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Investigation on the Mechanism of Blade Tip Recess Improving the Aerodynamic Performance of Transonic Axial Flow Compressor

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ABSTRACT

The development of high-performance aero engines put forward higher performance requirements for compressor components, and the improvement of aerodynamic performance of compressors has important engineering application value. The blade tip recess has great potential and advantages in improving the aerodynamic performance of compressors. In order to better understand the effect of the blade tip recess on the compressor aerodynamic performance, in this paper, the influence mechanism of the blade tip recess on the aerodynamic performance of the isolated rotor of a transonic axial compressor stage is discussed. Under the premise that the numerical method's results are almost consistent with the experimental test results, the full three-dimensional unsteady numerical results show that the main reason for the original blade rotor stall is the leading edge of the blade tip blockage, which is caused by blade tip clearance leakage vortex breakage. After adopting the measures of the blade tip recess, the study shows that the blade tip recess can increase the rotor stall margin by 2.10% without reducing the rotor efficiency and the total pressure ratio. A detailed analysis of the blade tip flow field shows that the blade tip recess can reduce the intensity of the tip clearance leakage flow by increasing the turbulence intensity of the blade tip near the casing wall, and reduces the leading edge of blade tip blockage, improves the rotor blade tip flow field, thereby achieving the purpose of enhancing rotor stability.

Keywords: Axial flow compressor; Blade tip recess; Aerodynamic performance; Stability.

NOMENCLATURE

$A_{Blockage\ area}$	blade tip blockage area
$A_{passage\ area}$	blade tip passage area
μ_t	turbulent viscosity
B	blade tip blockage coefficient
C	distance between suction surface of and pressure surface
Ca	normalized axial chord length
h	blade tip clearance
m_{Origin}	near stall point mass flow of origin schemes
m_{TR}	near stall point mass flow of blade tip recess schemes
LE	leading edge of the blade
PS	Pressure Surface
RD	Recess Depth
RL	Recess Location
RW	Recess Width
SMI	Stall Margin Improvement
SS	Suction Surface
T	time steps
TE	trailing edge of the blade
TR	blade tip recess scheme
W	velocity of the blade tip clearance leakage flow
π_{Origin}	near stall point total pressure ratio of Origin schemes
π_{TR}	near stall point total pressure ratio of blade tip recess schemes
μ	dynamic viscosity

1. INTRODUCTION

The rapid development of aviation technology, there is an increasing demand for the performance of aviation engines. The compressor is an important part of the aero engines, improving the aerodynamic performance of the compressor is particularly important for improving the performance of the aero engines. However, the working performance curve of the compressor is often limited by the surge and stall boundary, so it is of great significance to improve the stall margin of the compressor, improve the aerodynamic stability of the compressor, and ensure the safety stable work of the compressor. (Hoying *et al.* 1998; Davis *et al.* 2006)

At present, there are mainly two methods for improving the stall margin of the compressor: the active control method and the passive control method. Active control methods include blade tip inject (Li *et al.* 2019; Wang *et al.* 2017), plasma excitation (Degiorgi *et al.* 2016), etc. Passive control methods include slot-type casing treatment (Zhang *et al.* 2020), groove-type casing treatment (Liu *et al.* 2016), end wall molding technology (Chu *et al.* 2016), etc. Compared with the active control method, the passive control method has the advantages of simple structure, easy implementation and significant enhancing stability effect, so it is favored by the majority of researchers. However, numerous research results have shown that while the passive control method improves the compressor stall margin, it often causes a decrease in compressor efficiency (Hembera and Danner 2008; Rabe and Hah 2002; Bailey 1972; Fujita and Takata 1984; Takata and Tsukuda 1977; Moore *et al.* 1971; Nezym 2004; Wilke and Kau 2002). Therefore, looking for a flow control method while improving the compressor stall margin, it has little effect on the compressor efficiency, that is the goal pursued by the researchers.

The blade tip recess is the passive flow control measure. It was first used on the turbine blade to cool the turbine blade, had been widely used and studied on the turbine blade (Ameri *et al.* 1998; Key and Arts 2006; Mischo *et al.* 2008; Ahmed and Jehanzed 2009). However, the application and research of blade tip recess on compressor blades are relatively few. Gourdain and Lebueof (2009) conducted a study on the blade tip recess to improve the compressor rotating stall on a subsonic axial compressor, found that the blade tip recess can increase the turbulent intensity near the casing and improve the compressor stall margin. Khan *et al.* (2012) carried out a study on the combination of circumferential grooves, blade tip recess and blade tip inject models to improve the performance of transonic axial compressors. The study found that the combination of circumferential grooves, blade tip recess and blade tip inject models could be obtained better enhancing stability effect. Ma *et al.* (2012) studied the effects of grooved tip clearances on the flow field in a compressor cascade passage, found the pressure gradient from the pressure side to the suction side on the blade tip is reduced due to the existence of the grooved tip, the leakage flow is weakened and the high-blockage and high-loss region caused by the leakage flow is narrower with the grooved tip. Jung *et al.* (2016) conducted a study on the blade tip recess to improve the compressor stall margin on a transonic axial compressor. The study found that a reasonable choice of the length and depth of the blade tip recess is conducive to the improvement of the compressor stall margin.

The study of the blade tip recess is mainly carried out on the turbine blade. The influence of the blade tip recess on the performance of the compressor is relative less studied. The physical mechanism of the blade tip recess affecting the performance of the compressor has not been fully understood, which restricts the application and development of the blade tip recess. In this paper, the isolated rotor of transonic axial compressor is taken as the research object, and the mechanism of the blade tip recess to improve the performance of the compressor is investigated to provide a reference for the application and development of the blade tip recess.

2. NUMERICAL METHOD CHECK AND BLADE TIP RECESS SCHEME DESIGN

2.1 Research Object

The research object is the isolated rotor NASA Rotor 35 of transonic axial compressor stage. Table 1 gives the relevant performance parameters and design parameters of the rotor.

Table 1 Relevant performance and design parameters of the rotor

parameters	Value
Design mass flow/（kg/s）	20.20
Design total pressure ratio	1.865
Design adiabatic efficiency	0.8650
Blade tip velocity/（m/s）	455
Blade number	36
Blade tip relative Mach number	1.49
Solidity of blade tip	1.3
Design rotation speed/（r/min）	17188.7

2.2 Check by Numerical Method

The commercial fluid mechanics software NUMECA is used to perform the full three-dimensional numerical steady and unsteady calculation, unsteady calculation is carried out when the rotor is working at the near stall condition, for other work conditions, steady calculation is adopted. The flow field solution is obtained by solving the three-dimensional Reynolds average Navier-Stokes equation. The mesh topology of rotor is that the inlet and outlet are H-shaped grids, the blade adopts O-4H grid topology. The blade surface is O-shaped grid, the H-shaped grid surrounds the O-shaped grid, the number of grid points is 153, 73, 49 along the axial, radial, and circumferential directions, respectively. The blade tip clearance adopts the butterfly mesh topology structure, the radial grid points are 21, and the rotor single-passage grid points are about

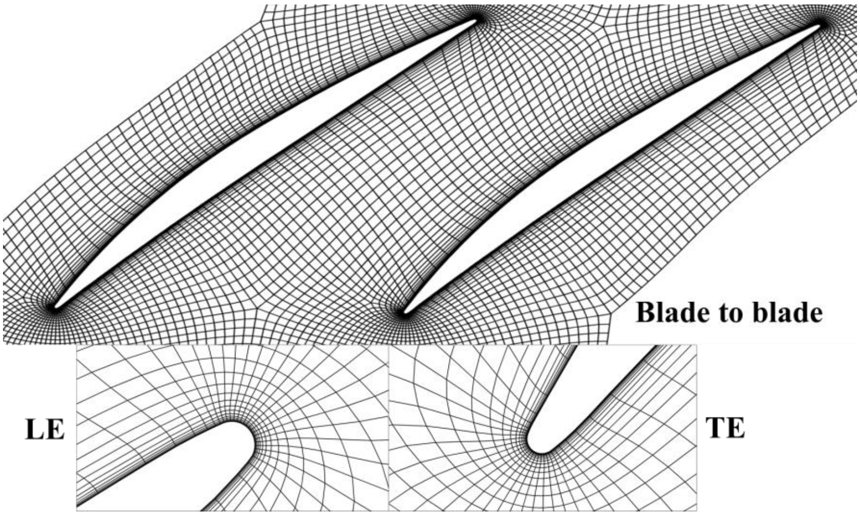


Fig. 1. Schematic diagram of the mesh topology of the rotor passage.

800,000. The mesh topology of the leading and trailing edge, blade to blade cross surface of the rotor is shown in Fig.1. The calculation boundary conditions are set as follows, the rotor inlet is given a total temperature (288k) and a total pressure (101325Pa), the outlet is given an average static pressure, and the wall is an adiabatic, non-slip boundary. The Spalart-Allmaras is used as the calculation turbulent model, the difference format is the second-order upwind style, and the physical time step is 4.8484×10^{-6} s, the internal iteration time step is 20. The performance curve of the rotor is calculated by gradually increasing the back pressure of the rotor outlet. When the rotor is close to the near stall boundary, the outlet back pressure is increased by 100Pa in turn until the stable convergence solution cannot be obtained.

Figure 2 is the comparison between the numerical calculation and the experiment. The numerical calculation of the rotor's total performance characteristic curve is basically consistent with the experimental test value, the difference is that the numerical calculation near-stall point mass flow and the total pressure ratio are slightly lower than the experimental test value, but if taking into account the experiment measurement error of the mass flow, the numerical calculation of the rotor's near-stall point mass flow is basically consistent with the experiment.

Figure 3 shows the comparison of the distribution of the total temperature ratio and the total pressure ratio at the rotor outlet along the blade height under the near stall point. Below 80% blade height, the distribution of numerical calculation total pressure ratio and total temperature ratio along the blade height is almost consistent with the experiment. While above 80% blade height, there is difference between the numerical calculation and the experiment. This is because when the rotor is closer to the stall condition, the boundary layer flow and the shock wave will occur strongly interactions with various vortex systems in blade tip region, the flow of blade tip region show a stronger unsteady. While the maximum measurement point of the radial position during the experiment is 95% blade height, and the flow capture ability of the blade tip region of the experiment is slightly insufficient, which may be one of the reasons for the difference between the numerical calculation and the experiment. In a word, within the tolerance of the error, the prediction of the rotor performance by numerical calculation is basically consistent with the experimental test. The numerical method can be used to study the effect of the blade tip recess on the compressor performance.

2.3 Blade Tip Recess Design

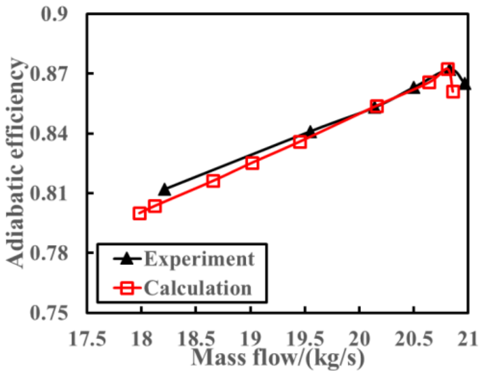
The blade tip surface is recessed and form the blade tip recess. Figure 4 shows the two-dimensional and three-dimensional structural diagrams of the blade tip recess. The structural parameters of the blade tip recess mainly include the recess width (RW), the recess depth (RD) and the recess location (RL). In this paper, the research on the blade tip recess improves the performance of the axial compressor, the design of the blade tip recess is that the RW is 80% C, and the RD is 50 % h, the RL takes the blade suction surface as the reference point, and the distance from the center of the recess to the reference point is 50% C. The range of blade tip recess in the flow direction is 100%Ca, the number of grid points is 401, 21, 17 along the axial, radial, and circumferential directions, respectively. The full non-matching connection is adopted between the top of the blade tip recess grid and the blade passage grid.

3. RESULTS AND DISCUSSIONS

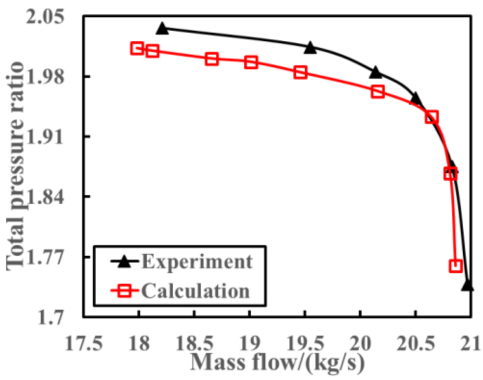
3.1 Analysis of Compressor Performance

Figure 5 shows the comparison of the total performance characteristic curve of with and without blade tip recess. The near-stall point of the without blade tip recess is marked in Fig.5 (b). In the Fig.5, Origin represents the without blade tip recess scheme, and TR represents the blade tip recess scheme. Compared with the Origin scheme, after adopting the blade tip recess measure, the efficiency characteristic curve and the total pressure ratio characteristic curve of the rotor will hardly change during the throttling of the rotor, indicating that the blade tip recess will not change the rotor efficiency and total pressure ratio characteristics. Looking at the change of the rotor near-stall point mass flow rate, the near-stall point mass flow rate of the TR scheme is lower than the Origin scheme, indicating that the blade tip recess widens the rotor's stable working mass flow range，that is beneficial to the stability of the rotor. The stall margin improvement (SMI) is used to characterize the contribution of blade tip recess to rotor stability, which is defined as follows:

$$\mathrm{SMI} = \frac{\pi_{TR}}{\pi_{Origin}} \times \frac{m_{Origin}}{m_{TR}} - 1 \tag{1}$$



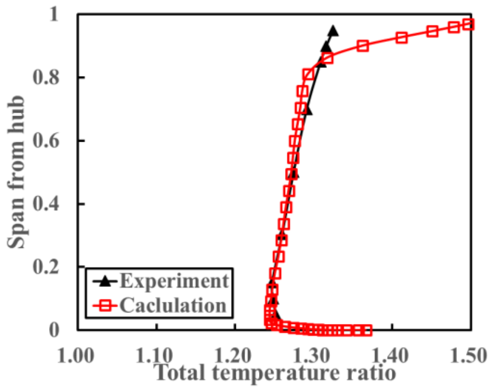
(a)Adiabatic efficiency



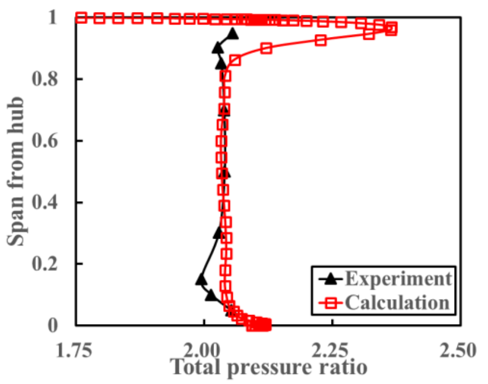
(b)Total pressure ratio

Fig. 2 Comparison of numerical calculation and experimental results (total performance characteristic curve).

In the formula, π_{Origin} and m_{Origin} represent the total pressure ratio and mass flow at the near-stall point of the Origin scheme, respectively. π_{TR} and m_{TR} represent the total pressure ratio and mass flow at the near-stall point of the rotor after adopting the blade tip recess measure. According to the calculation, after the blade tip recess is used, the rotor stall margin improvement is 2.10%. Compared with other passive flow control methods, the enhancing stability effect of the blade tip recess is not as good as that of the casing treatment, but while it enhances the stability of the compressor, there is not sacrifice compressor efficiency like other casing treatment, it shows that the blade tip recess has good engineering application value. The following will analyze the flow mechanism of the blade tip recess to improve the stability of the axial compressor rotor.



(a)Total temperature ratio



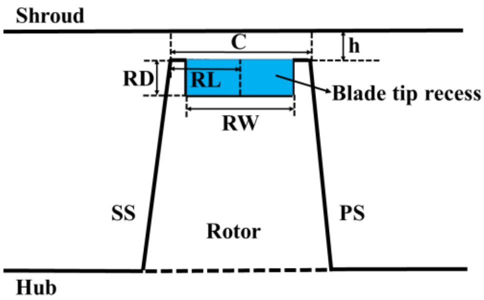
(b)Total pressure ratio

Fig. 3 Comparison of numerical calculations and experimental results (total temperature ratio and total pressure ratio at near stall point).

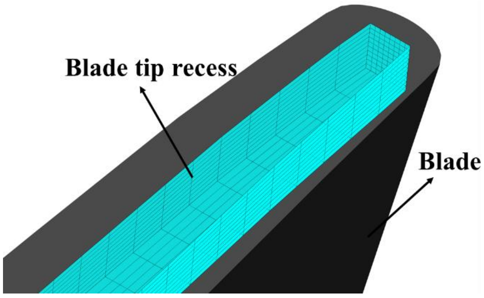
3.2 Analysis of flow Field in the Compressor

3.2.1 Analysis of Origin Scheme Blade Tip Flow Field

Figure 6, Fig. 7 and Fig. 8 give cloud diagrams of the tip clearance leakage flow line, high-entropy area of the blade tip and relative Mach number of the blade tip at different moments when the Origin scheme is under near-stall mass flow conditions. Due to the pressure difference between the pressure surface and the suction surface of the rotor blade, the airflow will spontaneously flow from the pressure surface through the tip clearance to the suction surface, forming the tip clearance leakage flow. The leakage flow from the blade tip clearance quickly rolls up in the rotor passage after encountering the blade tip inflow, that will form the tip clearance leakage vortex near the pressure surface of the blade, the tip clearance leakage vortex may occur breakdown in this region, thereby forming the low velocity flow area in the Fig.6. Comparing the changes of the tip clearance leakage flow lines of the rotor passage at near stall work conditions at different times, it was found that from T/4 to 4T/4, the tip clearance leakage flow lines near the pressure surface are intertwined, and the fracture occurred, all forming a large low velocity flow area.



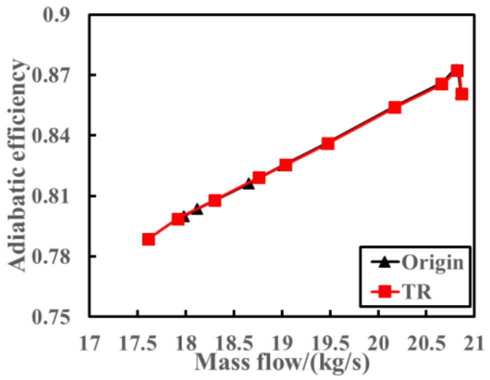
(a) Two-dimensional structure diagram



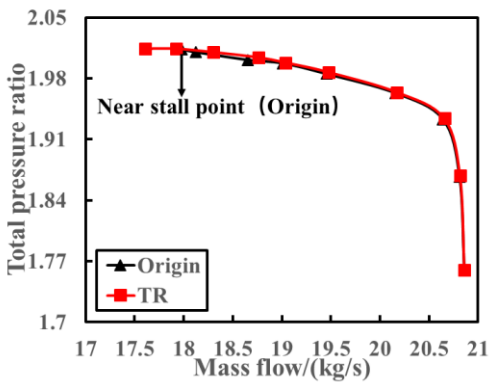
(b) Three-dimensional structure diagram

Fig. 4. Schematic diagram of the blade tip recess structure.

The generation of entropy is irreversible. Once it is generated, it will generate irreversible flow loss. The generation of entropy is closely related to the flow loss. When vortex breakdown occurs in the flow field, the flow loss of the vortex breakdown region is relative larger, and the corresponding entropy value is also larger. It can be seen from the Fig. 7 that there is a large number of high-entropy region in the leading edge of the blade tip, which spans the entire rotor passage. In addition, the high-entropy region corresponds to the trajectory of the tip clearance leakage flow line. In the rotor passage, from the suction surface to the pressure surface, it can be seen that the high-entropy region decreases first and then increase. The high-entropy region near the suction surface is due to the large tip clearance leakage velocity, which collides with the blade tip inflow. The high-entropy area near the pressure surface is due to the tip clearance leakage vortex occur breakdown in this region, forming the large number of low velocity flow regions, where the fluid swirls around, forming the high-entropy area. Comparing the changes of the high-entropy region of the blade tip at different times, it was found that from the T/4 to 4T/4, the high-entropy region of the blade tip gradually gathered toward the middle of the rotor passage, indicating that the tip clearance leakage vortex breakdown region gradually moved toward the center of the rotor passage. At the time of 4T/4, a large area of high entropy appeared in the middle of the rotor passage, indicating that the tip clearance leakage vortex breakdown region occurred in the middle region of the rotor passage. Breakdown of the tip clearance leakage vortex will inevitably form a low velocity flow blockage area in the flow field. At this time, it can be seen from Fig. 8 that there are a large number of low velocity flow blockage areas near the pressure surface of the blade tip leading edge, and with time from T/4 to 4T/4, the blockage area of the blade tip leading edge still occupies more than half of the rotor passage, and gradually move toward the middle of the rotor passage, blocking the incoming flow into the rotor passage, making the blade tip flow field deteriorated seriously, and finally put the rotor into an unstable working condition, triggering the rotor stall. In addition, the change of the low velocity flow blockage area of the blade tip leading edge in Fig. 8 corresponds to the change in the high-entropy area in Fig.7, which indirectly indicates that the blockage area of the blade tip leading edge is caused by the breakdown of the tip clearance leakage vortex. The type of rotor stall is a typical blade tip leading edge blockage, and the rotor stall has a very important relationship with the tip clearance leakage flow.



(a) Adiabatic efficiency



(b) Total pressure ratio

Fig. 5. Comparison of the total performance characteristic curve of the TR scheme and the Origin scheme.

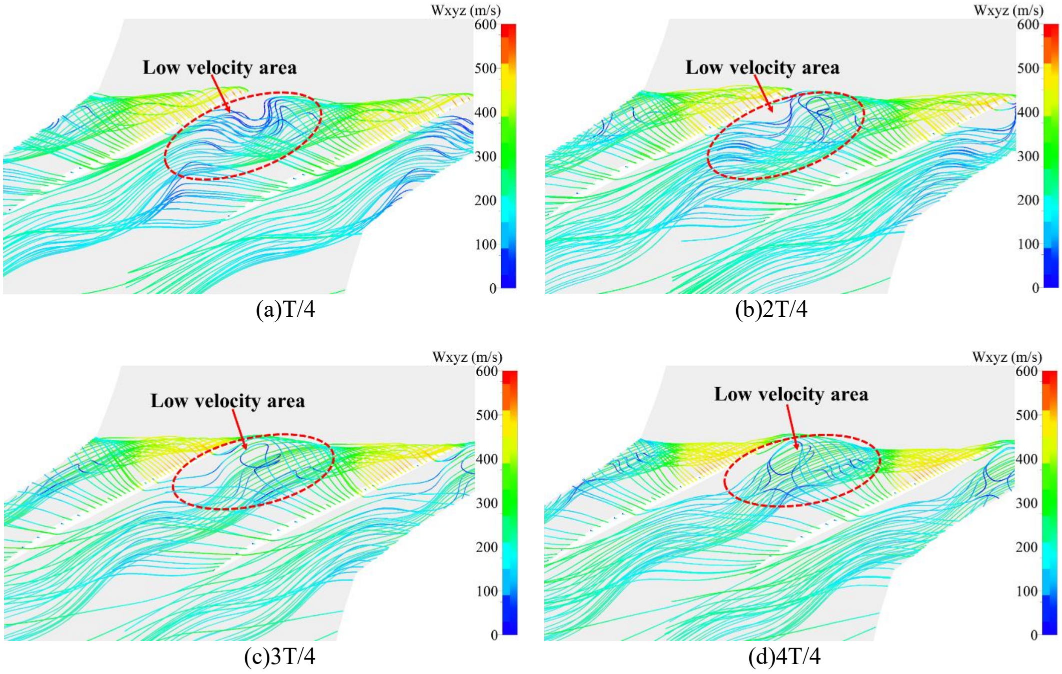


Fig. 6. Comparison of the distribution of tip clearance leakage flow at different times when the rotor is near the stall point (Origin scheme, 98% blade height).

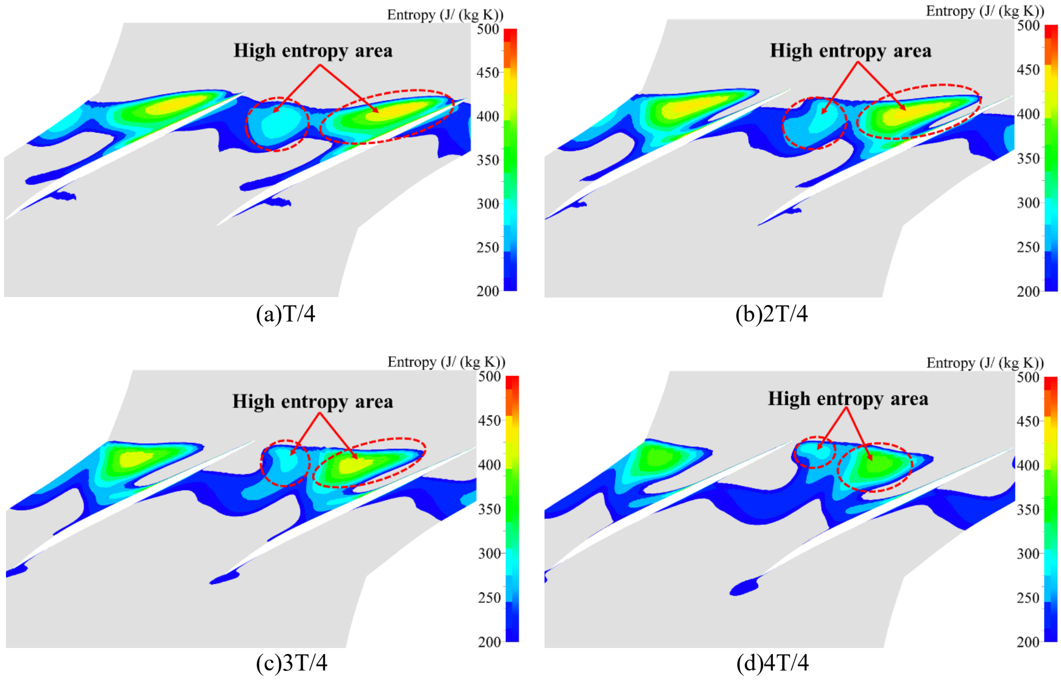


Fig. 7. Comparison of the distribution of high entropy area of blade tip at different times when the rotor is near the stall point (Origin scheme, 98% blade height).

3.2.2 Analysis of the flow Mechanism of Blade Tip Recess to Improve Rotor Stability

In order to clarify the change of the flow field in the blade tip region after adopting the measures of the blade tip recess, under the near-stall mass flow of the blade tip recess scheme is consistent with the Origin scheme, Fig.9 and Fig.10 show the comparison of the tip clearance leakage flow line and the high-entropy region cloud diagram at different times after using the blade tip recess. It can be seen from Fig. 9 that after the blade tip recess measure are taken, the leakage flows from the tip clearance are twisted and intertwined near the leading edge of the blade tip pressure surface, and the leakage flow line breaks in this region, forming a low velocity area, and the low velocity area near

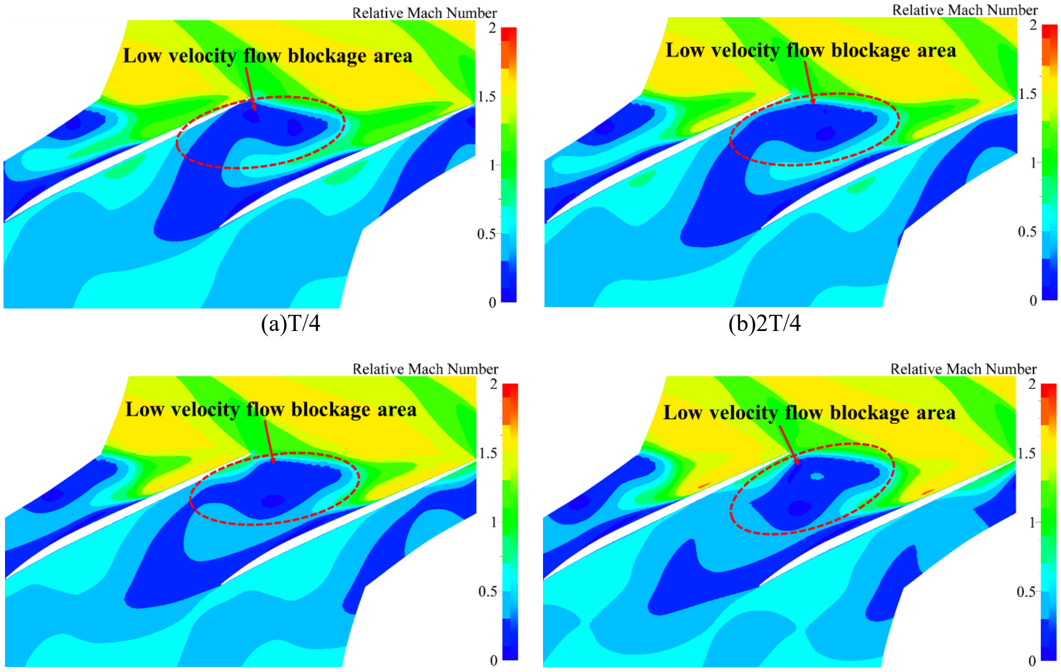


Fig. 8. Comparison of the distribution of Relative Mach Number of blade tip at different times when the rotor is near the stall point (Origin scheme, 98% blade height).

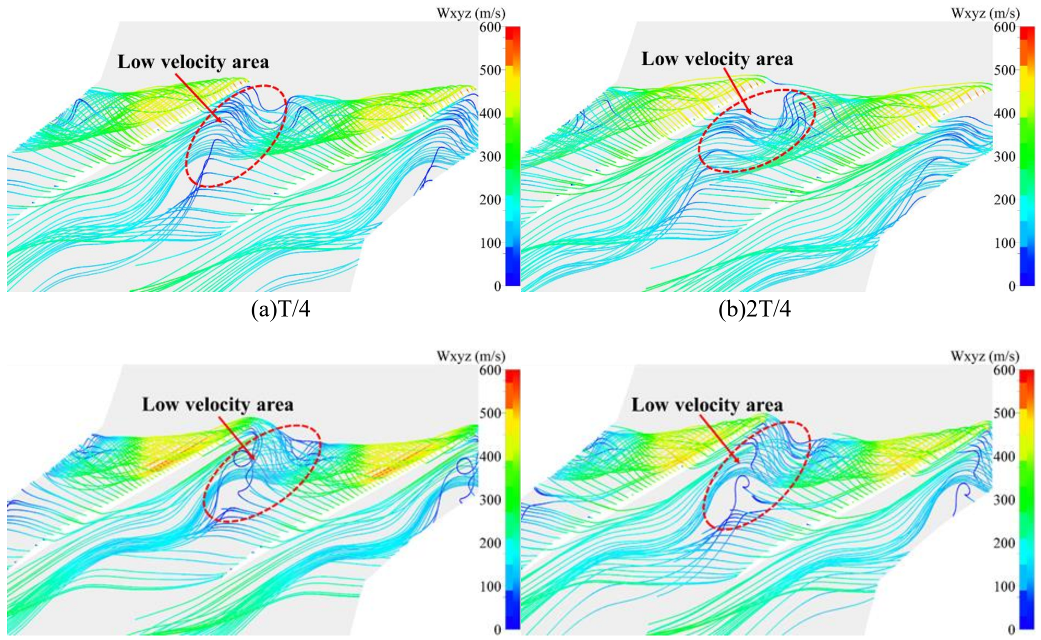


Fig. 9. Comparison of the distribution of tip clearance leakage flow at different times when the rotor is near the stall point (blade tip recess scheme, 98% blade height).

the leading edge of the blade tip pressure surface at different times is similar. At this time, it can be seen from Fig.10 that the distribution of the high-entropy region of the blade tip recess is similar to the Origin scheme, there are still two high-entropy regions in the blade tip, and from the suction surface to the pressure surface, the high-entropy region decreases first and then increase. From T/4 to 4T/4, the high-entropy region gradually moves toward the center of the rotor passage. Compared with Fig. 7, it can be found that the high-entropy area of the blade tip is significantly reduced after adopting the blade tip recess measure, indicating that the tip clearance leakage flow intensity and the tip clearance leakage vortex breakdown area are both reduced. The application of the blade tip recess suppresses the tip clearance leakage flow intensity, which reduces the breakdown area of the tip clearance leakage vortex,

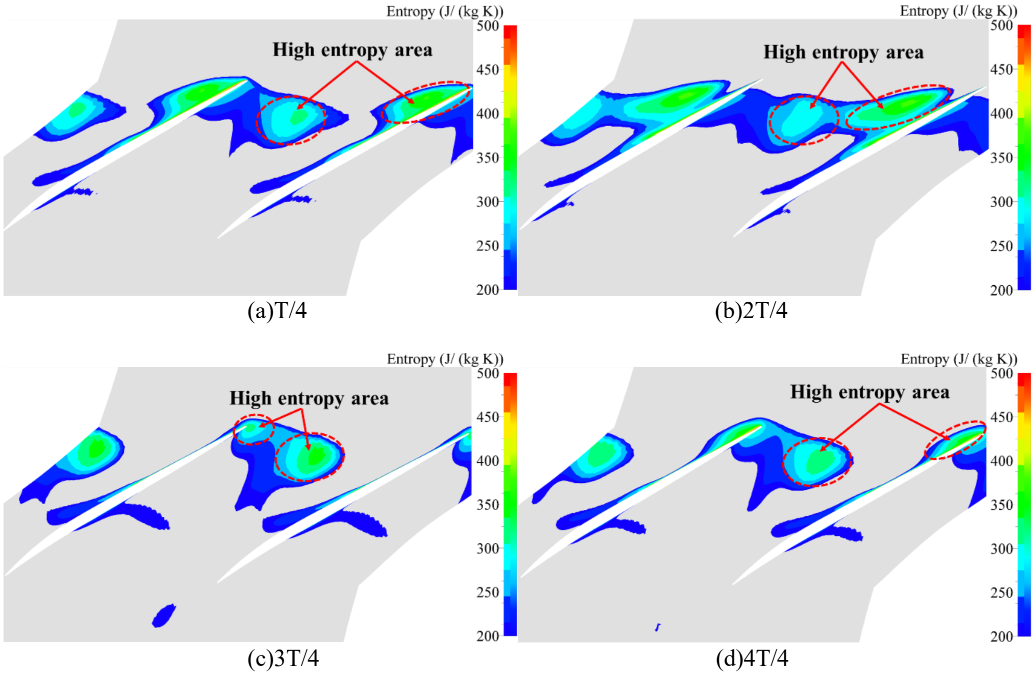


Fig. 10. Comparison of the distribution of high entropy area of blade tip at different times when the rotor is near the stall point (blade tip recess scheme, 98% blade height).

which is beneficial to reduce the flow blockage at the leading edge of the blade tip, thereby improving the flow field of blade tip area, it is beneficial to improve rotor stability.

The blade tip recess can improve the blade tip flow field by suppressing the blade tip clearance leakage flow intensity and reducing the blade tip clearance leakage vortex breakdown area, thereby enhancing the rotor stability. In order to study the physical mechanism of the blade tip recess weakening the tip clearance leakage flow intensity, under the Origin scheme and the blade tip recess scheme are the same mass flow condition, Fig.11 shows comparison of the distribution of the turbulent viscosity ratio cloud diagram and flow line in the tip clearance at different axial chord length. In the Fig.11, SS represents the suction surface of the blade, and PS represents the pressure surface of the blade (the left picture is the Origin scheme, and the right picture is the blade tip recess scheme).

Boussinesq proposed that the Reynolds stress is caused by the chaotic movement of fluid micelles. The turbulent transport can be expressed as an increase in the turbulent viscosity ratio, so the size of the turbulent viscosity ratio can be used to measure the intensity of turbulent transport in the flow field. It can be seen from Figs.11 (a)-(d) that after the blade tip recess is adopted, the turbulent viscosity ratio increases in the blade tip clearance, it means that the application of the blade tip recess increases the turbulent transport intensity in the blade tip clearance, and the fluid turbulent intensity is larger in the blade tip clearance. When the position of the chord length changes from 7.5%Ca to 37.5%Ca, it can be found that the growth rate of the turbulent viscosity ratio gradually increases in the tip clearance, indicating that the increase of turbulent intensity in the tip clearance gradually increases along the flow direction. Change of the intensity of fluid turbulent will inevitably lead to the flow change of airflow in the tip clearance.

It can be seen from Fig. 11 (a) that after the blade tip recess measure are taken, when the airflow flows from the pressure surface to the suction surface through the blade tip clearance, a counterclockwise vortex will be formed inside the blade tip recess. The exchange of momentum and velocity between the vortex and the main flow of the tip clearance causes the flow lines to slightly deflect toward the hub direction, changing the direction of the tip clearance leakage flow and increasing the turbulent intensity of the air flow. In Figs.11(b)-(d), the flow inside blade tip clearance is similar in this axial chord length range. The main flow of the tip clearance is affected by the recess, and the main flow is slightly deflected toward the hub direction, and when the chord length position changes from 7.5% Ca to 37.5% Ca, the degree of main flow deviation toward the hub gradually becomes larger. It can be clearly found in Fig.11(d) that the main flow of the tip clearance is affected by blade tip recess and deflects toward the hub direction.

Adopting the blade tip recess design at the rotor blade tip can increase the turbulent intensity of the airflow in the blade tip clearance, that changes the flow direction of the main flow of blade tip clearance. In order to explore the suppression effect of the blade tip recess on the tip clearance leakage flow intensity, under the Origin scheme and the blade tip recess scheme are the same mass flow condition, Fig.12 shows comparison of the distribution of the velocity of the tip clearance leakage flow with the time step at different axial chord lengths. It can be found from Fig.12 that after the blade tip recess is

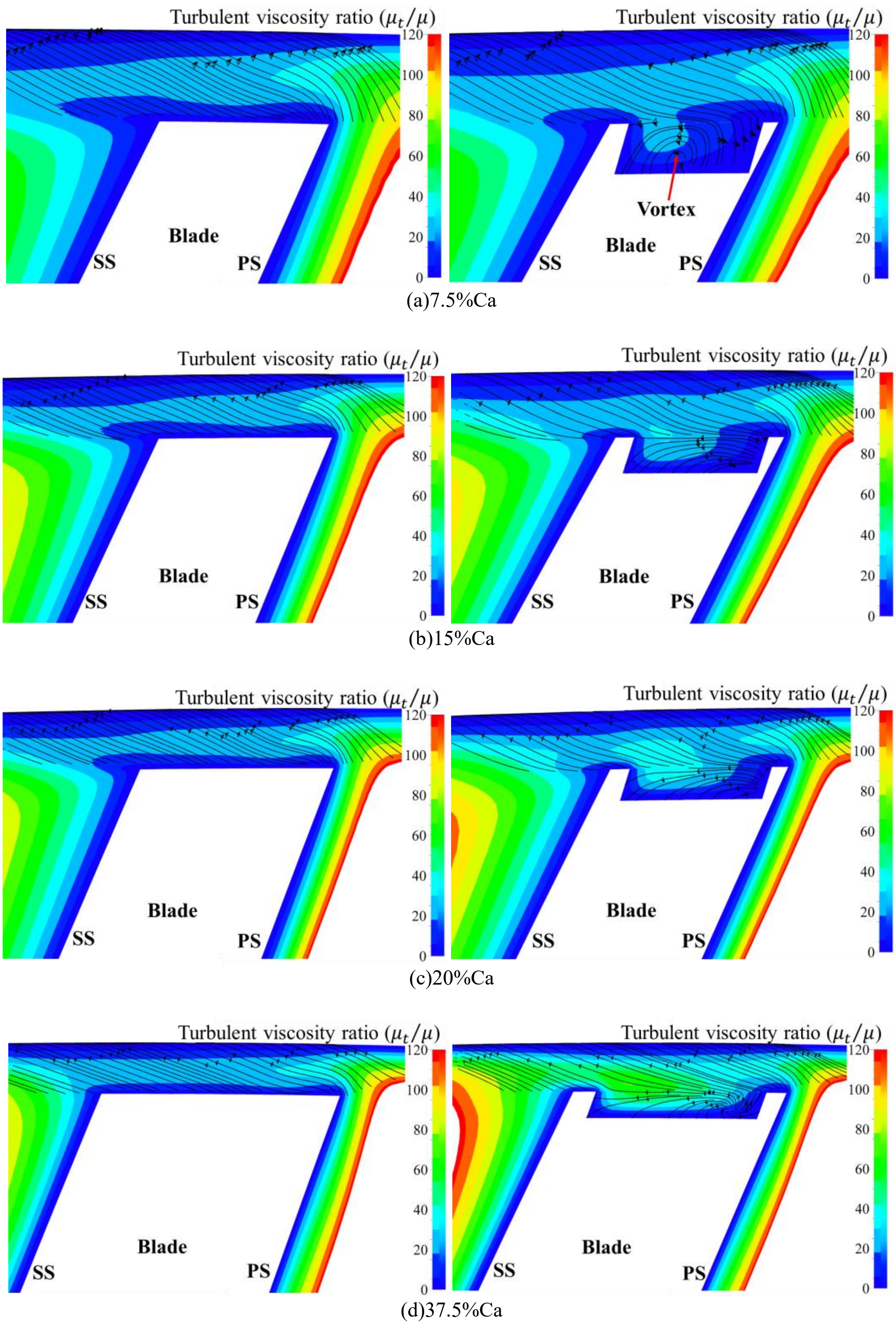
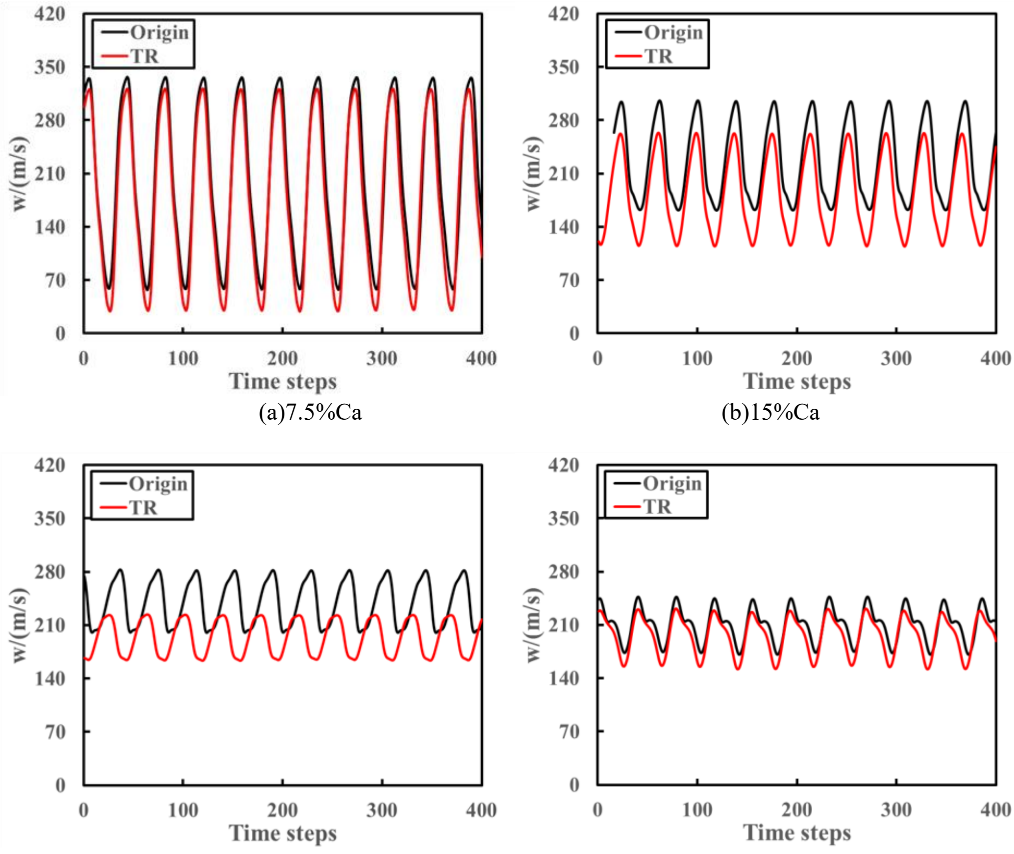


Fig. 11. Comparison of the distribution of the turbulent viscosity coefficient cloud diagram and flow line in the tip clearance at different axial chord lengths when the Origin scheme and the blade tip recess scheme are the same mass flow condition.

used, the velocity of the tip clearance leakage flow is reduced, indicating that the blade tip recess can effectively reduce the intensity of the blade tip clearance leakage flow. Comparing the changes of the tip clearance leakage flow at different axial chord lengths, it was found that as the chord length changes from 7.5% Ca to 37.5% Ca, the tip clearance leakage flow velocity gradually decreased. After adopting the measures of the blade tip recess, the decreasing rate of the tip clearance leakage flow velocity



(a)7.5%Ca (b)15%Ca

(c)20%Ca (d)37.5%Ca

Fig. 12. Comparison of the distribution of the velocity of the tip clearance leakage flow with the time step at 10% h from the blade tip suction surface at different axial chord lengths when the Origin scheme and the blade tip recess scheme are the same mass flow condition.

increase first and then decrease, the effect of suppressing the tip clearance leakage flow velocity is the best at 20%Ca, indicating that the suppression effect of the tip clearance leakage flow intensity also increases first and then decrease. Reasonable selection of the recessed axial chord length range is beneficial to the recess effect, It is known from the previous analysis that the cause of the rotor stall is the leading edge of the blade tip blockage. In order to quantitatively describe the change of leading edge of the blade tip blockage after applying the blade tip recess, the blockage coefficient B of the leading edge of the blade tip is defined as follows:

$$B = \frac{A_{Blockage\ area}}{A_{Passage\ area}} \qquad (2)$$

In the formula, define the area where the relative Mach number is less than 0.3 as the blockage area, $A_{Blockage\ area}$ represents the blockage area of the blade tip leading edge, and $A_{Passage\ area}$ represents the rotor passage area. Figure 13 shows the comparison of the blockage coefficient of the blade tip leading edge between the Origin scheme and the blade tip recess scheme during throttling. From the Fig.13, it can be found that in large mass flow conditions, due to the smaller blade tip blockage area, the blockage coefficient B of the leading edge of the blade tip will not almost change after using the blade tip recess. In the case of small mass flow conditions, the tip clearance leakage is very serious at this time, and the low velocity flow blockage area caused by the tip clearance leakage vortex breakdown is larger. At this time, the blade tip recess suppresses the tip clearance leakage flow intensity, and make the tip clearance leakage vortex breakage area reduce, thereby reducing the blockage degree of the leading edge of the blade tip, and the blockage coefficient B of the leading edge of the blade tip is smaller than that of the Origin scheme. In a word, it is known that the blade tip recess reduces the blockage degree of the blade tip by suppressing the tip clearance leakage flow intensity, improves the rotor blade tip flow field and increases the stability of the rotor.

4. CONCLUSION

In this paper, the effect of the blade tip recess on the performance of the isolated rotor of the transonic axial compressor stage is investigated. The main mechanism of the blade tip recess on the rotor stability is studied. Through detailed data analysis and flow field analysis, several conclusions are summarized as follows:

1. The blade tip recess can improve the rotor stability without reducing the rotor efficiency

and the total pressure ratio. The blade tip recess scheme studied in this paper can improve the rotor stall margin improvement by 2.10%.

2. Under the Origin scheme, the leading edge of the blade tip blockage caused by the tip clearance leakage vortex breakage is the main reason for the deterioration of the blade tip flow field. The rotor stall type is the blade tip blockage.

3. The blade tip recess suppresses the tip clearance leakage flow intensity by increasing the turbulence intensity of the air flow near the casing wall, and changes the flow direction of the main flow in the blade tip clearance, thereby reducing the blade tip clearance leakage vortex breakdown area and weakening the blockage degree of the blade tip leading edge, improves the rotor blade tip flow field and increases the rotor stability.

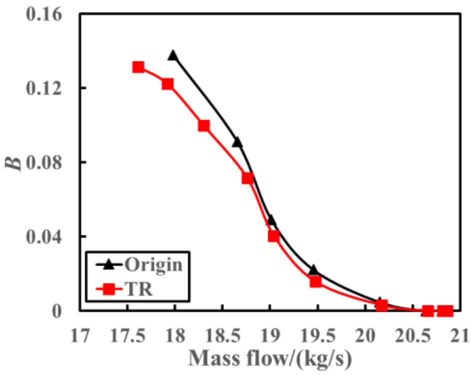


Fig. 13. Comparison of the blockage coefficient of the leading edge of the blade tip during the throttling process between the Origin scheme and the blade tip recess scheme.

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