showed little change. The most effective reduction in the surface peak heat transfer was achieved through injection near the reattachment point, with hydrogen injection resulting in higher cooling effectiveness because of its higher specific heat and lower molecular weight. The shadowgraphs indicated that, with increased injection rates, the separation shock moved upstream until it reached a stationary position. The recompression shock location was initially unchanged, then moved upstream and away from the wall. The pitot and static pressure measurements indicated that the separated region was split into two, with a small separation bubble downstream of the injection slot, which was attributed to the nontangency of the injection.[56] This was confirmed with a zero velocity line deduced from the crossover point of the pitot pressure measurements, with upstream and downstream facing probes.

Viosinet[57] studied the combined effect of roughness and blowing on the turbulent skin friction in the boundary layer on a flat plate at $M = 2.9$. The systematic variation of roughness in the experiments ranged between aerodynamically smooth to full rough regimes ($K = 0.0$–0.49). The blowing rates through the rough porous wall ranged from zero up to and including boundary-layer blow-off. Both roughness and blowing caused the boundary-layer profile to be less full, and increased the boundary layer thickness. The results with and without blowing were found to correlate well with the incompressible flow data after the compressibility transformation.

Smits and Wood[52] discussed some of the experimental studies of boundary layers with a step change in wall roughness. For the Mach 6 flow, Berg's measurements[58] indicated relaxation distances for the mean flow properties of ten and fourteen times the initial boundary-layer thickness, for rough to smooth and smooth to rough, respectively. The relaxation distances for the fluctuating properties were about one and one-half times as long. Benhachmi[43] presented the measured surface static pressure and pitot pressure profiles for supersonic flow over a flat plate with a rough wall segment with and without an oblique shock. Roughness was found to have only a small effect on the pitot pressure very close to the wall. The roughness caused a decrease in the slope at the wall; the slope then increased slightly after the transition back to smooth wall.

Flow Separation in Shock Boundary-Layer Interactions

Strong shock boundary-layer interactions leading to flow separation have been investigated experimentally over a wide range of Mach and Reynolds numbers. The tests were conducted in various configurations, including two-dimensional flows resulting from an oblique shock interacting with a flat plate boundary layer,[50,59,60] compression corner,[50,60,61,65–67] near normal shocks in transonic flows inside symmetric[68] and nonsymmetric[69] channels, axisymmetric flow at the junction of a bump with a cylinder,[70] and three-dimensional shock boundary-interaction in different configurations[71,72] (see Tables 1 and 2).

Delery[39] gave a detailed description of the flow and the shock system configurations in separated shock wave boundary-layer interactions. Kuehn[50] conducted the first extensive study to determine the incipient separation conditions in shock wave turbulent boundary-layer interactions in compression corners, curved surfaces of various radii, and incident waves. He demonstrated the equivalence between the overall pressure ratio for incipient separation in the compression corner and incident shock. Law[60] compared the experimental results of an oblique shock-flat plate boundary-layer interaction and a compression corner. He found that the incident shock wave configuration resulted in a larger length of separation approximately 15% above that in the compression corner for the same overall pressure ratio. The incipient separation angle was approximately the same for both configurations, and increased with the Reynolds number. Law's[60] results did not show the constant pressure plateau associated with large separation lengths[73] that are greater than seven boundary-layer thicknesses. The location of the separation point was determined using three techniques, namely, the first inflection point in the static pressure distribution, the location of oil accumulation, and the extrapolation of the separation shock wave to the flat plate surface. The three techniques were in reasonably good agreement for large regions of separation, but were inadequate or gave questionable results for small regions of separation.

Ardonceau[65,66] presented the results of a very detailed study of the mean flow and turbulence at attached, incipiently separated, and separated flow conditions. The presented results for the isobar maps[65] showed the factorization of the pressure waves outside the boundary layer. The inclination of the isobars was small near the wall, but increased at the outer edge of the boundary layer, and the influence of compression extended far upstream in the inner part of the boundary layer. The comparison between the hot wire and the LDA mean velocity data for the incipiently separated flow (at 13° corner angle) revealed discrepancies near the wall. The higher LDA velocities were attributed to the lower seeding level near the wall.[65,66] The accuracy of the LDA-measured longitudinal component of the mean velocity, which was presented for the three compression angles, is also questionable inside and near the separation bubble.

Bachalo and Johnson[70] observed larger separation regions at higher Mach numbers in axisymmetric transonic flows.

Kussoy et al.[71] studied experimentally the interaction of the boundary layer inside a cylinder with a shock that was generated by a cone inclined to the axis inside the tube. The resulting three-dimensional separated boundary layer included a significant region of streamwise recirculation. The yaw angle contours indicated a significant flow turning in a relatively small fluid layer close to the wall (15% of the boundary-layer height). The results for the surface static pressure and surface skin friction indicated a dramatic circumferential variation in the skin friction. The results did not show a pressure plateau, which is often used to imply the existence of streamwise separation. The static pressure contours in the windward plane showed strong streamwise and normal pressure gradient in the interaction zone, with peak pressure near the confluence of the incident and separation shock. Peak turbulent kinetic energy and turbulent shear stress were also present in the same region, whereas the turbulent kinetic energy varied only slightly on the leeward plane. The wall static pressures, the three color LDV measurements, and the flow visualization of Benay et al.[72] revealed a very complex flow pattern, and the existence of a number of separation lines, in a transonic channel with a hump on one wall.

Effect of Reynolds Number

It is important to understand the effect of the Reynolds number, if the flow measurements in inlet experiments are to be used in actual inlet design. The Reynolds numbers in these tests are usually different from in-flight conditions, because of the smaller test models and the test pressures and temperatures not corresponding to flight conditions. Law[60] reported that the ratio between the separation length and the undisturbed boundary-layer thickness just upstream of the separation was found to increase with decreasing Reynolds number. Results obtained by other investigators in a compression corner exhibited various trends, ranging between decreasing[74] and constant[75,76] normalized separation length with increasing Reynolds number.

Kuehn[50] plotted the incipient overall pressure ratio and the compression corner angle vs the Reynolds number based on the boundary-layer thickness before the interaction for the different Mach numbers. His results indicated that the resistance to separation decreased with the increase in Mach and Reynolds numbers. Roshko and Thomke[61] investigated the conditions for incipient separation at *Re* that are two orders of magnitude higher than in Kuehn's experiment, and documented a trend reversal, showing increased resistance to separation with the Reynolds number. Thomke[61] explained the trend reversal with the Reynolds number through an inviscid rotational analysis of the supersonic outer portion of the boundary layer in the shock interaction region. Elfstrom,[62] using the inviscid roational analysis model of Roshko and Thomke,[61] computed the incipient separation angle over a wide range of Mach, Reynolds numbers, and surface cooling conditions, and reported very close agreement with a large number of experimental measurements. Hamed[63] discussed in detail the large number of incipient separation studies, and the effect of different flow separation detection methods on the results.

The Reynolds number also affects the surface pressure and friction in unseparated flows. Mateer and Viegas[64] reported that increasing the Reynolds number steepened the pressure gradient across the normal shock, but had little effect on the friction coefficient. On the other hand, the results indicated that, as the Mach number increased, the pressure gradient at the shock decreased. The minimum value of friction coefficient also decreased, and the location of minimum friction moved downstream with increased Mach number.

Effect of Shock Boundary-Layer Interaction on Turbulence

Rubesin[77] presented an excellent review and assessment of the comparative performance of various turbulence models, for several examples of external and one internal flowfield. It has been shown in external flow computations that the turbulence model affects the predicted shock location,[69,78] the extent of separation[69,78,79] and whether or not separation is predicted. Turbulence measurements in shock boundary-layer interactions were reported in a small number of experimental investigations for a normal shock on a flat plate,[80] followed by an adverse pressure gradient,[81] inside a cylinder,[82] in a supersonic compression corner,[65,66,83–87] and over a cylinder with flare[88] (see Tables 1 and 2).

Delery[80] presented the turbulence kinetic energy and the Reynolds stress at several locations across the shock boundary-layer interaction zone in a symmetric channel at three transonic Mach numbers corresponding to incipient separation, separated flow with a large separation bubble. Both quantities increased an order of magnitude in the interaction, and then started to decrease near reattachment. The maximums were detached from the wall, and the Reynolds stress maximum was at the same location of maximum mean velocity slope. A strong anisotropy developed with $|u'| \approx 4|v'|$ even before the shock impact. Delery concluded that, in separated flows, the normal stress plays an important role and must be modeled in both the turbulence production and momentum equations.

Sajben et al.[81] investigated experimentally the flow in a normal shock/turbulent boundary-layer interaction followed by an adverse pressure gradient produced by a 5° divergence angle of the bottom wall. The velocity contours indicated a rapid thickening of the boundary layers at the shock, and the presence of a high-velocity overshoot region. The loci of the maxima for the kinematic shear stress $\langle -u'v' \rangle$ and the turbulent kinetic energy κ were virtually coincident, and, initially, occurred very close to the wall, then gradually migrated outward to the middle of the boundary layer. The maximum kinetic energy was found to peak near the shock, whereas the peak in the maximum shear stress appeared about 50 boundary layers further downstream. The distributions were similar to those reported in Ref. 80, but were only one-third in magnitude. Plots of $\hat{v}/\hat{u}$ revealed regions of large anisotropy (low values of the ratio) associated with the regions of velocity overshoots. Low $\hat{v}/\hat{u}$ values were also reported in the freestream portion of the shock, which were attributed to shock oscillations. Time mean velocity, surface oil flow, and schlieren visualization all indicated the absence of separation. The streamwise velocity component in the LDA data was negative 8% of the time, indicating a state of incipient detachment in a small region close to the wall, slightly downstream of the shock. The turbulence measurements of Mateer et al.[82] indicated an order of magnitude increase in the fluctuation, followed by relaxation towards constant levels far downstream. The downstream values were independent of the radial position, with double the intensity of the upstream core.

The measurements of Muck and Smits[83] and Hayakawa et al.[84,85] indicated that the turbulence structure was considerably distorted by the interaction, and the relaxation process was very slow. The *rms* mass flow fluctuation level $\langle (\rho u)' \rangle$ doubled in the interaction and continued to increase in the recovery zone to 2.8 times its initial value. The mass weighted shear stress $\overline{(\rho u)'v'}$ increased by as much as 16 times through the interaction zone, then decreased significantly further downstream. The turbulence structure was strongly disturbed even when the surface skin friction and pressure measurements[89] indicated little or no separation at $\alpha = 16°$. Hayakawa et al.[84–86] presented the evolution of the turbulence intensities along three selected streamlines across the interaction and recovery zone. A rapid increase in $\overline{u'^2}$ was displayed along all the streamlines in the interaction zone of severe pressure gradients. The increase was greater for the higher compression angles. The behavior of the turbulence intensity in the recovery zone downstream was found to depend on the streamline distance from the wall. The turbulence intensities decayed rapidly near the wall but continued to increase in the outer part, indicating that the large eddies in the outer part of the boundary layer respond more slowly than the motions near the wall.

The turbulence measurements of Ardonceau[65,66] demonstrated that both $\langle u' \rangle$ and $\langle v' \rangle$ increased with longitudinal distance and spread towards the outer part of the boundary layer, with a maximum at 3.4 mm away from the wall. The influence of the compression corner was noticeable at a distance of 12 mm upstream of the hinge line, and the fluctuation level was still two to three times that in the undisturbed boundary layer, at $x = 52$ mm downstream. The longitudinal component reacted more strongly than the normal component, particularly in the inner part of the boundary layer. The ratio between $\langle u' \rangle$ and $\langle v' \rangle$ became higher than the upstream equilibrium value in the interaction region, and returned to a value of 2 downstream. The Reynolds shear stress $\overline{u'v'}$ also increased considerably through the interaction, and the correlation coefficient $\overline{u'v'}/\langle u' \rangle \langle v' \rangle$ was considerably modified in the vicinity of the separation. A study of the sensitivity of the *rms* mass flux data from the hot wire measurements[63] indicated strong dependence in the Mach number range $0.6 < M < 1.4$. This would indicate low confidence in the hot wire's turbulence data in this range, since such influence was not included in the data regression procedure. Debieve et al.[87] indicated a 50% increase in $\overline{u'^2}$, along a mean mass weighted streamline at $y/\delta = 0.4$.

Brown et al.[81] obtained their turbulent measurements using LDV in the boundary layer at the junction of a 30° half-angle conical flare inclined at $\alpha = 0$, 5 and 10° to the 5.06 diam cylinder axis at $M = 2.85$ and $Re = 0.6 \times 10^6$/m. The surface pressure measurements indicated that the separation length increased, and the start of the separation moved upstream with increased α. At the plane of symmetry, the maximum turbulence kinetic energy amplified eight to nine folds, even prior to separation. The magnitude of the overall rise was uneffected by geometry, but the first increase moved upstream with α. Scaling the axial distance by the interaction length collapsed the turbulence data in the three shock strengths.

In the case of three-dimensional shock boundary-layer interaction inside a cylinder with an inclined shock generating cone, Kussoy et al.[69] found the turbulent kinetic energy to vary only slightly from the equilibrium upstream value in the leeward plane. On the other hand, strong streamwise and normal variations in the turbulent kinetic energy were measured in the windward plane. Peak turbulent shear stress and kinetic energy appeared in the region of the convergence of the incident and separation shock, and above the separation bubble in the windward plane. Negative turbulent shear stress was reported in the vicinity of the separation bubble, in a region where du/dy is positive; however, the authors indicated that the reduction technique of the shear stress from the hot wire measurement might be responsible.[90]

Dolling and Murphy[91] reported large amplitude fluctuations throughout the interaction, particularly near the separation and reattachment points. Hayashi et al.[92] measured the surface temperature fluctuations in the interaction of an incident shock with a turbulent boundary layer. They reported a sharp peak in surface temperature fluctuations at the separation point, similar to Dolling and Murphy's pressure fluctuation peak. The highly unsteady shock wave structure generated intermittency in the wall pressure[91] and temperature.[92] Holden[93] also reported large surface pressure fluctuations at hypersonic speeds (70% of the mean value), close to the separation and reattachment points. He determined that the separation point oscillates in the streamwise direction at 1–10 KHz frequencies and $\delta/4$–$\delta/3$ amplitude, and defined separation as where the time average of the surface shear was zero at one point on the surface.[94] Mikulla and Horstman[95] proposed a coupling between the turbulent energy and the separation bubble unsteadiness, and supported their hypothesis with the detailed power spectra distributions of the measured fluctuating quantities. Although this problem still needs further investigation, it is clear that a laminar sublayer model is unrealistic in separated shock wave turbulent boundary interactions.

Summary

The review of available experimental data related to the control of shock boundary-layer interaction through bleed in supersonic inlets can be summarized as follows:

1) Experimental data for surface and pitot pressure is available for a number of configurations, and demonstrates that the boundary-layer separation can be controlled.

2) Local bleed distribution measurements are very scarce in these experiments, which leads to difficulty in using the data for code assessment.

3) Some contradictions exist between the results of different investigations regarding the effectiveness of bleed configuration (upstream, downstream, or across the shock) in controlling separation.

4) Additional systematic data on unchoked bleed hole efficiency and recovery is needed in order to simulate plenum conditions in bleed systems at off-design performance.

5) Turbulence measurements indicate significant changes in the turbulence across the shock boundary-layer interaction, especially if the flow separates. No turbulence data is available in interactions with bleed.

6) The flow is highly unsteady in the separated region, and the separation point oscillates in the streamwise direction, with possible coupling between the turbulent energy and the separation bubble unsteadiness.

7) Large experimental uncertainties are associated with the data reduction techniques of pitot probes and hot wire data under shock wave boundary-layer interaction conditions.

Acknowledgment

This work was partially supported by Universal Energy Systems Contract No. F49620-88-C-0053/SB5581-0378.

References

[1]Sugiyama, Y., Tabakoff, W., and Hamed, A., "J85 Surge Transient Simulation," *Journal of Propulsion and Power*, Vol. 5, No. 3, 1989, pp. 375–381.

[2]Williams, J., "Aircraft Performance Considerations for Noise Reduction,"AGARD LS-67, May 1974, pp. 12–14.

[3]Bowditch, D. N., Coltrin, R. E., Sanders, B. W., Sorensen, N. E., and Wasserbauer, J. F., "Supersonic Cruise Inlets," *Aircraft Propulsion*, NASA SP-259, Nov. 1970, pp. 283–312.

[4]Krohn, E. O., "Inlet Buzz in Ramjets and Its Suppression," 8th International Symposium on Air Breathing Engines, *AIAA Symposium Vol.*, June 1987, pp. 568–578.

[5]Hiller, W. H., Batterton, P. G., Burstadt, P. L., and Szuch, J. R., "Propulsion Systems Technology," *Aeronautical Propulsion*, NASA SP-381, May 1975, pp. 387–440.

[6]Imfeld, W. F., "Development Program for the F-15 Inlet," *Journal of Aircraft*, Vol. 13, No. 4, 1976, pp. 268–291.

[7]Rettie, I. H., and Lewis, W. G. E., "Design and Development of an Air Intake for a Supersonic Transport Aircraft," *Journal of Aircraft*, Vol. 5, No. 6, 1968, pp. 513–521.

[8]Bowditch, D. N., "Some Design Considerations for Supersonic Cruise Mixed-Compression Inlets," AIAA Paper No. 73-1269, Nov. 1973.

[9]Syberg, J., and Koncsek, L., "Bleed System Design Technology for Supersonic Inlets," AIAA Paper No. 72-1138, Nov. 29–Dec. 1 1972.

[10]Sorensen, N. E., Smeltzer, D. B., and Latham, E. A., "Advanced Supersonic Inlet Technology," *Journal of Aircraft*, Vol. 10, Oct. 1973, pp. 278–282.

[11]Smeltzer, D. B., and Sorensen, N. E., "Analytical and Experimental Performance of Two Isentropic Mixed Compression Axisymmetric Inlets at Mach Numbers 0.8 to 2.65," NASA TN-D-7320, June 1973.

[12]Sorensen, N. E., and Smeltzer, D. B., "Performance Estimates for a Supersonic Axi-Symmetric Inlet System," *Journal of Aircraft*, Vol. 9, Sept. 1972, pp. 703–706.

[13]Syberg, J., and Koncsek, J. L., "Experimental Evaluation of an

Analytically Derived Bleed System for a Supersonic Inlet," *Journal of Aircraft*, Vol. 13, Oct. 1976, pp. 792–797.

[14]Hingst, W. R., and Towne, C. E., "Comparison of Theoretical and Experimental Boundary Layer Development in a Mach 2.5 Mixed Compression Inlet," NASA TM-X-3026, April 1974.

[15]Reyhner, T. A., and Hickox, T. E., "Combined Viscous Inviscid Analysis of Supersonic Inlet Flow Fields," *Journal of Aircraft*, Vol. 9, No. 8, 1972, pp. 589–595.

[16]Seebaugh, W. R., Paynter, G. C., and Childs, M. E., "Shock Reflection From a Turbulent Boundary Layer with Mass Bleed," *Journal of Aircraft*, Vol. 5, No. 5, 1968, pp. 461–467.

[17]Pinckney, S. Z., "Semiempirical Method for Predicting Effects of Incident-Reflecting Shocks on the Turbulent Boundary Layers," NASA TN-D-3029, Oct. 1965.

[18]Hingst, W. R., and Johnson, D. F., "Experimental Investigation of Boundary Layers in an Axisymmetric, Mach 2.5 Mixed Compression Inlet," NASA TM-X-2903, Oct. 1973.

[19]Smeltzer, D. B., and Sorensen, N. E., "Investigation of a Nearly Isentropic Mixed-Compression Axisymmetric Inlet System at Mach Numbers 0.6 to 3.2," NASA TN-D-4557, May 1968.

[20]Koncsek, J. L., and Seyberg, J., "Transonic and Supersonic Test of a Mach 2.65 Mixed Compression Axisymmetric Intakes," NASA CR-1977, March 1972.

[21]Surber, L., and Sedlock, P., "Effects of Airframe-Inlet Integration or Half-Axisymmetric and Two-Dimensional Supersonic Inlet Performance," AIAA Paper 78-960, July 1978.

[22]Martin, A. W., Koslin, L. C., and Sidney, D. M., "Dynamic Distortion at the Exit of a Subsonic Diffuser of a Mixed Compression Inlet," NASA CR-1644, Dec. 1970.

[23]Knight, D. D., "Calculation of High Speed Inlet Flows Using the Navier-Stokes Equations, Vol. I: Description of Results," Air Force Flight Dynamics Lab-TR-79-3138, Feb. 1980.

[24]Knight, D. D., "Improved Calculation of High Speed Inlet Flows. Part I Numerical Algorithm," *AIAA Journal*, Vol. 19, No. 1, 1981, pp. 34–41, "Part II: Results," *AIAA Journal*, Vol. 19, No. 2, 1981, pp. 172–179.

[25]Newsome, R. W., "Numerical Solutions for Steady and Unsteady Oscillatory Flow About an Axi-Symmetric Inlet," Air Force Wright Aeronautical Labs-TR-83-3058, April 1985.

[26]Carter, T. D., and Spong, E. D., "High Speed Inlet Investigation, Vol. I: Description of Program and Results; Vol. II: Data Summary," Air Force Flight Dynamics Lab-TR-77-105, Nov. 1977.

[27]Adamson, T. C., and Messiter, A. F., "Analysis of Two-Dimensional Interactions Between Shock Waves and Boundary Layers," *Annual Review of Fluid Mechanics*, Vol. 12, 1980, pp. 103–138.

[28]Anderson, W. E., and Wong, N. D., "Experimental Investigation of a Large-Scale, Two-Dimensional, Mixed Compression Inlet System—Performance and Design Conditions, $M_\infty = 3.0$," NASA TM-X-2016, May 1970.

[29]Wong, N. D., and Anderson, W. E., "Experimental Investigation of a Large-Scale, Two-Dimensional, Mixed-Compression Inlet System—Internal Performance and Drag at Transonic Conditions, $M_\infty = 0.6$ to 1.28," NASA TN-D-7445, Oct. 1973.

[30]Gubbison, R. W., Meleason, E. T., and Johnson, D. F., "Performance Characteristics from Mach 2.58 to 1.98 of an Axisymmetric Mixed Compression Inlet System with 60 Percent Internal Contraction," NASA TM-X-1739, Feb. 1969.

[31]Wong, W. F., and Hall, G. R., "Suppression of Strong Shock Boundary Layer Interaction in Supersonic Inlets by Boundary Layer Blowing," AIAA Paper No. 75-1209, Sept./Oct. 1975.

[32]Wong, W. F., "Application of Boundary Layer Blowing to Suppress Strong Shock Induced Separation in Supersonic Inlets," AIAA Paper No. 77-147, Jan. 1977.

[33]Fukuda, M. K., Hingst, W. G., and Reshotko, E., "Control of Shock Boundary Layer Interactions by Bleed in Mixed Compression Inlets," NASA CR-2595, 1975.

[34]Fukuda, M. K., Hingst, W. G., and Reshotko, E., "Bleed Effects on Shock/Boundary-Layer Interactions in Supersonic Mixed Compression Inlets," *Journal of Aircraft*, Vol. 14, No. 2, 1977, pp. 151–156.

[35]Sorensen, N. E., and Smeltzer, D. B., "Performance Estimates for a Supersonic Axisymmetric Inlet System," *Journal of Aircraft*, Vol. 9, No. 10, 1972, pp. 703–706.

[36]Smeltzer, D. B., and Sorensen, N. E., "Investigation of a Mixed Compression Axisymmetric Inlet System at Mach Numbers 0.6 to 3.5," NASA TN-D-6078, 1970.

[37]Wong, W. F., "The Application of Boundary Layer Suction to Suppress Strong Shock-Induced Separation in Supersonic Inlets," AIAA Paper No. 74-1063, Oct. 1974.

[38]Delery, J., and Marvin, J. G., "Shock-Wave Boundary Layer Interactions," AGARD AG-280, Feb. 1986.

[39]Delery, J. M., "Shock Wave/Turbulent Boundary Layer Interaction and Its Control," *Progress in Aerospace Sciences*, Vol. 22, No. 4, 1985, pp. 209–280.

[40]Strike, W. T., and Rippy, J., "Influence of Suction on the Interaction of an Oblique Shock with a Turbulent Boundary Layer at Mach 3," Arnold Engineering Development Center-TN-61-129, Oct. 1961.

[41]Lee, D. B., and Leblanc, R., "Interaction onde de Choc Oblique-Couche Limite sur Paroi Poreuse avec Aspiration," Paper 23, AGARD CP-365, "Improvement of Aerodynamic Performance Through Boundary Layer Control and High Lift Systems," Aug. 1984.

[42]Hingst, W. R., and Tanji, F. T., "Experimental Investigation of Two-Dimensional Shock-Boundary Layer Interaction with Bleed," AIAA Paper 83-0135, 1973; also, NASA TM-83057, 1973.

[43]Benhachmi, C., "Shock Wave/Turbulent Boundary Layer Interaction with Continuous Suction," Ph.D. Thesis, Case Western Reserve Univ., Cleveland, Ohio, Aug. 1988.

[44]Benhachmi, D., Greber, I., and Hingst, W., "Experimental and Numerical Investigation of an Oblique Shock-Wave/Turbulent Boundary Layer Interaction with Continuous Suction," AIAA Paper 89-0357, Jan. 1989.

[45]Tanner, L. H., and Gai, S. L., "Effects of Suction on the Interaction Between Shockwave and Boundary Layer at a Compression Corner," Aeronautical Research Council CP-No. 1087, 1970.

[46]Barnhart, P. J., Greber, I., and Hingst, W. R., "Glancing Shock Wave-Turbulent Boundary Layer Interaction with Boundary Layer Suction," AIAA Paper No. 88-0308, Jan. 1988.

[47]Toby, A. S., and Bogdonoff, S. M., "An Exploratory Study of Corner Bleed on a Fin Generated Three-Dimensional Shock Wave Turbulent Boundary Layer Interaction," AIAA Paper 89-0356, Jan. 1989.

[48]Seebaugh, W., and Childs, M., "Conical Shock Wave Boundary Layer Interaction Including Suction Effects," *Journal of Aircraft*, Vol. 7, No. 4, 1970, pp. 334–340.

[49]Thiede, P., Krogmann, P., and Stanewsky, E., "Active and Passive Shock/Boundary Layer Control on Supercritical Airfoils," AGARD CP-365, "Improvement of Aerodynamic Performance Through Boundary Layer Control and High Lift Systems," Paper No. 24, Feb. 1985.

[50]Kuehn, D. M., "Experimental Investigation of the Pressure Rise Required for the Incipient Separation of Turbulent Boundary Layers in Two Dimensional Supersonic Flow," NASA Memo 1-21-59A, Feb. 1959.

[51]Seebaugh, W. R., "An Investigation of the Interaction of a Shock Wave and Turbulent Boundary Layers in Axially Symmetric Internal Flow Including the Effects of Mass Bleed," Ph.D. Thesis, Univ. of Washington, Pullman, WA, 1968.

[52]Smits, A. J., and Wood, P. H., "The Response of Turbulent Boundary Layers to Sudden Perturbations," *Annual Review of Fluid Mechanics*, Vol. 17, 1985, pp. 321–358.

[53]Squire, L. C., and Smith, M. G., "Interaction of a Shock Wave with a Turbulent Boundary Layer Disturbed by Injection," *Royal Aeronautical Society*, May 1980, pp. 85–110.

[54]Yanta, W. J., Collier, A. S., and Smith, T. S., "Measurements of a Supersonic Turbulent Boundary Layer with Mass Addition," AIAA Paper 89-0135, Jan. 1989.

[55]Collier, F. S., and Schetz, J. A., "Injection into Turbulent Boundary Layer Through Different Porous Surfaces," *AIAA Journal*, Vol. 22, No. 6, 1984, pp. 839–841.

[56]Alzner, E., and Zakkay, V., "Turbulent Boundary Layer Shock Interactions With and Without Injection," AIAA Paper No. 70-91, Jan. 1970.

[57]Voisinet, R. L. P., "Combined Influence of Roughness and Mass Transfer on Turbulent Skin Friction at Mach 2.9," AIAA Paper No. 79-0003, Jan. 1979.

[58]Berg, D. E., "Surface Roughness Effects on the Hypersonic Turbulent Boundary Layer," Ph.D. Thesis, California Inst. of Tech., Pasadena, CA, 1977.

[59]LeBalleur, J. C., and Delery, J., "Etude experimentale de l'interaction onde d'une de chock plane et d'une couche limite turbulente a $M_\infty = 1.92$," ONERA Report No. 4/7078, Oct. 1970.

[60]Law, C. H., "Supersonic Shock Wave Turbulent Boundary Layer Interactions," *AIAA Journal*, Vol. 14, No. 6, 1976, pp. 730–734.

[61]Roshko, A., and Thomke, G. J., "Supersonic Turbulent Boundary Layer Interaction with a Compression Corner at Very High Reynolds Number," *Proceedings of the 1969 Symposium on Viscous Interactions Phenomena in Supersonic and Hypersonic Flows*, University

of Dayton Press, Dayton, OH, 1964, pp. 109–133.

[62]Elfstrom, G. M., "Turbulent Hypersonic Flow at a Wedge-Compression Corner," *Journal of Fluid Mechanics*, Vol. 53, Part 1, May 1972, pp. 113–127.

[63]Hamed, A., "Flow Separation in Shock Wave-Boundary Layer Interactions," NASA CR-4274, Feb. 1990.

[64]Mateer, G. G., and Viegas, J. R., "Mach and Reynolds Number Effects on a Shock-Wave/Boundary-Layer Interaction," AIAA Paper 79-1503, July 1979; *AIAA Journal*, Vol. 18, No. 8, 1980, pp. 1016–1018.

[65]Ardonceau, P. L., "The Structure of Turbulence in a Supersonic Shock-Wave/Boundary-Layer Interaction," *AIAA Journal*, Vol. 22, No. 9, 1984, pp. 1254–1262.

[66]Ardonceau, P., Lee, D. H., de Roquefort, T. L., and Goethals, R., "Turbulence Behavior in a Shock Wave/Boundary Layer Interaction," AGARD CP-271, Paper 8, Jan. 1980.

[67]Settles, T. J., Fitzpatrick, T. J., Bogdonoff, S. M., "Detailed Study of Attached and Separated Compression Corner Flowfield in High Number Supersonic Flow," *AIAA Journal*, Vol. 17, No. 6, 1979, pp. 579–585.

[68]Delery, J., "Experimental Investigation of Turbulence Properties in Transonic Shock/Boundary Layer Interaction," *AIAA Journal*, Vol. 21, No. 2, 1983, pp. 180–185.

[69]LeBalleur, J. C., and Blaise, D., "Methode Numerique d'Interaction Visqueux-Non Visqueux pour des Ecoulements Internes Decolles et L'Interaction Couche Limite-Onde de Chock," AGARD CP-401, *Transonic and Supersonic Phenomena in Turbomachines*, Paper 13, Sept. 1986.

[70]Bachalo, W. D., and Johnson, D. A., "An Investigation of Transonic Turbulent Boundary Layer Separation Generated on an Axisymmetric Flow Field," AIAA Paper No. 79-1479, Jul. 1979.

[71]Kussoy, M. I., Viegas, J. R., and Horstman, C. C., "Investigation of a Three-Dimensional Shock Wave Separated Turbulent Boundary Layer," *AIAA Journal*, Vol. 18, No. 12, 1980, pp. 1477–1484.

[72]Benay, R., Pot, T., and Delery, J., "Etudes Fundamentales sur les Interactions onde de Choc-Couche Limite dans un Canal Tridimensional," AGARD CP-280, Paper 1, Mar. 1986.

[73]Zukoski, E. E., "Turbulent Boundary-Layer Separation in Front of a Forward Facing Step," *AIAA Journal*, Vol. 5, No. 10, 1967, pp. 1746–1753, Oct. 1967.

[74]Chapman, D. R., Kuehn, D. M., and Larson, H. K., "Investigation of Separated Flows in Supersonic and Subsonic Streams with Emphasis on the Effect of Transition," NASA Report 1356, Mar. 1957.

[75]Roshko, A., and Thomke, G. J., "Flare-Induced Separation Lengths in Supersonic Turbulent Boundary Layers," AIAA Paper No. 75-6, Jan. 1975.

[76]Settles, G. S., Bogdonoff, S. M., and Vas, I. E., "Incipient Separation of a Supersonic Turbulent Boundary Layer at Moderate to High Reynolds Numbers," AIAA Paper No. 75-7, Jan. 1975.

[77]Rubesin, M., "Turbulence Modeling for Aerodynamic Flows," AIAA paper 89-0606, Jan. 1989.

[78]Coakley, T. J., "Impact of Turbulence Modeling on Numerical Accuracy and Efficiency of Compressible Flow Simulation," NASA TM-88333, Sept. 1986.

[79]Horstman, C. C., and Johnson, D. A., "Prediction of Transonic Separated Flows," *AIAA Journal*, Vol. 22, No. 7, 1984, pp. 1001–1002.

[80]Delery, J. M., "Experimental Investigation of Turbulence Properties in Transonic Shock Boundary Layer Interactions," *AIAA Journal*, Vol. 21, No. 2, 1983, pp. 180–185.

[81]Sajben, M., Morris, T. J., Bogar, T. J., and Kroutil, J. C., "Confined Normal Shock/Turbulent Boundary-Layer Interaction Followed by an Adverse Pressure Gradient," AIAA Paper No. 89-0354, Jan. 1989.

[82]Mateer, G. G., Brosh, A., and Viegas, J. R., "A Normal-Shock Wave Turbulent Boundary-Layer Interaction at Transonic Speeds," AIAA Paper 76-161, Jan. 1976.

[83]Muck, K. E., and Smits, A. J., "The Behavior of a Compressible Turbulent Boundary Layer Under Incipient Separation," *Proceedings of the 4th Turbulence Shear Flow Conf.*, Karsruhe, Germany, 1983.

[84]Hayakawa, K., Smits, A. J., and Bogdonoff, S. M., "Hot Wire Investigation of Unseparated Shock Wave/Turbulent Boundary Layer Interaction," AIAA Paper No. 82-0985, June 1982.

[85]Hayakawa, K., Muck, K. C., Smits, A. J., and Bogdonoff, S. M., "The Evolution of Turbulence in Shock-Wave/Boundary Layer Interactions," Presented at 8th Australian Fluid Mechanics Conf., Univ. of Newcastle, New South Wales, Nov. 28–Dec. 2, 1983.

[86]Hayakawa, K., Smits, A. J., and Bogdonoff, S. M., "Turbulence Measurements in a Compressible Reattaching Shear Layer," *AIAA Journal*, Vol. 22, No. 7, pp. 889–895.

[87]Debieve, J. F., Gouin, H., and Gaviglio, J., "Evolution of the Reynolds Stress Tensor in a Shock Wave-Turbulence Interaction," *Indian Journal of Technology*, Vol. 20, March 1982, pp. 90–97.

[88]Brown, J. D., Brown, J. L., Kussoy, M. I., Holt, M., and Horstman, C. C., "Two Component LDV Investigation of Three-Dimensional Shock/Turbulent Boundary-Layer Interactions," *AIAA Journal*, Vol. 26, No. 1, 1988, pp. 52–56.

[89]Settles, G. S., Fitzpatrick, T. J., and Bogdonoff, S. M., "Detailed Study of Attached and Separated Compression Corner Flow Fields in High Reynolds Number Supersonic Flow," *AIAA Journal*, Vol. 17, No. 6, 1979, pp. 579–585.

[90]Horstman, C. C., and Rose, W. C., "Hot Wire Anemometry in Transonic Flows," *AIAA Journal*, Vol. 15, No. 3, 1972, pp. 395–401.

[91]Dolling, D. S., and Murphy, M. T., "Unsteadiness of the Separation Shock Wave Structure in a Supersonic Compression Ramp Flowfield," *AIAA Journal*, Vol. 21, No. 12, 1983, pp. 1628–1634.

[92]Hayashi, M., Aso, S., and Tan, A., "Fluctuation of Heat Transfer in Shock Wave/Turbulent Boundary-Layer Interaction," *AIAA Journal*, Vol. 27, No. 4, 1989, pp. 399–404.

[93]Holden, M. S., "Shock Wave-Turbulent Boundary Layer Interaction in Hypersonic Flow," AIAA Paper No. 72-74, Jan. 1972.

[94]Hankey, W. L. Jr., and Holden, M. S., "Two-Dimensional Shock Wave Boundary Layer Interactions," (AGARDograph No. 203) Part II—"Experimental Studies in Shock Wave-Boundary Layer Interactions," by M. S. Holden, June 1975, pp. 41–101.

[95]Mikulla, V., and Horstman, C. C., "Turbulence Measurements in Hypersonic Shock-Wave Boundary-Layer Interaction Flows," *AIAA Journal*, Vol. 14, No. 5, 1976, pp. 568–575.