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FLOW CONTROL RESEARCH AT NASA GLENN FOR GAS TURBINE ENGINES

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Summary

Flow control is used on aerodynamic surfaces to improve flow quality by reducing or eliminating undesirable flow features such as thick boundary layers and possible flow separation that lead to large energy losses resulting in negative impact on performance of the flow device. As the demand for more efficient gas turbine engines increases in the future, flow control technology in various components of the engine will be critical in enabling the future designs to meet the necessary performance requirements.

Technology advancement in gas turbine engines used for aerospace propulsion has been focused on achieving significant improvement in thrust-to-weight ratio as well as component and system efficiencies with the overall goals of reducing engine weight, fuel burn, emissions, and noise. In future, to meet ever-increasing stringent noise, emission, and fuel burn reduction goals, different airframe and engine architectures compared to the conventional tube and wing configurations need to be considered. One such advanced configuration is the hybrid wing-body (or blended wing body) airframe with embedded distributed propulsion system. In addition to the benefits of large bypass ratio of distributed propulsion and additional benefit of higher propulsion efficiency due to boundary layer ingestion, a number of technical challenges such as the effect of flow field distortion due to the embedded nature of the engine inlets and boundary layer ingestion on engine performance and strategies to mitigate the detrimental impact on performance and life need to be addressed as part of the technology advancement. In addition, to realize higher thermal efficiency, high pressure core engine technology and the complexities of technical challenges such as small blade heights and tight clearance requirements in the aft stages of compressors as well as NOx challenge due to high temperatures in combustion chambers are some of the advanced technology candidates considered for active research.

At a component level, these goals translate to aggressive designs of all the engine components well beyond the state of the art. Compressors and turbines would need highly loaded turbomachinery blades resulting in dramatic increase of work absorption/output of compressor/turbine without unduly increasing rotational speed

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and maintaining high efficiencies as well as adequate stable operability margins. Inlets and nozzles should be able to diffuse and expand the flow in much smaller regions maintaining minimum total pressure losses and satisfying operability, cost, weight, signature, life, and acoustic requirements at the same time. Combustor designs need to deliver targeted emission reductions through efficient combustion at lower peak temperatures in order to eliminate or significantly reduce NOx, CO2, and unburnt hydrocarbons. All these advancements invariably need some type of flow control technology in each of the components mentioned above. Objective of this paper is to present an overview of the flow control research currently underway at NASA Glenn Research Center. Some representative results of application of the flow control technology for various engine components such as compressors, combustors, turbines, inlets, and nozzles are presented in the following paragraphs.

Compressors:

Thermal efficiency of a gas turbine engine increases as the overall pressure ratio increases. To achieve very high pressure ratios, compressor design becomes very challenging for the rear stages of the high pressure compressor due to extremely short blade heights and resultant tight blade clearances that need to be maintained for reasonable efficiency. Next generation of engines will have compressors with highly complicated three-dimensional turbomachinery blades with flow passages where viscous flow losses will present a significant design challenge to achieve high efficiency while maintaining a healthy stability (stall) margin for realistic blade tip clearances. Flow control will be one of the critical technologies to overcome the technical challenges mentioned above. Following is a brief description of a few flow control research efforts in the compressor area carried out over the past several years at NASA Glenn.

Flow control using impulsive injection from the suction surface of a stator vane has been applied in a low speed axial compressor to reduce losses due to separation. Impulsive injection has been shown to reduce the required amount of injection air in comparison to steady injection. In addition, impulsive injection has been shown to significantly reduce separation relative to steady injection. Variations in injected mass, frequency, and duty cycle have been explored (Ref. 1).

Flow aspiration is applied for an axial compressor stage where low-energy viscous flow is aspirated from diffusion-limiting locations on the blades and flowpath surfaces of the stage enabling a very high pressure ratio in a single stage. The aspirated fan stage is designed to achieve a pressure ratio of 3.4:1 at 1500 feet/sec which is well beyond the current state of design capability (Ref. 2).

The operational envelope of gas turbine engines is constrained by the stability limit of the compression system. The dangers of exceeding this limit are severe, with the potential for engine failure and loss of the aircraft. To avoid such failures, compressor designers provide an adequate stability (stall) margin in the compressor design to account for inlet distortions, degradation due to wear, throttle transients, and other factors. State-of-the-art approaches to increasing stability tend to decrease the efficiency of the compressor. The focus of research at NASA GRC is to increase the stall margin of compressors without decreasing their efficiency by using

flow control techniques that rely on the inherent energy rise that is imparted by a compressor rotor. The higher energy fluid from the aft of the rotor is recirculated back to the front of the rotor, thus energizing the low-momentum flow in the rotor casing end wall region that typically sets the stability limits of the compressor (Ref. 3).

Combustor:

NOx (Oxides of Nitrogen) emission in a combustor directly depends on the bulk temperature of the gases and for that reason lean combustion is shown to be advantageous for reducing NOx emissions as well as enhancing turbine temperature distribution and efficiency. However, lean combustion is also more prone to thermo-acoustic instabilities. These instabilities are typically the result of the coupling of the fluctuating heat release (of the combustion process) with the lightly damped acoustics of the combustion chamber. Fuel flow management is currently the only means used to control aircraft engine combustor operation (Ref. 4). The extremely hostile environment in combustors preclude using variable geometry features to affect the airflows. Fuel flow control is driven by externally located valves directed by the FADEC (Full Authority Digital Engine Control) that apportions the fuel flows to multiple groups of fuel injector nozzles.

Fuel staging is the technique of apportioning fuel through fuel injector nozzles of staggered sizes to allow the combustors to operate stably through the large range of fuel flows from engine idle to full-powered takeoff conditions. A typical arrangement has a small pilot nozzle nested inside a larger main nozzle concentrically. At engine idle, only the small pilot is used. As the engine power demand increases, fuel flow in the main nozzle is turned on and increased.

In order to focus control development toward realistic combustor instabilities, a combustor rig that replicates an aero engine combustor instability has been designed and fabricated (Ref. 6). The combustor rig successfully replicates the observed real-world engine instability and operates at engine pressure and temperature conditions. This is a single-nozzle combustor rig, which has many of the complexities of the actual engine combustor.

Turbines:

Flow Separation in the Low Pressure Turbine (LPT) is a major barrier that limits further improvements of aerodynamic designs of the turbine airfoils. The separation is responsible for performance degradation and prevents design of highly loaded airfoils .The separation can be delayed, reduced, or eliminated completely by using flow control techniques. Successful flow control technology will enable breakthrough improvements in gas turbine performance and design.

Active flow control of boundary-layer separation using glow-discharge plasma actuators is studied experimentally (Ref. 7) in a closed circuit wind tunnel.

Inlets:

One of the efforts in the inlet area is the supersonic diffuser research with an objective to develop a mechanically simple, non-bleed quiet supersonic jet external compression inlet to operate up to a Mach number of 2.0 and to support long range aircraft technologies in propulsion airframe integration. In supersonic inlets, air flow decelerates and gets compressed through a series of oblique shocks and these shocks interact with the boundary layer and thickens it resulting in potentially separated flow leading to large energy losses and decreased system performance. Traditionally, shock boundary layer interaction is controlled by bleed which removes low-momentum flow from the boundary layer using suction through porous surface on the inlet wall. However, bleed systems decrease mass flow through the engine and introduce additional drag in addition to being heavy and complex leading to engine performance degradation . In an effort to minimize or eliminate bleed, vortex generators are considered as possible replacement to bleed systems. The approach used is to develop a capability optimal control of shock wave/turbulent boundary layer interactions using arrays of micro-ramp actuators on the diffuser wall. Numerical Simulation and experiment are used to establish the ability of micro-array flow control to manage shock wave turbulent boundary layer interactions, determine the design and robustness characteristics of micro-ramp array flow control, and to evaluate the effectiveness of micro-ramp array flow control relative to conventional inlet boundary layer bleed (Ref. 8).

Engine Noise Reduction:

Engine noise reduction efforts at GRC using flow control have been focused primarily on fan and jet noise. For fan noise, trailing edge blowing experiments and analyses have been performed over the past few years to reduce rotor/stator interaction noise sources. Tests were first performed using a low-speed fan to verify that both tone and broadband noise reduction can be achieved (Ref. 9). The basic idea is to fill the fan wakes by injecting air at the trailing edge of the fan. CFD was used to guide the design of the flow passages through the hub and fan blades. Tests have been performed in the NASA Glenn 9x15 wind tunnel for a higher speed fan that is representative of a higher bypass ratio engine application. Both aerodynamic performance and acoustic data have been obtained.

For jet noise, there has been some work looking at exciting the jet to promote mixing by circumferential flow injection. Recently, there has been an effort to use microjets at the nozzle exit to enhance mixing characteristics of the jet flow field resulting in noise reduction (Ref. 10). Microjets have been shown to produce large effects on supersonic primary jets apparently due to a weakening of shock structure resulting in an attenuation of shock-associated noise. The experiment was conducted in an open jet facility at NASA Glenn Research Center. Three convergent-divergent nozzles and a convergent nozzle were used, all having the same exit diameter of 37.6 mm. The four nozzles were designated as M10, M15, M18, and M22 to denote the design Mach numbers as 1.0, 1.5, 1.8, and 2.2 respectively. The microjets were injected via a ring manifold mounted on the primary nozzle .There were six threaded outlets on the manifold to fit six tiny cylindrical injectors around the circumference to inject air flow perpendicular to the primary jet axis.

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