MECHANISM STUDY ON THE IMPACT OF PULSED JET AT THE BLADE TIP LEADING EDGE ON THE FLOW FIELD IN COMPRESSOR

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Abstract

A pulsed jet hole is introduced at leading edge on the casing wall of a simplified model of compressor rotor cascade and LES is conducted. The mechanism of periodic flow control is found to resonate with the tip gap vortex, leading to more organized vortices that wrap around the primary leakage vortex with smaller pitch and radius, thereby reducing the affected area and delaying vortex breakdown.

Keywords: compressor; unsteady flow control; tip leakage vortex; large eddy simulation.

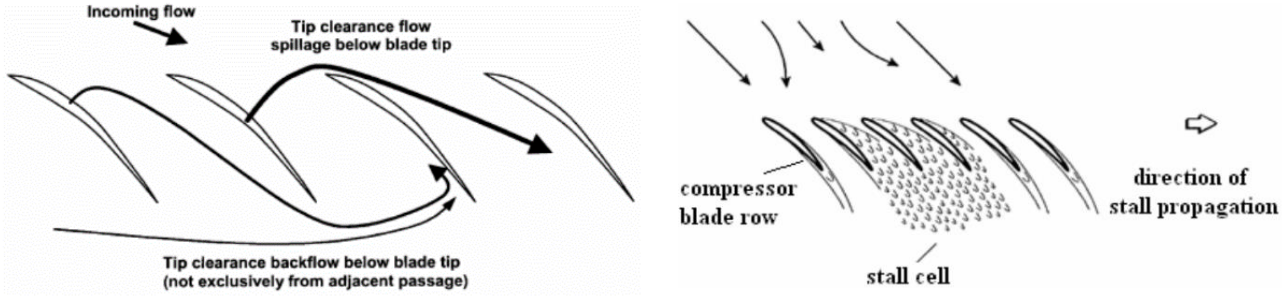
1. Introduction

The compression capability (pressure ratio) and efficiency of the compressor/fan in aviation propulsion systems determine the overall performance level, serving as crucial indicators of fluid machinery performance. Meanwhile, stability margin fundamentally dictates the practicality and safety of the engine [1]. Currently, advanced military turbofan engines achieve pressure ratios of around 28, while civilian high-bypass turbofan engines reach pressure ratios as high as 40. The next generation of high-bypass turbofan engines is estimated to reach pressure ratios of around 50, indicating an increased demand for stability margin with enhanced compression capability.

Flutter and stall are two unstable conditions of the engine. During stall, stall vortices propagate around the circumferential channels of different blades of the compressor at speeds lower than the rotation speed, resulting in a sudden deterioration of compressor performance and even the risk of engine fire. Moreover, compressor blades are prone to fatigue under such high-frequency alternating aerodynamic loads. When the frequency of the stall vortex matches the natural frequency of the blades, significant vibrations may occur, leading to blade damage. If stall intensifies, engine flutter may occur, representing a highly unstable state where fluid in the engine exhibits reciprocating motion along the axial direction. This can cause unstable operation of the gas turbine and even accidents. Past compressor flutter incidents have caused significant losses in the aviation industry. For example, the crash of American Airlines Flight 383 in 1965 resulted in 62 deaths and 11 injuries. Investigations revealed that compressor flutter may have contributed to the fatigue fracture of the turbine fan disk. Additionally, the exacerbation of flutter in American Airlines Flight 191 in 1979 led to structural stress in the engine, resulting in the engine separation, 271 fatalities, and 2 injuries. In practical design, the stability margin of compressors can reach 20% to 30%.

However, there exists an inherent contradiction among high compression capability, high efficiency, and high stability margin. Increasing the compression capability of the compressor requires either increasing the rotor blade rim speed or increasing the flow turning angle. This inevitably leads to the strengthening of vortices in the compressor rotor passage, represented by the tip leakage vortex and the backflow separation vortex, resulting in increased intensity, enlarged scale, and enhanced unsteadiness of the vortices, as well as increased flow field losses and decreased stability margin of the compressor. Especially in the region near the rotor blade tip where both the tip leakage vortex and the backflow separation vortex coexist, research has shown that the stall of the compressor flow field caused by the tip leakage vortex and the backflow separation vortex is the fundamental reason for compressor instability. The fragmentation of the tip leakage vortex is the primary cause of flow blockage.

It is generally believed that there are two pathways leading to compressor stall: Spike Initiated Stall and Modal Wave Stall (as shown in Figure 1). Spike Initiated Stall is typically caused by the instability of blade tip leakage vortices, with axial backflow appearing at the blade leading edge cross-section (usually at the tip) and axial backflow at the trailing edge cross-section being the two main conditions for Spike Initiated Stall [2]. Modal Wave Stall is usually caused by channel blockage, and the stall cells propagate periodically along the circumferential direction throughout the entire blade row [3]. Therefore, if the flow control method is possible to bring the trajectory of blade tip leakage vortices closer to the blade back and alleviate channel blockage, it can increase the stability margin of the compressor.



Spike Initiated Stall Modal Wave Stall

Figure 1 - Two pathways leading to compressor stall

According to previous research [4], employing periodic excitation with a frequency approximately matching the dominant frequency of the controlled vortex can achieve highly efficient unsteady flow control. However, in the compressor blade tip region under near-stall conditions, both blade tip leakage vortices and blade backflow separation vortices exhibit strong non-periodicity and mutual coupling. The quasi-periodic characteristics in this region differ from the flow field with a single separation vortex structure. Therefore, this paper simplifies the blade tip flow field of the compressor twisted blade into a geometric model of a planar cascade with clearances. The Large Eddy Simulation (LES) method is employed to conduct numerical research on unsteady flow control at the blade tip leading edge. The study aims to explore the unsteady excitation control laws and control mechanisms that can effectively act on the typical quasi-periodic vortex structures at the compressor blade tip, thereby achieving high efficiency and stall margin enhancement.

In the study conducted by Deng Xiangyang [5] on the isolated rotor planar cascade model of a compressor, non-uniform excitation slots (synthetic jets) were strategically placed on the top casing of the model, fixed relative to the blades as depicted in Figure 2.

The research investigated the mechanism of unsteady flow control on the rotor blade tip flow field through URANS simulations. However, the evaluation of flow control effectiveness in this study primarily focused on the relative total pressure loss coefficient and static pressure rise coefficient. Although these parameters correspond to the efficiency and pressure ratio of the compressor rotor flow passage, respectively, there was insufficient attention given to crucial indicators determining the stall margin of the compressor rotor, such as the trajectory of the leakage vortex, the scale and influence range of the leakage vortex and separation vortex, and the location of vortex breakdown. Additionally, the study employed excessive excitation intensity (excitation amplitude of 1.4% of the mainstream flow rate). The control effect resulting from the flow control method used in the study can be regarded as primarily originating from the steady flow control effect brought about by injecting momentum into the time-averaged flow field. The high-efficiency unsteady flow control effect resulting from the resonance between non-uniform external excitation and specific frequency quasi-periodic vortex structures in the flow field was overshadowed, as shown in Figure 3. The author of this paper believes that this may also be a possible reason for the minor impact of changing the excitation frequency on the flow field structure observed in the study.

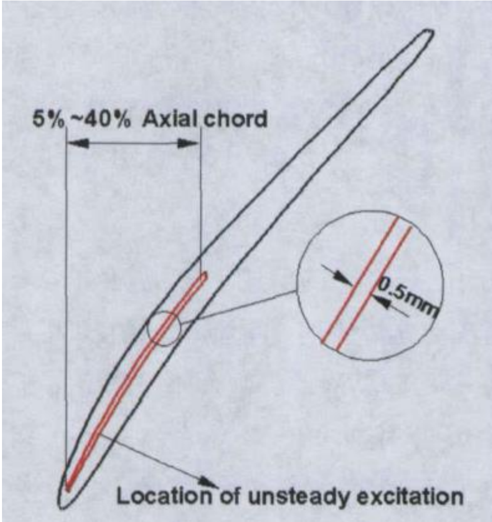


Figure 2 - The position of the non-uniform excitation slots on the top casing of the isolated rotor planar cascade model of the compressor

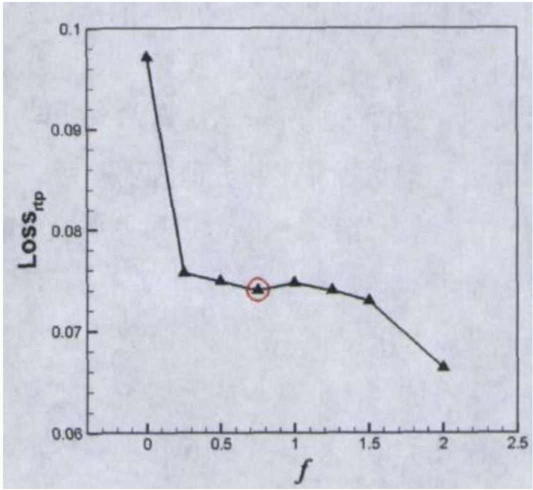


Figure 3 - The effect of unsteady excitation on the relative total pressure loss coefficient of the isolated rotor in the compressor

2. Numerical Computational Methods and Physical Models

Since the 1970s, computational fluid dynamics (CFD) methods have become important tools for exploring turbulent structures. According to the treatment of the turbulence terms in the Navier-Stokes equations, commonly used unsteady turbulent numerical simulation methods include Reynolds-Averaged Navier-Stokes Simulation (RANS) , Large Eddy Simulation (LES), and Direct Numerical Simulation (DNS). The accuracy of these numerical simulation methods increases in the order of RANS, LES, and DNS, but so does the computational cost. Moreover, higher-accuracy numerical simulation methods often result in larger data volumes, making it more challenging to extract regular conclusions from massive data. Studies have shown that LES can produce unsteady simulation results similar to DNS within acceptable computational power and time, and it is increasingly used to study the details of unsteady flow. If the subgrid scale model used is appropriate, LES can accurately simulate transient flow fields, allowing LES to capture the generation, development, and interaction of vortex structures larger than the inertial subrange more precisely and accurately compared to RANS simulations, while saving computational resources compared to DNS simulations. For these reasons, this study ultimately employs Large Eddy Simulation (LES) for the investigation.

Previous studies have been conducted on the two-dimensional blade cascade [6,7] and three-dimensional single-channel planar cascade [8] for a typical compressor blade profile. On the basis of the three-dimensional single-channel planar cascade model in the previous studies above, this study introduces a 0.5mm tip clearance layer between the blade tip and the upper wall of the planar cascade. The main geometric parameters of the airfoil and the incoming flow conditions are summarized in Table 1. The upper wall, initially subjected to periodic boundary conditions, is changed to a sliding solid wall. Its movement velocity in the y-direction matches the component of the incoming flow velocity in the y-direction. The left and right walls of the channel maintain periodic boundary conditions. To reduce computational complexity, the simulation considers only a partial blade height, with the blade height set at 8mm (16 times the tip clearance height). The lower wall is set as a symmetry plane. The entire grid is generated using Ansys ICEM CFD, employing structured meshing with a grid quantity of 5.58 million. The mesh is shown in the Figure 4.

Table 1 - The main parameters of the airfoil used in the numerical simulation study of the planar cascade

	Parameters
Chord length	60mm
Pitch	45mm
Geometric inlet angle	46°
Geometric outlet angle	-10°
Inlet angle of attack	9°
Inflow Mach number	0.1
Inlet Reynolds number	1.81×10^5

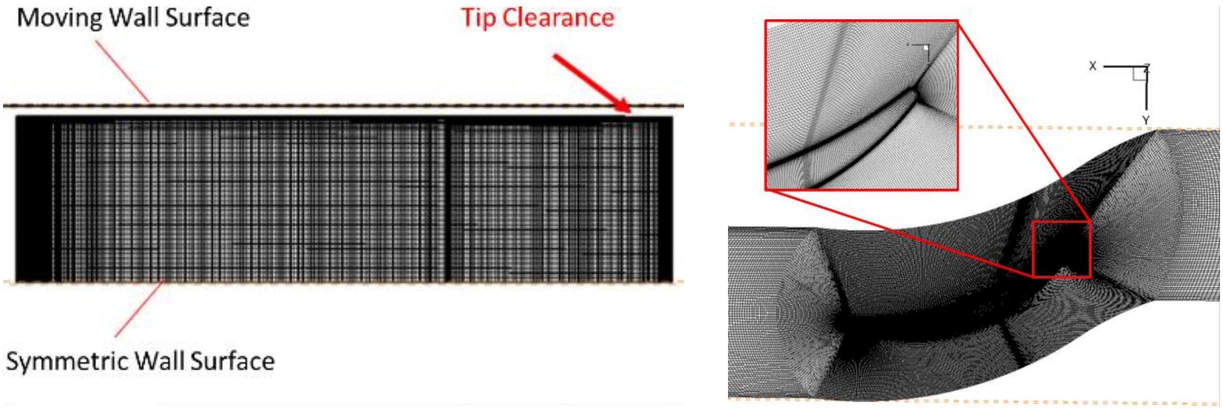


Figure 4 - Mesh of the Compressor Planar Cascade Model with Blade Tip Clearance

Utilizing Ansys Fluent's Meshing module, a square pulsating jet hole with a side length of 2mm was created on the casing wall near the leading edge and backside of the blade in the aforementioned planar cascade model with clearances. Additionally, the jet direction was specified to be at angle α and angle β with respect to the vertical and horizontal planes, as depicted in Figure 5. In the axial direction (X-axis, positive from leading edge to trailing edge), tangential direction (Y-axis, positive from blade tip to blade back), and spanwise direction (Z-axis, positive from root to tip), the jet direction forms an angle α=55° with the X-axis in the X-Y plane (aligned with the inflow direction) and an angle β=30° with the Y-axis in the Y-Z plane. The time-averaged jet flow rate is 1% of the time-averaged mainstream flow rate (equivalent to approximately 0.16% of the mainstream flow rate at a full blade height of 50mm).

The simulation computations were conducted using the commercial CFD software Ansys Fluent. The unsteady simulation was carried out using the Large Eddy Simulation (LES) method based on the results of the steady-state calculation. The time step for the unsteady calculation was set to 1×10^{-5}s. The total number of time steps for each case was set to be over 2×10^4 steps.

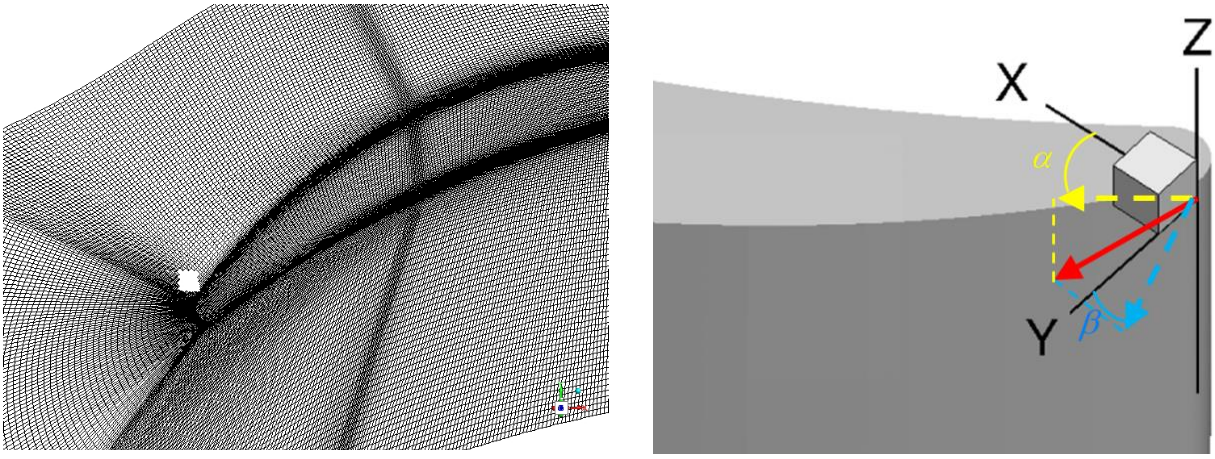


Figure 5 - Position, Shape, and Angle of the Casing Wall Pulsed Jet Hole

3. Numerical Study on the Influence of Different Jet Frequencies on Flow Field Structure

The excitation frequency is the most crucial factor affecting the effectiveness of unsteady flow control. Therefore, this paper primarily compares the influence of different jet frequencies on typical flow field structures. According to previous research [9], for this blade profile and under the given inlet and outlet conditions, the dominant frequency of blade backflow separation vortices is 480Hz. The parameter F^{+} is defined as the ratio of the excitation frequency of the pulsed jet to this frequency.

Considering that entropy values can intuitively reflect flow losses, the time-averaged entropy distribution at the section located at 50% axial chord length downstream of the trailing edge of the planar cascade, as shown in Figure 6, is compared for different excitation frequencies of the blade tip jet flow field in Figure 7.

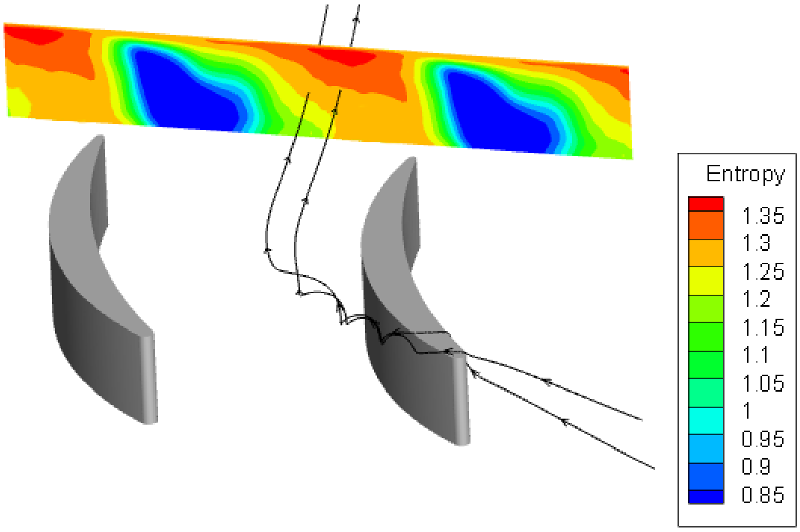


Figure 6 - Schematic Diagram of the Section Location Downstream of the Trailing Edge

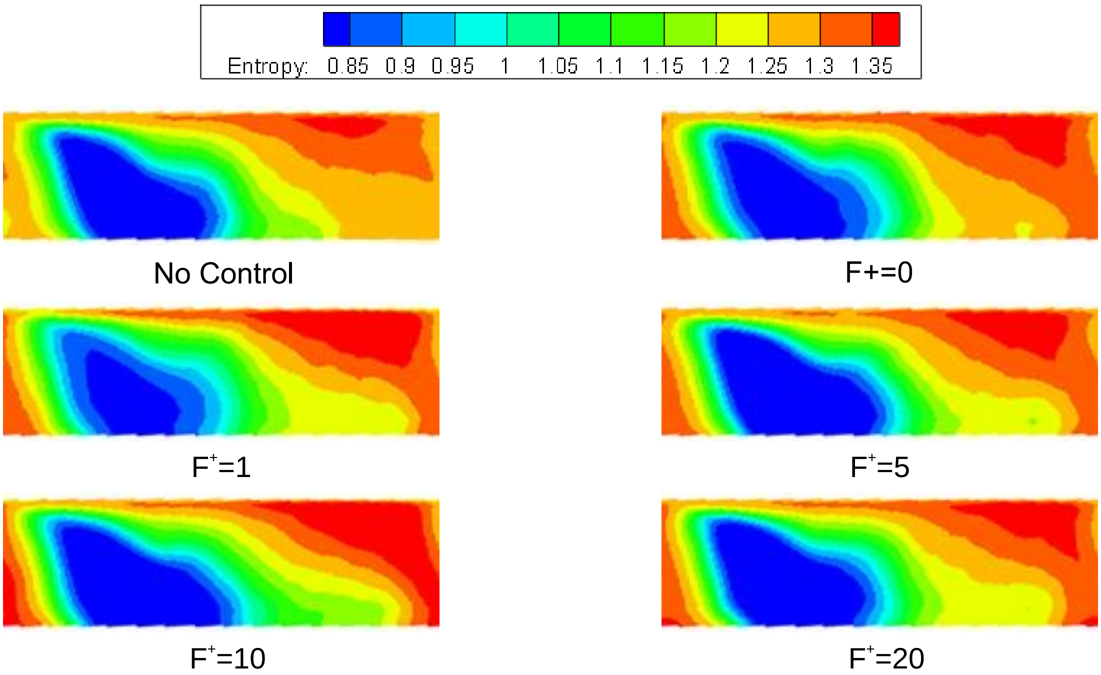


Figure 7 - Comparison of Entropy Distribution at 4 Downstream Trailing Edge Section Under Different Jet Frequencies

From the comparison in Figure 7, it can be observed that under various jet frequencies, the difference in entropy distribution on the downstream section of the trailing edge is not significant. This contrasts with the pulsating jet control targeting the blade backflow separation vortex, where the unsteady excitation effect amplifies as the separation vortex develops. Eventually, this manifests as a reduction in losses at the downstream trailing edge section, corresponding to an increase in efficiency of the compressor rotor or an enhancement in the static pressure recovery coefficient of the stator.

However, it is important to note that the minimal difference in flow impact on the downstream trailing edge section among different excitation frequencies does not imply a consistent effect on the entire flow field. Sections at 40% axial chord length from the leading edge were taken, as shown in Figure 8, and a comparison of the time-averaged entropy distribution at this section under different excitation frequencies is presented in Figure 9.

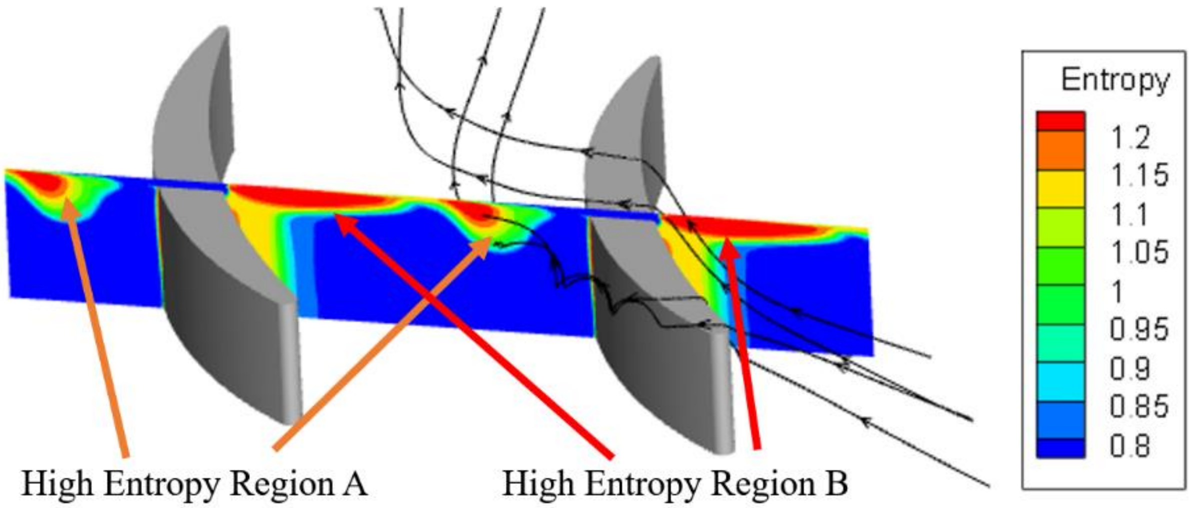


Figure 8 - Schematic Diagram of the Section Location at 40% Axial Chord Length

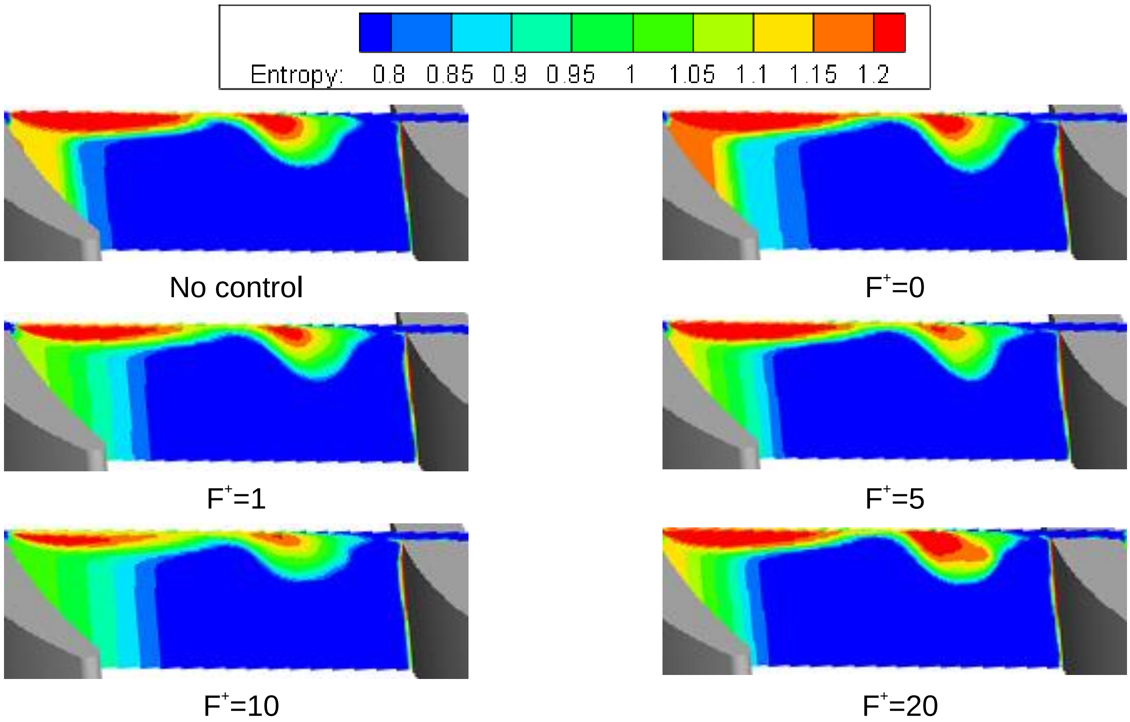


Figure 9 - Comparison of Entropy Distribution at 40% Axial Chord Length Section Under Different Jet Frequencies

From Figure 9, it can be observed that under excitation with F^+=10, the area of high entropy in this section is minimized. Taking Entropy value 1.2 as the boundary of the high entropy region, statistical analysis reveals a 52.29% reduction in the area of the high entropy region relative to the uncontrolled state and a 29.75% reduction relative to the F^+ = 1 state.

The transient Q criterion iso-surface near the blade tip clearance is shown in Figure 10 (all Q criterion iso-surfaces in the following are for Q=2×10^8, with pressure values relative to the inlet total pressure). The iso-surface plots of the transient flow field Q criterion from the leading edge to 50% (up) and from 50% to 90% (down) axial chord length on the backside of the blade at different excitation frequencies are shown in Figure 11.

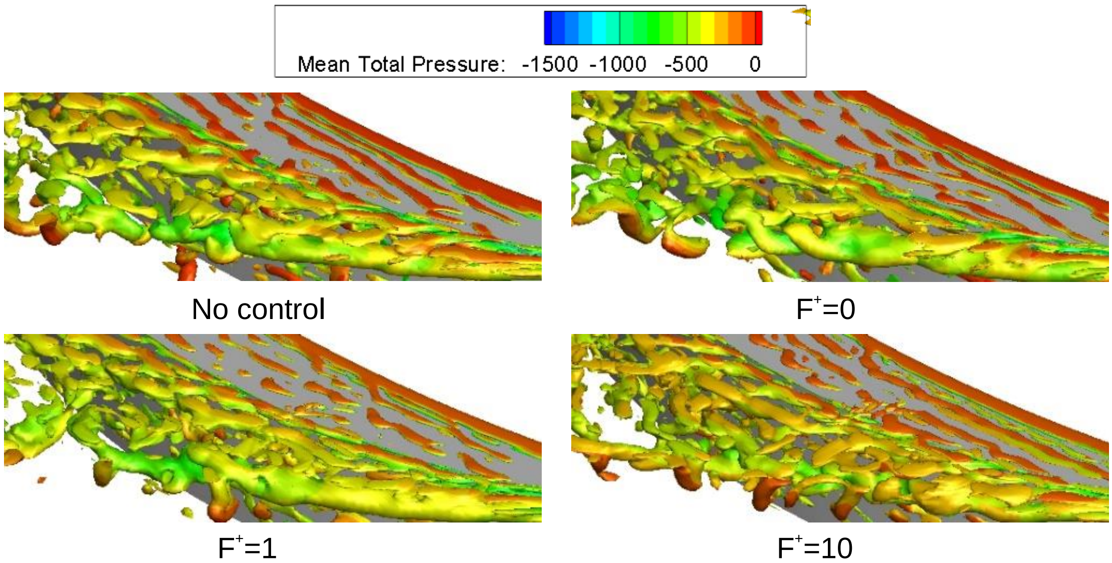
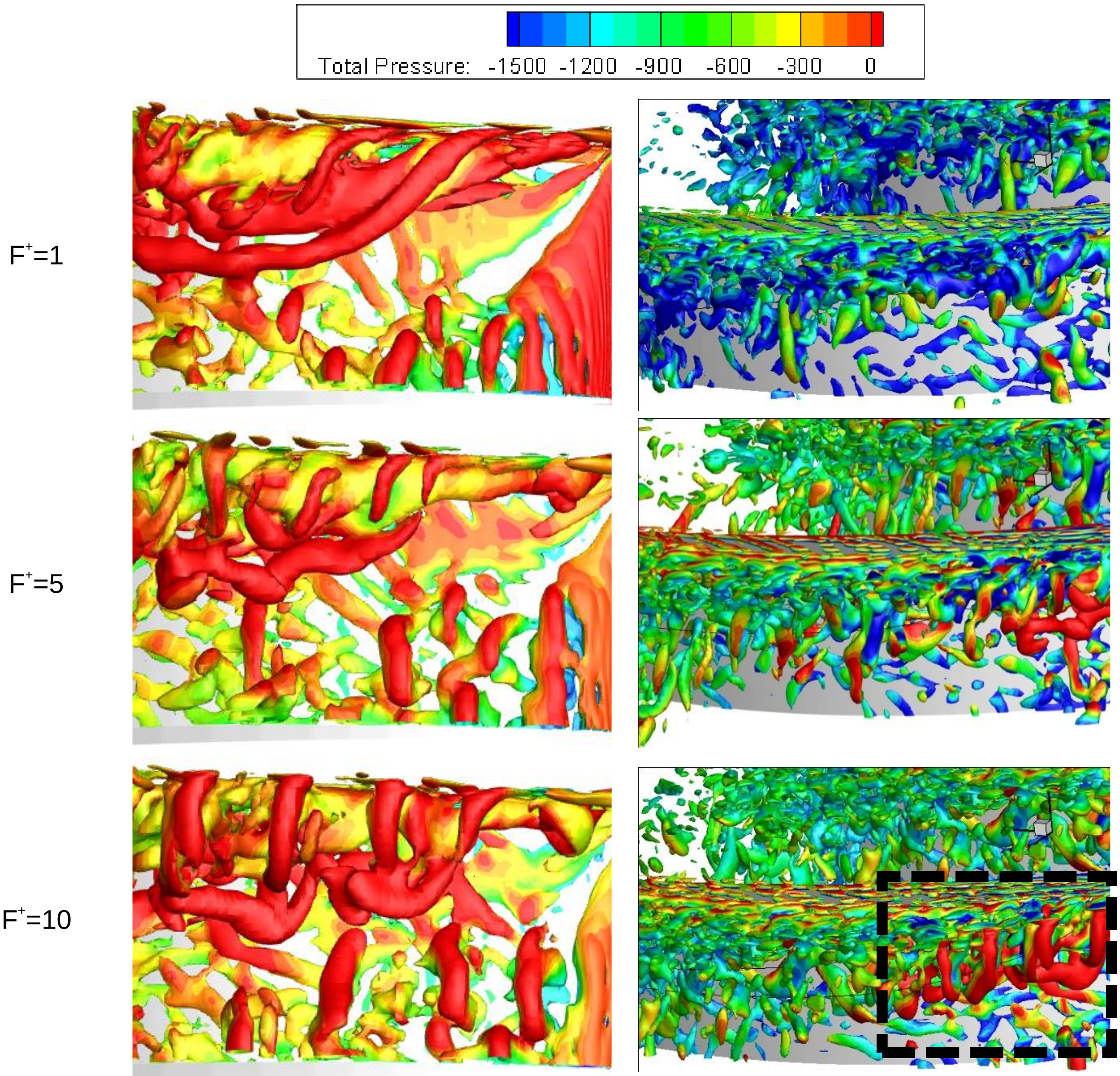


Figure 10 - Transient Q Criterion Iso-surface Near the Blade Tip Clearance

From Figure 10, it is evident that in the excitation state of F+=10, the strip of the tip clearance vortex is most intact, whereas in the excitation state of F+=1, the strip of the tip clearance vortex is most fragmented, and even more so compared to the uncontrolled state. This indicates that the excitation at F+=10 enhances the orderliness of the tip clearance vortex, while the excitation at F+=1, conversely, disrupts the orderliness of the tip clearance vortex. From the figure, it can be observed that the tip clearance vortex remains intact, leading to higher total pressure upon detachment from the blade back, and its involvement enhances the orderliness of the main leakage vortex compared to the uncontrolled state. It can be inferred that the unsteady excitation at the blade tip leading edge resonates with the tip clearance vortex, causing it to detach more orderly from the blade back and generate the secondary leakage vortex. The enhanced orderliness of the secondary leakage vortex, in turn, is entrained into the main leakage vortex, ultimately enhancing the orderliness of the leakage vortex.



(a) From the Leading Edge to 50% Axial Chord Length

(b) From 50% to 90% Axial Chord Length

Figure 11 - Transient Q Criterion on the Backside of the Blade

From Figure 11, it can be observed that in the excitation state of F^{+}=1, the secondary leakage vortex wraps around the primary leakage vortex with a large pitch and radius. In contrast, in the excitation state of F^{+}=10, both the pitch and radius of the wrapping structure significantly decrease. The secondary leakage vortex in this wrapping structure wraps around the primary leakage vortex without merging with it. This objectively delays the increase and breakup of the scale of the primary leakage vortex, thereby postponing the downstream propagation of the fragmented losses and ultimately delaying the blockage of the blade tip flow field. It is also evident from Figure 11 that under F^{+}=1 excitation, there is a noticeable increase in total pressure on the blade back vortex structure. This indicates that applying unsteady excitation at the blade tip leading edge with the same frequency as the blade back separation vortex can resonate with the blade back separation vortex. However, its effect on the blade tip flow field is extremely limited.

4. Conclusion

This study investigated the effect of varying excitation frequencies on the unsteady flow control effectiveness by placing a periodic jet hole at the leading edge corresponding position of the casing wall in a planar cascade model with blade tip clearance under large expansion ratio condition. The Large Eddy Simulation (LES) method was employed to explore the mechanism of unsteady flow control by periodic jet injection at the rotor leading edge tip to enhance the stability margin of the compressor. The main conclusions are as follows:

(1)Although the excitation frequency has minimal impact on the total pressure loss at the trailing edge section, excitation at a frequency matching the dominant frequency of the tip clearance vortex can significantly reduce the scale and intensity of the blade tip leakage vortex alongside. This results in alleviation of surge and modal wave stall.

(2)Excitation at a frequency matching the dominant frequency of the tip clearance vortex enhances the orderliness of the blade tip clearance vortex. This leads to the formation of secondary leakage vortex, which wrap around the primary leakage vortex with smaller pitch and radius. Ultimately, this reduces the overall scale and delays the breakup of the leakage vortex.

(3)Although excitation at a frequency matching the dominant frequency of the separation vortex can resonate with the separation vortex, it suppresses the orderliness of the blade tip clearance vortex, thus unable to effectively act on the leakage vortex.

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