MICRO ADAPTIVE FLOW CONTROL

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Abstract

The Defense Advanced Research Projects Agency (DARPA) Micro Adaptive Flow Control program is described using a series of illustrative examples. Adaptive flow control is defined to be the active modification of aerodynamic flows using small time dependent actuators in judiciously chosen locations. A diverse set of large scale applications are described, in which micro adaptive flow control has a substantial overall system benefit.

1 Introduction

Micro adaptive flow control is defined to be the large-scale modification of flows having technological significance through selective interaction with flow instabilities. These instabilities have the ability to extract energy from the external flow. They can grow exponentially to large amplitudes, and can be triggered selectively by small-scale actuation.

A fraction of one percent of the overall flow is adequate for actuation if the location and frequency of actuation are selected appropriately. This approach enables low power, highly distributed, redundant actuation systems. The technology is inherently more practical than the previous generation of flow control systems which used steady blowing, where high velocity flows must be provided through complex and high-loss ducting. Amplitudes are typically an order of magnitude smaller than those used for steady actuation.

The DARPA program has the objective of transitioning flow control technology from the laboratory to full-scale applications. High payoff applications have been selected where active flow control can lead to large system performance benefits. The performance benefits are in turn supported by system studies, which have aided the selection of applications with the greatest potential benefits.

The elements of adaptive flow control are illustrated in Fig. 1. The flow control subsystem in its most general form is comprised of integrated actuators, sensors, and a feedback control system. The actuation inputs are typically time-dependent, so actuators can take the form of modulating flow valves and electrically driven devices. Sensor arrays and controllers are in their infancy, and the current suite of activities in the program is for the most part open loop. Design tools that take into account time dependent flows are also a key element in implementing active systems.

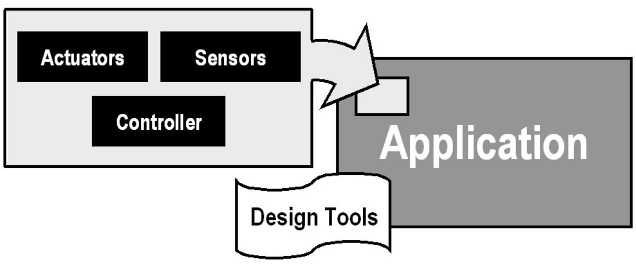


Fig. 1 Elements of micro adaptive flow control

The projects that comprise the DARPA Micro Adaptive Flow Control program are summarized in the following sections. The prime contractors are highlighted, but integrated teams of researchers and aerospace systems developers conduct the individual projects.

2 Aspirated Compressor; Massachusetts Institute of Technology

Boundary-layer aspiration is part of a systematic approach to improving the performance of engine compression systems. Aspiration works by removal of low-energy flow from the flow path. This low-energy flow would otherwise limit the diffusion (hence the work) of the compressor by causing flow separation from the surface of the blades. The approach depends crucially on the viability of a quasi-three dimensional design procedure that enables accurate prediction of the behavior of boundary layer flows in response to changes in blade shape, and also the effect of removal of the boundary layer fluid on the downstream behavior of the boundary layer. More aggressive blade designs have been enabled (Fig. 2), with larger turning angles without the consequences of flow separation.

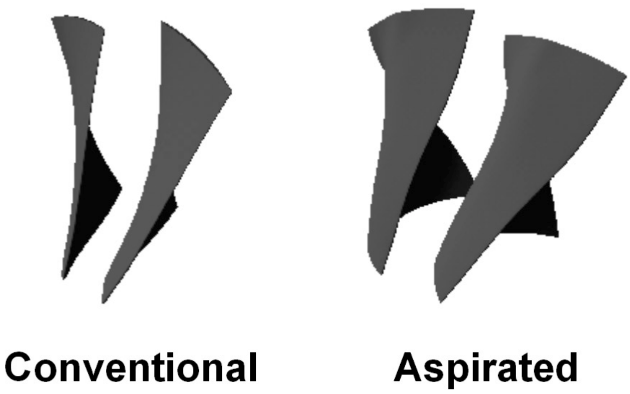


Fig. 2 Comparison of conventional and aspirated blades

Separation control is achieved by aspirating a portion of the boundary layer flow through appropriately placed slots on the blade surface. The inflow may be supersonic (Fig. 3), and the aspiration slot is located at a place along the blade suction surface where separation is incipient. In this context the flow control is actually a passive method that suppresses the growth of instabilities and energizes the boundary layer through mass removal.

The increased work enabled by aspiration can be realized only by taking advantage of the synergism between the blade shaping and the effects of boundary layer removal. Designs which are validated by three-dimensional analyses and experiments have shown that the work of a compressor stage can be doubled (at fixed blade speed and efficiency) by removing as little as one percent of the flow. A separate but related point is that removal of the high-entropy viscous flow can have a beneficial effect on compressor efficiency.

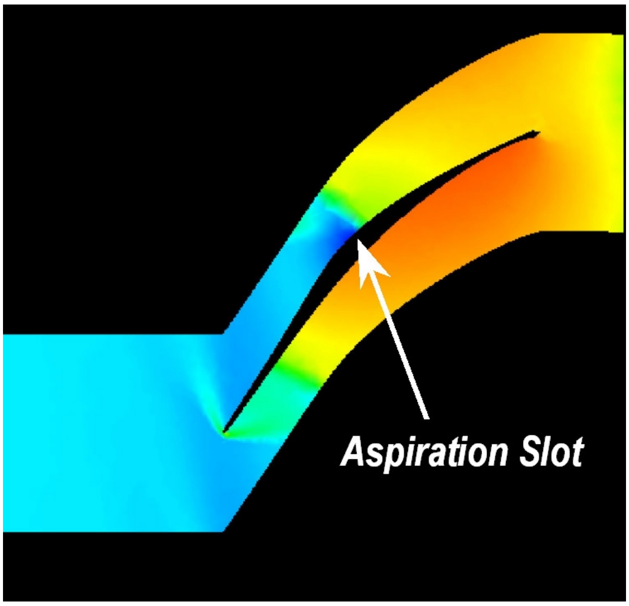


Fig. 3 Computed blade flowfield with aspiration

Two stages have been designed and experimental results have validated the performance of one of these stages. One stage produces a pressure ratio of 1.6 at a tip speed of 750 ft/sec, while the other gives a pressure ratio of 3.4 at a tip speed of 1,500 ft/sec. The first has been built and tested, the second has been designed, analyzed and is in construction.

The stages have a tip shroud (Fig. 4) that is composite wrapped to provide structural integrity for a stage with thin aspirated blades. The tip shroud also facilitates withdrawal of air through the blade tips and into a suction plenum. By approximately doubling the work capability of a stage, aspiration allows a reduction of stage count by about one half, with a concomitant reduction in weight of the compression system. Further, by combining the benefits of aspiration and counter-rotation of successive stages, even more dramatic reductions in stage count are possible.

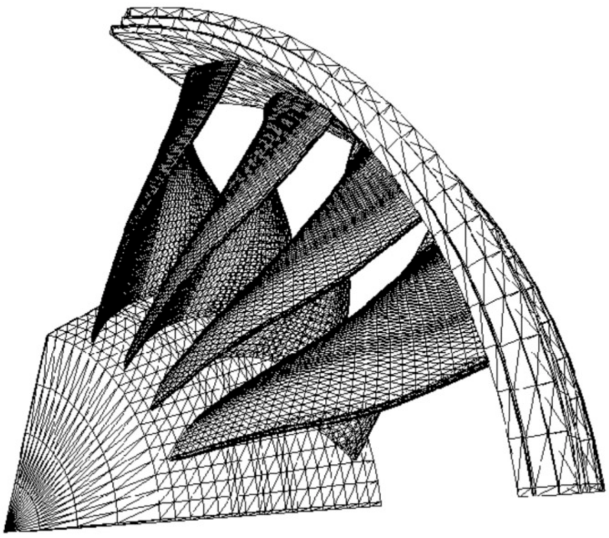


Fig. 4 Pressure ratio 3.5 shrouded aspirated stage

3 Active Core Exhaust (ACE) Control; Boeing

Active Core Exhaust (ACE) control is a fully integrated, non-intrusive pulsed injection system which excites large-scale instability of the core-exhaust plume resulting in enhanced mixing of the plume with surrounding ambient air. The plume instability is induced by injecting a small amount (1.5%) of high-pressure compressor bleed air perpendicular to the core exhaust flow at the core nozzle exit. In this application, natural instabilities of the flow are driven to saturation by selectively exciting these instabilities through injector ports at the nozzle exit. As seen in Fig. 5, the enhanced mixing results in a significant and rapid decrease of the exhaust plume temperature.

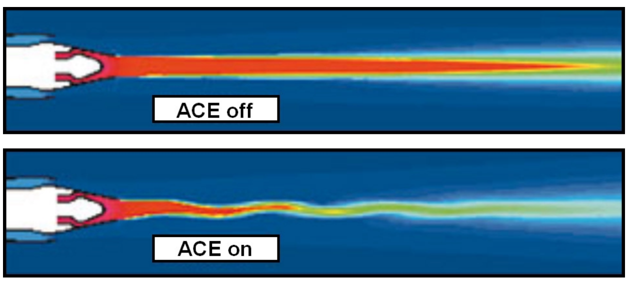


Fig. 5 ACE plume mixing enhancement

The reduced exhaust temperature permits removal of the Core Thrust Reverser (CTR) on the C-17 aircraft. Analysis and data indicate the CTR is not required to meet the C-17's backing requirements and is primarily used to meet human effectiveness temperature requirements for the load masters during engine-running ground operations.

Since ACE is an on-demand system, it can be used to reduce the thermal load on the trailing edge flaps during take-off and descent and then turned off during the cruise phase of the mission. Integration of the ACE control system will reduce the cost of each new C-17 aircraft built by approximately $1.2M and reduce the weight of every C-17 aircraft built or retrofitted by approximately 1200 lbs.

The engine bleed air is directed through a fluidic actuator, which provides periodic air injection to ports around the periphery of the nozzle exit (Fig. 6). The redesigned nozzle fully integrates all tubing and injector channels into the shell of the nozzle and the current engine housing. The redesign does not change the external profile of the nozzle. The compressor bleed air will be extracted from an existing port at the 17th stage and routed through a fluidic actuator, which diverts the bleed air to the appropriate injector. This actuator, which does not have any moving parts, is designed to provide a constant load to the compressor bleed port. A derivative benefit is obtained by allowing the integration of a plug within the nozzle.

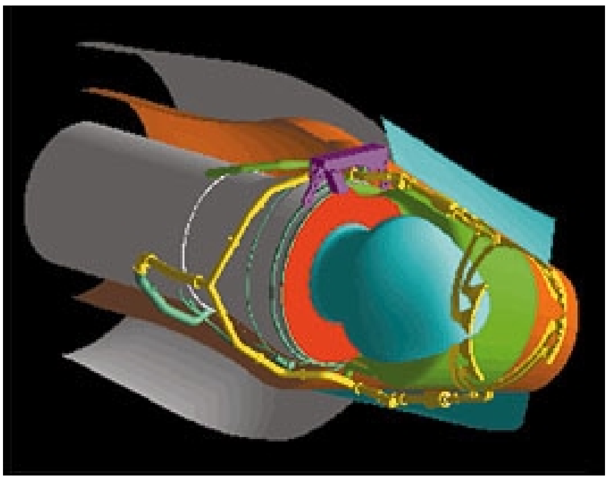


Fig. 6 C-17 nozzle with ACE system

4 Adaptive Virtual Aerosurface (AVIA); Georgia Institute of Technology

The Adaptive Virtual Aerosurface is based on the control the aerodynamic performance of a lifting surface through the modification of apparent shape by surface-mounted synthetic jet actuators. The distinctive feature of these actuators is that they do not require an external fluid source. Although there is no net mass injection into the overall system, there is momentum transfer into the external flow. The jets are able to change the effective camber of airfoils, control flow separation, and manipulate vortex flow phenomena both statically and dynamically.

Fig. 7 shows a schematic of a simple synthetic jet actuator. The jet is generated by an orifice and cavity backed by an actuated diaphragm. On the actuator downstroke, fluid is drawn into the cavity from all directions; on the upstroke fluid is directionally expelled from the cavity. The schlieren image of a two-dimensional synthetic air jet extends approximately 60 jet widths in the streamwise direction. The image clearly shows a vortex pair formed near the orifice and the outline of a turbulent jet farther downstream.

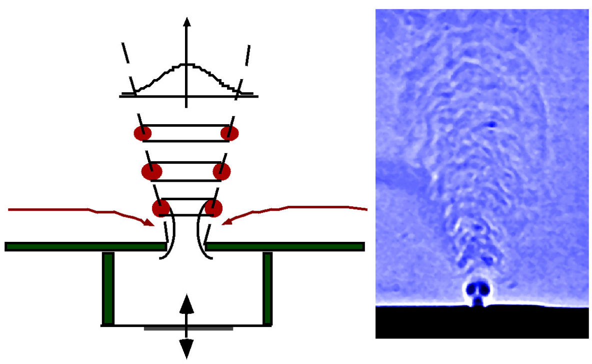


Fig. 7 Synthetic jet schematic and schlieren visualization

The synthetic jets have no net mass flux, and their interaction with an external flow results in formation of closed recirculation regions. In turn this produces an apparent modification of the surface shape and the surface pressure distribution. The accumulation of vorticity within these recirculating regions results in trapped or standing vortices (Fig. 8). These vortices displace the local streamlines of the cross flow thereby inducing the ‘apparent’ or ‘virtual’ change in the shape of the surface and of its global circulation. It is important to note that the operating frequency of the jet actuators is typically chosen to be at least an order of magnitude higher than the characteristic time scale of the flow. On the global time scale of the flow, the recirculating flow regions are virtually "steady" and the induced aerodynamic forces are unaffected by the operating frequency of the actuators.

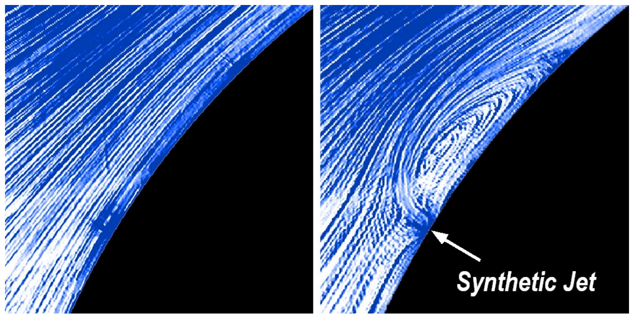


Fig. 8 Trapped vortex induced by synthetic jet

A distinction should be made here regarding different mechanisms for lift modification using these actuators. "W-type" actuation (developed by Wygnanski) is accomplished by coupling to an instability of the separating shear layer through pulsed blowing or zero-mass-flux actuation on the time scale of the flow about the airfoil. This approach relies explicitly on the receptivity of the separating shear layer to upstream control input and does not affect aerodynamic performance in the absence of flow separation.

In AVIA, "G-type" actuation (developed by Glezer) operates on a time scale that is at least an order of magnitude shorter than the characteristic time scale of the flow. It does not rely on coupling to a global instability, and can be effective (in principle) throughout the entire flight envelope.

Since the aerodynamic characteristics of an airfoil depend critically on the location of its front and rear stagnation points (and on its camber and thickness) synthetic jet actuators can alter these characteristics without the use of movable flaps. Jet arrays that create closed recirculating flow regions along the leading and trailing edges and along the upper and lower surfaces of an airfoil can displace front and rear stagnation points. The apparent thickness and camber of the airfoil are thus changed.

The objective of AVIA is the development and demonstration of aerodynamic flow control methodology based on synthetic jet actuators and. The potential aerodynamic benefits to be gained by active flow control were estimated (Fig. 9) by comparing idealized linear potential solutions with fully viscous Navier-Stokes calculations for a generic vehicle. The potential and viscous calculations provide upper and lower bounds, respectively, on the lift versus drag curve. It is projected that AVIA technology could provide up to 100% increase in turn rate along with a 75% reduction in flight control weight. A new generation of high power actuators will be developed that are suitable for high-speed flows. The effort will culminate in a planned demonstration on a semi-span model at high subsonic speeds.

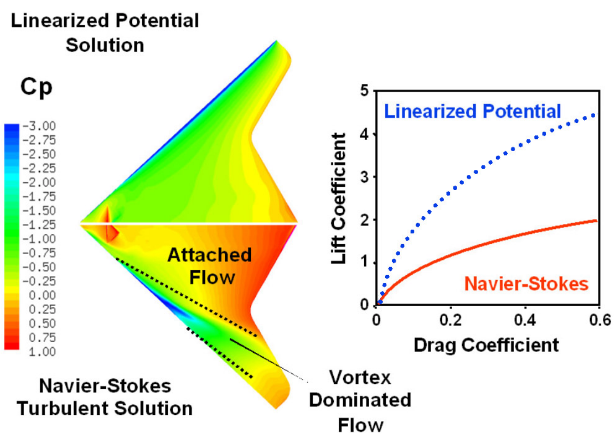


Fig. 9 Range of flows available with actuation

5 V-22 Download Alleviation; Boeing

When the V-22 is in hover (Fig. 10), the wing does not rotate with the engines. The wings are effectively at 90 degrees angle of attack relative to the propeller downwash. The downwash produces as much as 3000 pounds of download on the wings, compared to a payload of about 5000 lbs. for some V-22 missions. The download can be reduced by using "W-type" flow control to eliminate the flow separation on the trailing edge flap, thereby allowing the flap to be effective at larger. deflection angles. This net benefit is a payload increase of over 1000 lbs., a greater than 20% increase for some missions. Further range/payload benefits are accrued by using flow separation control for cruise drag reduction.

Fig. 10 V-22 Osprey in near-hover

Two types of actuators have been studied, a synthetic jet and a vibrating surface called a flipperon. As illustrated in Fig. 11, both are mounted toward the trailing edge of the V-22 wing. They control flow separation when the nacelles are essentially vertical (during hover). The synthetic jet actuation uses multilayer PVDF piezo-polymers in a recurve configuration to provide sufficient force and amplitude. The system advantage is that only electrical connections are required, avoiding the plumbing that limits performance. The synthetic jet is located at the leading edge of the trailing edge flap (called flapperon because they also provide the aileron roll function).

A seal covers the gap between the flap and wing, particularly when the flapperon is at its furthest downward deflection of 70 degrees during takeoff and hovering flight. The alternative actuation concept is to vibrate all or part of the seal, with a device called a flipperon. This is illustrated at the upper right in Fig. 11. The flipperon serves to eject fluid like the synthetic jet, but it also physically interacts with the local flow field.

One of the important advantages of the V-22 application is that the download flow velocity is on the order of Mach 0.1. This feature makes actuation much easier than for other higher speed applications, as the actuator velocity is typically on the order of the flow velocity. An important scaling parameter is the ratio of actuation flow momentum to the prime flow momentum. Measurements show that at amplitudes of order 1%, most of the benefits are achieve

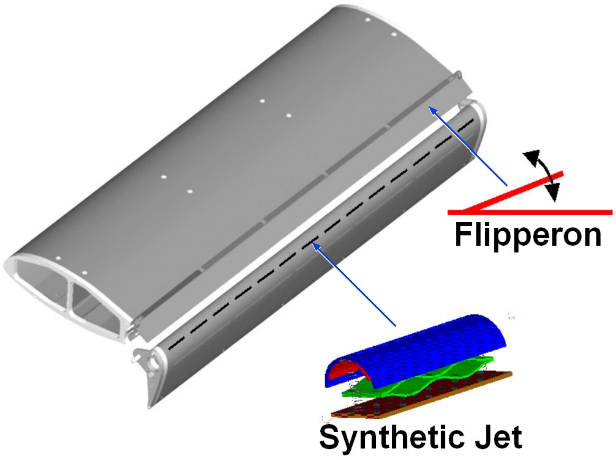


Fig. 11 Flipperon and synthetic jet actuator applied to V-22 flapperon

The quantification of the V-22 payload benefits comes from two-dimensional wind tunnel tests of a section of the wing, mounted almost perpendicular to the flow to simulate the hover flight condition (Fig. 12). The baseline drag is at a minimum when the flapperon angle is about 70 degrees. At higher angles, the flow separates and drag increases. With actuation, the flapperon angle can be increased to 90 degrees without separation. This results in a reduction in wing drag coefficient from 1.4 to 0.8, which corresponds to a payload increase of 1000 lbs. Subsequent testing with a flipperon actuator has also shown reduced drag.

The evaluation of system realizability measures the practicality of implementing AFC on the V-22 in the near future. Realizability includes such things as reliability, cost, weight, risk, technology status, and environmental compatibility. The actuation is assumed highly redundant as it is piezo driven with electronics co-located with the actuators to minimize high voltage wiring. The electronics are commanded from the redundant flight computer, and only programmed commanding will be necessary to cope with 3D effects. The actuation electronics will be energy recovery and storage type to minimize overall power and cooling requirements. One of the key realizability study results was that failure of actuation on an entire wing would not normally be catastrophic. This is because the flight control loops changing rotor pitch (thrust) are sufficiently rapid to cope with the upsetting rolling moment.

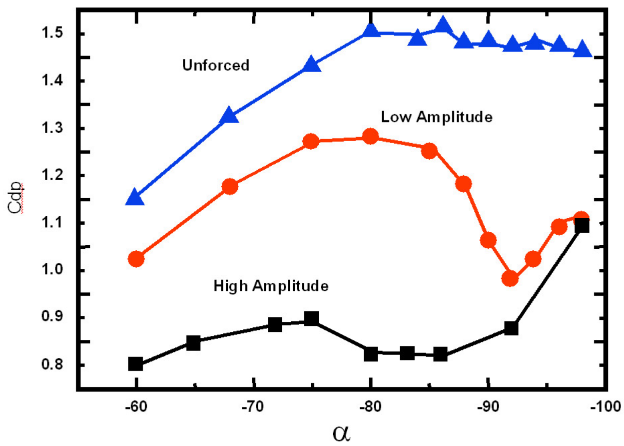


Fig. 12 Form drag reduction available at different levels of excitation

6 Retreating Blade Stall Control; United Technologies Research Center

Future high performance rotorcraft are likely to need significant improvements in attributes such as increased range for global self-deployability, increased speed and performance (maneuverability and agility) for fast agile missions, increased payload, reduced external noise emissions, and reduced cabin noise and vibration. Active flow control technologies can delay blade stall to minimize unsteady loads and vibration and increase rotor maximum load capability and aerodynamic efficiency in forward flight.

Retreating blade stall (RBS) establishes limits on rotor load and flight speed. The flow mechanism involved in blade stall is boundary layer separation near the leading edge of the rotor blade during rapid motion to high angle of attack. Some separation is usually present on the inboard portion of the retreating blade in forward flight, but the region of separation grows rapidly as load and speed increase. Fig. 13 is based on model rotor experimental data and shows the relatively small stall regions for a moderate load condition, and significantly larger stall regions at a high load condition that is very close to the stall boundary.

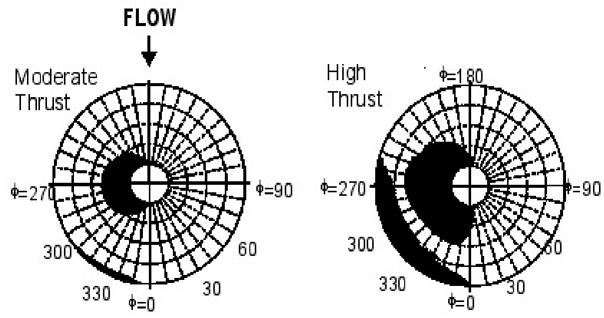


Fig. 13 Retreating blade stall penalty increasing with thrust

A practical way of avoiding or significantly delaying RBS has not yet been demonstrated under flight conditions. Separation control must be effective in an unsteady, compressible, high Reynolds number flow. The stall control approach that is applied on the retreating blade side must not create a significant drag penalty on the advancing side, where the relative external Mach numbers are 0.6 to 0.85. Further, the approach must be incorporated into a rotating blade. Mechanical high lift devices such as a leading edge slats have been investigated and found to be able to delay stall, but they have tended to either create excess drag if left in place on the advancing side, or to be impractical to deploy and retract each rotor revolution.

"W-type" actuation has been selected as a means to postpone stall for a rotorcraft blade. The primary separation control technique is a Directed Synthetic Jet (DSJ). A synthetic jet is formed with an exit neck optimized for separation control (Fig. 14). The curved neck allows low momentum fluid to be ingested during the suction phase of the DSJ and high momentum fluid to be ejected during the blowing phase. Both phases energize the boundary layer.

The impact of RBS control on conventional rotor lift boundaries is shown in Fig. 15 as rotor thrust versus rotor advance ratio for baseline and enhanced configurations. Enhanced rotor performance from a separation control system is shown for two moderate levels of improvement. The first increment represents a 5-degree increase in the stall angle of attack. The second increment adds an additional 10% increase in maximum lift coefficient. For maneuvers that occur at the minimum power airspeed of about 85 knots (an advance ratio of 0.20), the improvements increase blade load by 10%. Resultant increases in turn rate would improve the ability of an aircraft engaged in combat to turn on a target.

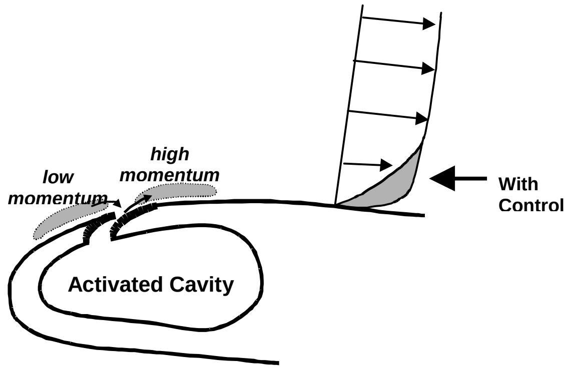


Fig. 14 Directed synthetic jet in leading edge of rotor

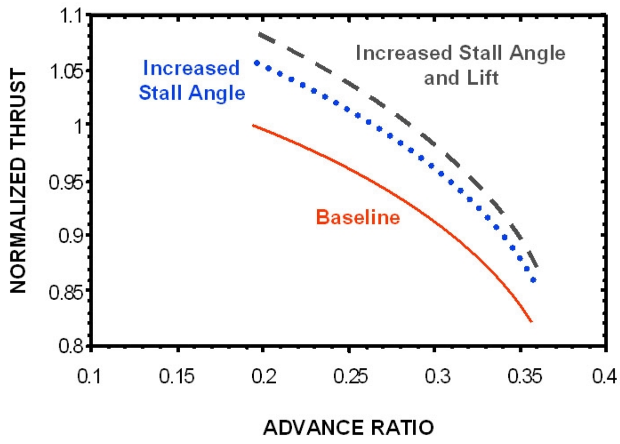


Fig. 15 Effect of improved stall limits on helicopter stall boundaries

7 Mesoflaps for Aeroelastic Transpiration; University of Illinois Urbana-Champaign

A concept termed Mesoflaps for Aeroelastic Transpiration (MAT) has the capability of controlling shock/boundary-layer interaction through passive cavity recirculation. The MAT system consists of a matrix of small flaps covering an enclosed cavity that are designed to undergo local aeroelastic deflection to achieve proper mass bleed or injection when subjected to shockwave loads. MAT leads to improved post-shock impingement boundary-layer characteristics (reduced boundary layer thickness and turbulence intensity) over the baseline no-bleed case. The interaction between shock waves and turbulent boundary layers is a primary determinant of the performance of high-speed aircraft. Control of the shock/boundary-layer interaction region has been suggested as a promising method to reduce the detrimental effects of strong shock waves, especially for supersonic engine inlets.

Fig. 16 shows a schematic of a supersonic mixed-compression inlet where boundary layer bleed is used to reduce separation at shock impingement locations. The bleed bands are used for both the internal oblique shocks as well as the terminating normal shock interactions with the engine cowl and centerbody boundary layers. Most engine inlets on military aircraft operating at speeds above Mach 2 employ active bleed control, which requires ducting of bleed flow to an external surface where it is discharged. The amount of bleed required increases significantly with Mach number and is on the order of 10-15% of the engine mass flow for Mach 3. Associated penalties including drag, weight, and cost of the overall vehicle are directly related to this bleed fraction

The goal of the MAT program is to efficiently prevent flow separation by shock boundary layer interaction with no net mass removal, thereby improving the resulting downstream boundary layer characteristics. The MAT concept consists of a matrix of mesoflaps covering an enclosed cavity as shown in Fig. 17. Each of the flaps is rigidly fixed over a small portion of its upstream end, but can aeroelastically deflect at its downstream end based on the pressure difference between the supersonic flow above and the subsonic cavity flow below. Under shock-free conditions the pressures above and below the flaps are nearly equal such that no transpiration is induced. Since the surface is nearly aerodynamically smooth for the case of no shock impingement, the roughening of the surface caused by conventional transpiration holes or slots is avoided.

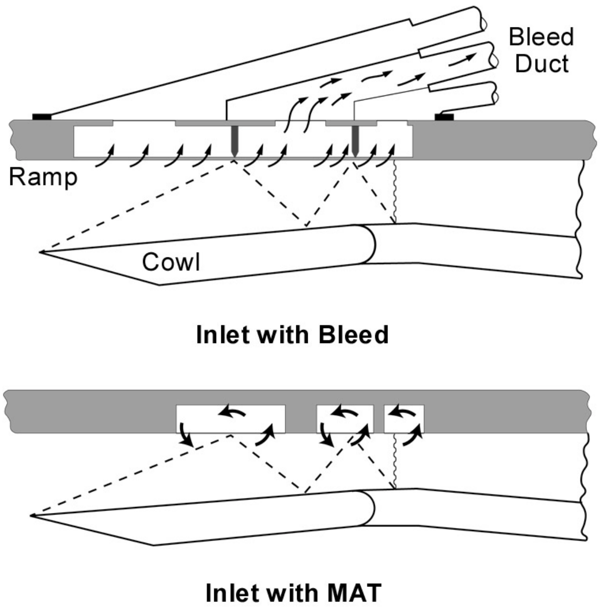


Fig. 16 Mixed compression inlet with bleed, and with MAT

Fig. 17 shows the MAT system as if an oblique shock were impinging on the mesoflaps between the second and third row, which induces a strong streamwise pressure variation. The nearly constant cavity pressure will lie roughly between the low pressure of the pre-shocked flow and the high pressure of the shocked flow. Consequently, the flaps upstream of the impingement location will deflect upwards allowing flow injection angled into the boundary layer and the flaps downstream will deflect downward to allow angled bleed from the boundary layer into the cavity.

The MAT concept retains the simplicity of a conventional passive-transpiration system (eliminating bleed airflow ducting and dumping), while also featuring three additional advantages based on the system's unique structural geometry. (1) The geometry allows improved bleed and injection aerodynamics (i.e., increased sonic mass coefficient) by employing angled transpiration. (2) It allows variable streamwise position and sweep angle of the shock while retaining angled transpiration. Therefore, the exact shock impingement locations are not required to be known a priori as a function of flight/inlet conditions. (3) The MAT system allows high aerodynamic efficiency in subsonic flow (i.e., skin friction consistent with that of a solid wall) since the system will simply revert to a nearly smooth flat plate when no shocks are present.

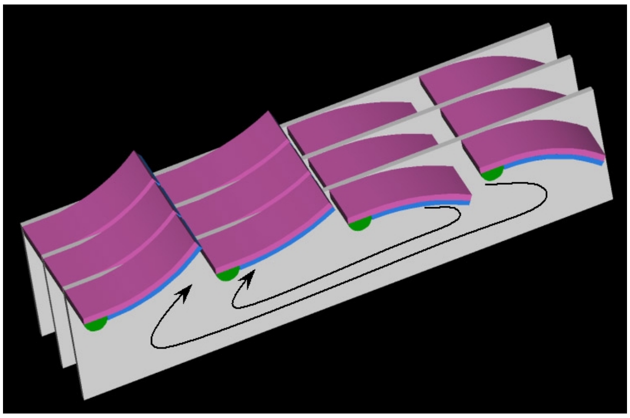
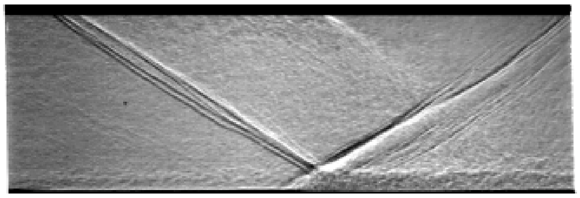


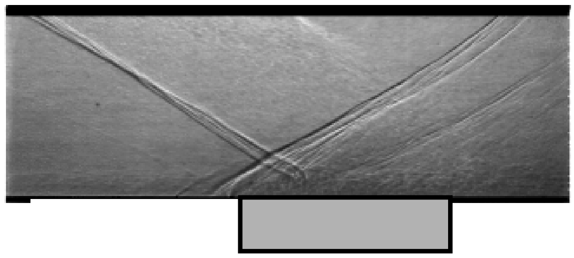
Fig. 17 MAT with aeroelastically deformed flaps

The typical features of a classic shock boundary layer interaction are identifiable in Fig. 18, which shows the flow over the solid wall at the location of the shock impingement. The incident shock impinges on the lower wall and reflects and coalesces close to the wall, with a shock front developed as a result of the rapid thickening of the boundary layer. A slip line, indicating the presence of a shear layer internal to the boundary layer, was also observed initiating at the intersection of the incident and reflected shocks.

A typical image with a MAT array is shown also in Figure 18. The leading oblique shock previously associated with the rapid thickening of the wall boundary layer in the solid wall case was located further upstream at a position over the first flap. This is clear evidence of the upward deflection of the upstream flaps, coupled with injection of bleed flow from this flap location and is typical of other work on passive control of shock/boundary-layer interactions. The slip line seen for the solid wall has effectively been suppressed.



Solid Wall



MAT Over Cavity

Fig. 18 Shock boundary layer interaction with solid wall and with MAT array

8 Conclusion

Micro adaptive flow control is just beginning to be applied to full-scale systems. System studies indicate that the benefits of this technology are substantial for a variety of applications. The DARPA Micro Adaptive Flow Control program will provide us with our first experiences with real systems, albeit open loop in nature.

Beyond these initial applications of open-loop systems that either suppress or saturate instabilities, we look toward true closed-loop flow control. It is in this next generation of adaptive flow control that we expect to see the true payoff of this promising technology.