A Study on the Nozzle-Rotor Interactions of Partial Admission Supersonic Turbines

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Abstract

The performance characteristics of partial admission supersonic turbines are analyzed by using the commercial CFD program FLUENT6.0. The governing equations were discretized with Euler implicit method in time and 2nd-order upwind scheme of FVM in space. The k-ε turbulence model was utilized to describe the turbulent flow field.
In order to investigate the nozzle–rotor interactions and the effect of partial admission, the flows in supersonic turbine rotor cascades with a nozzle are computed. Extensive computations of partial admission supersonic turbines provide the shock structures and flow patterns in the nozzle and rotor. It is clearly shown that the nozzle flow is highly affected by the shocks or expansion waves propagated from the rotor leading edge. And the rotor flow is also affected by the shocks or wakes originated from the nozzle.

Introduction

In the liquid rocket or induced weapons, the propellant feed system is essential. This system often includes the high-pressure turbo-pump system. The turbine of a turbo-pump system is usually operated at supersonic condition due to its high loading characteristics. And Partial admission axial turbine generates very high power output even though it is small and light weighted. So is widely used for power generation of rocket or induced weapons.
General axial turbines are well known by various experimental and computational studies.
But the flow characteristics of the partial admission supersonic turbine are quite different from common turbine.
So it is very difficult to estimate the performance or design the turbine well. In this situation, the investigation of flow characteristics within supersonic cascades is very important for the development of a turbo-pump system. So the flow characteristics within supersonic cascades are numerically investigated by using a commercial CFD code "FLUENT6.0".
In this study, supersonic cascade with nozzle is analyzed which is in use. And then to decrease the loss , the supersonic cascade designed by method of characteristics[1] was computed.

Numerical Method

Governing Equations and Schemes

In this study, the commercial CFD program FLUENT6.0 was used. Generally the turbine is worked in high speed, so the compressible effect should not neglect. So unsteady compressible 2-D Navier-Stokes equations was used. For turbulent effect, the k-ε turbulence model was utilized. And Euler implicit method in time, 2nd-order upwind scheme of FVM in space, Gauss-Seidel method was used.

Results and Discussions

In this study, the computations with the numerical domain, as shown in Fig. 1, have been performed and compared to the experimental results of J. J. Cho to verify the "FLUENT6.0".
The experiment had been performed by a small wind tunnel. The wind tunnel supplies nitrogen gas with a total pressure of 1000 psi and a total temperature of 293K and produces Mach number of 3.8 at exit of nozzle.

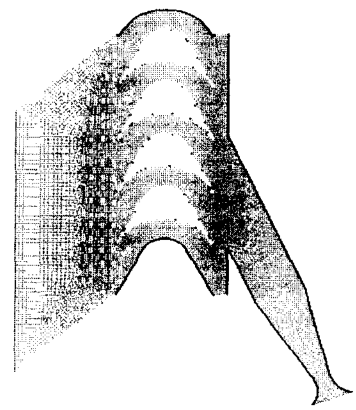
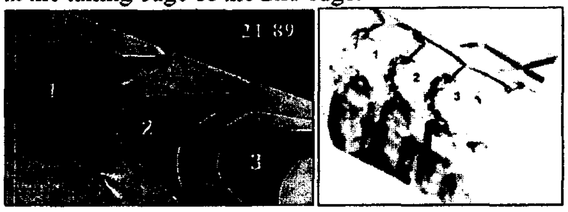


Fig. 1 Grid system of supersonic cascades and nozzle for verification

Fig. 2 (a) shows a Schlieren image of the experimental result. Detached shocks are formed at the leading edge of 1st and 2nd blade. And the shocks propagated from the nozzle are observed. The fish tail shocks are also observed at the tailing edge of the blade. The separation is formed along the pressure surface of the 1st and 2nd blade.

Fig. 2. (b) shows a computed Schlieren image which is density gradient contours represented as grayscale. The flow characteristics and shock pattern of the result are similar to those of the experimental result.
Detached shocks are observed at the leading edge of blade. And The oblique shock is propagated from the nozzle and reflected from the suction surface of the 1st and 2nd blade. The separations are observed along the pressure surface. And Fish tail shock is also observed at the tailing edge of the 2nd edge.



(a) Experimental image (b) Computed image
Fig. 2 Resuelt of Schlieren images

In order to investigate the nozzle-rotor interaction, a supersonic partial admission turbine with stationary nozzle and moving rotor are computed. The nozzle is 2-d, area ratio of Nozzle is 9.23 and throat area is 19.63mm^2. Grid system is shown in Fig. 3. The inlet boundary is set by the total pressure, static pressure, and total temperature. And the moving grid is used for the domain of cascade. Velocity of the moving grid is calculated form RPM and mean diameter of a real turbine considered in this study.
Compressed nitrogen gas considered as operating gas of the turbine. Properties of the gas and flow conditions of supersonic turbine are shown in Table 1. The mean rotor-blade speed is calculated on RPM and mean diameter of rotor.

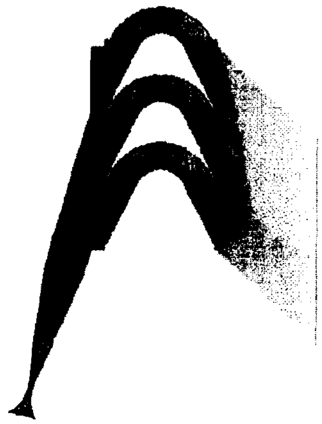


Fig. 3 Grid system of supersonic cascades and nozzle

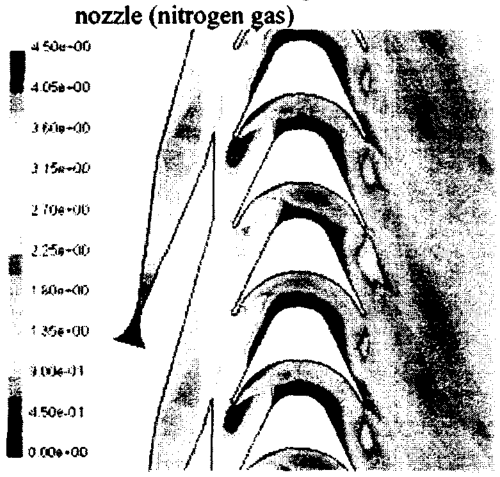
Nitrogen gas is considered as operating gas of the turbine. Properties of the gas and flow conditions of supersonic turbine are shown in Table 4.

(i) RPM : 14600

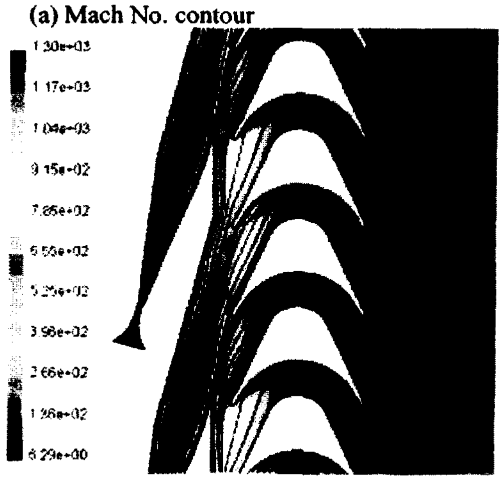
Fig. 3 (a) shows Mach number contours computed at condition of RPM=146000. Flow within nozzle is accelerated from the inlet of nozzle, reaches Mach number of 1 at the throat of nozzle, and continue to be accelerated up to the end of diverge part. Oblique shocks is observed at the end of the diverge part due to the bent wall of the nozzle. The shocks are reflected at the nozzle wall and affect the inlet flow between cascades as well as the internal flow of the nozzle. The detached shock is formed at leading edge of blade due to the bluntness of the leading edge and propagated as far as 30 percent of the chord on the pressure surface. The flow on suction surface is decelerated by the oblique shock propagated from the leading edge. At the one third of the suction surface, the detached shock and separation get together, so the separation becomes stronger. The separation region is formed from 40 percent of the chord to 75 percent of the chord. In the channel, the shocks are reflected and the flow is decelerated and accelerated according to the position. A wake propagated from the end of nozzle to the inlet of channel is observed

RPM	14600	18800	23100	27300
Rotor-blade speed (m/s)	53.6	69.0	84.5	100.2
RPM	42100	62900	84100	104800
Rotor-blade speed (m/s)	154.2	230.5	308.4	384.2
Nozzle inlet pressure (psi)	1300			
Nozzle inlet temperature (K)	300			
Outlet pressure (psi)	14.7			

Table 4 Flow conditions of supersonic cascades and nozzle (nitrogen gas)



(a) Mach No. contour



(b) Total pressure contour and path line
Fig. 3 Computational result (RPM 14600)

Fig. 3 (b) shows total pressure contours. The loss of total pressure is caused by the boundary layer on the nozzle wall, the detached shock and the oblique shocks. And channels are narrowed by the separation bubble formed at 40 percent of the chord on the suction surface.

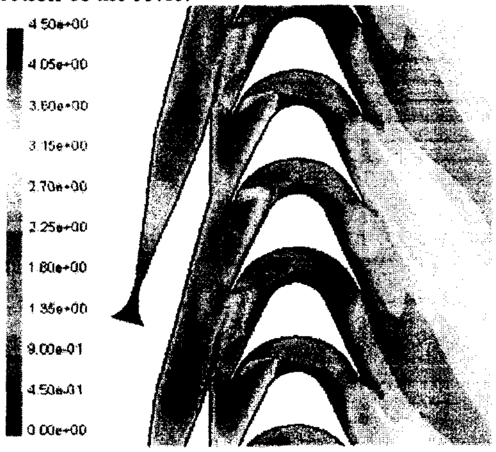
(ii) RPM : 18800, 23100, 27300, 42100

Fig. 4 show the result at the RPM is 14600. The flow characteristics including the shock pattern of the result are similar to that of RPM=14600. The more RPM is increased, the more the size of the separation is decreased. Moreover at more than 42100, subsonic region is locally occurred at the region of the downstream.

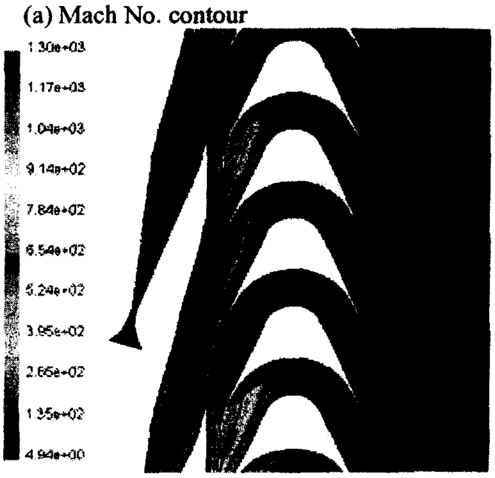
Comparing Fig. 4 (b) to the results of the low RPM, the size of separation is smaller than that of the low RPM. As RPM increase, the detached shock propagated from the leading edge is weaken.

(iii) RPM : 62900, 84100, 104800

Fig. 5 shows the result at RPM=104800. The detached shock is weakened and the separation on the suction surface is disappeared. The detached shock is almost disappeared by the decrease of the Mach Number. The downstream is also subsonic from two third of the chord. The more the RPM is increased, the more the direction of the downstream is curved in the rotation direction of the rotor.

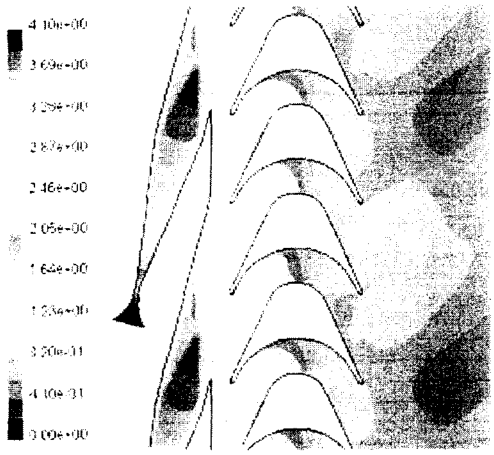


(a) Mach No. contour

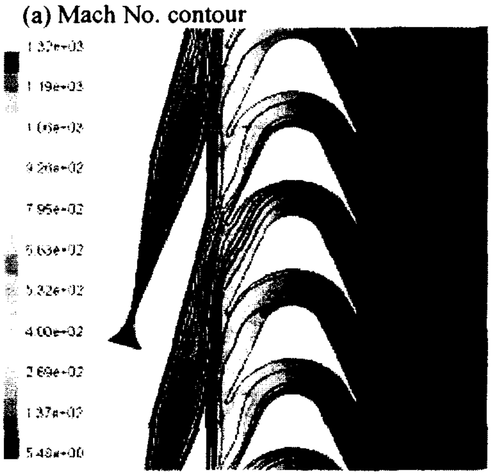


(b) Total pressure contour and path line

Fig. 4 Computational result (RPM 42100)



(a) Mach No. contour



(b) Total pressure contour and path line

Fig. 5 Computational result (RPM 104800)

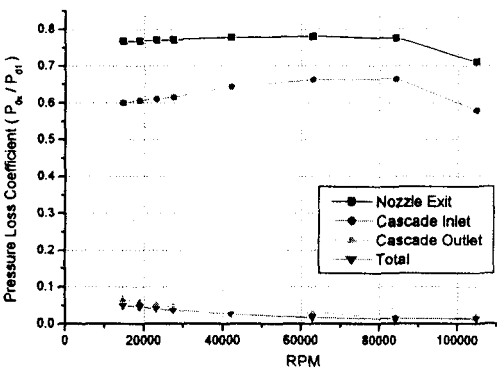


Fig. 6 Pressure loss coefficient for each part according to the RPM (nitrogen gas)

Fig. 6 shows the loss coefficient of total pressure caused by nozzle, blade, and total turbine.

Though the most of the total loss is caused at the cascade, over 20 percent of that is caused at the nozzle and about 40 percent of that is also caused by the detached shock formed the leading edge of blade and the oblique shock formed at the nozzle.

Fig. 7 shows the efficiency of the turbine according to RPM. The more RPM is increased, the more the

efficiency is increased, but for more than 84100 RPM, the more the efficiency is oppositely decreased.
This is caused by the subsonic flow formed within the channel as well as at the exit of channel and the wake attached the end of nozzle.

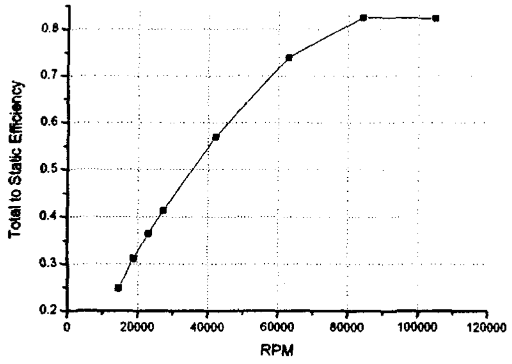


Fig. 7 Total to static efficiency according to the RPM (nitrogen gas)

Using method of characteristics, cascades are designed, and analysis is performed. Three types of cascades was designed according to the method of charateristic[1]. All type were designed with the relative inlet angle of 64.365, cascade inlet and outlet Mach no. (Mi, Mo) of 3.0. And the first type cascade, the pressure surface Mach no. (Ml) is 2.5, suction surface Mach no. (Mu) is 3.5. The second types, the pressure surface Mach no. (Ml) is 2.25, suction surface Mach no. (Mu) is 3.75. The other type, the pressure surface Mach no. (Ml) is 2.0, the suction surface Mach no. (Mu) is 4.0.
Exhaust gas is used in real turbines as operating gas.
Properties of the gas and flow conditions of supersonic turbine are shown in Table 5.

RPM	35000	45000	55000	65000
Rotor-blade speed (m/s)	128.3	164.9	201.6	238.2
RPM	100000	150000	200000	250000
Rotor-blade speed (m/s)	366.5	549.8	733.0	916.3
Nozzle inlet pressure (psi)	1000			
Nozzle inlet temperature (K)	1453			
Outlet pressure (psi)	14.7			
Gas constant (J/KgK)	388.6			
Specific heat ratio	1.18			
Viscosity (kg/ms)	2.617e-06			
Molecular weight (mol)	21.3956			

Table 5 Flow condition of supersonic cascades and nozzle (designed by MOC)

Fig 9. shows the grid system. The periodic condition is used at the front and behind the cascade. And moving grid is accepted for the cascade.

Fig. 10 (a) shows Mach number contours computed at condition of Ml : 2.5, Mu : 3.5. Flow Separation is not observed within the channel. And the flow just turns along the channel and exit the channel. But Oblique shocks is observed at the end of the diverging part and affects the inlet flow between cascades and reflects in the channel.

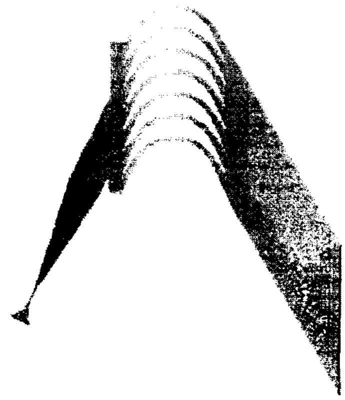
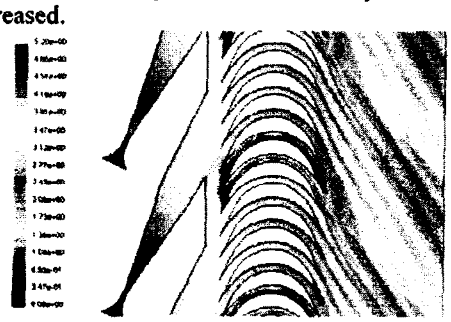
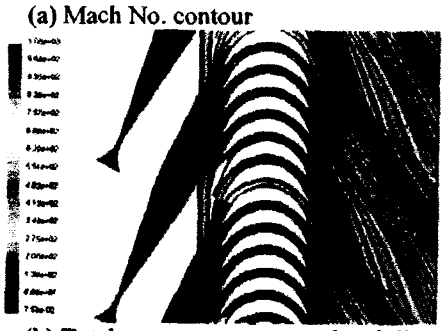


Fig. 9 Grid system of supersonic cascades and nozzle

Fig. 10 (a) shows Mach number contours computed at condition of Ml : 2.5, Mu : 3.5. Fig. 10 (b) shows the total pressure contour and path line. The loss of total pressure due to the oblique shock formed at the nozzle. And the total pressure loss is caused by the boundary layer on the nozzle and cascade wall. But the flow separation is disappeared, so the total pressure loss is decreased.

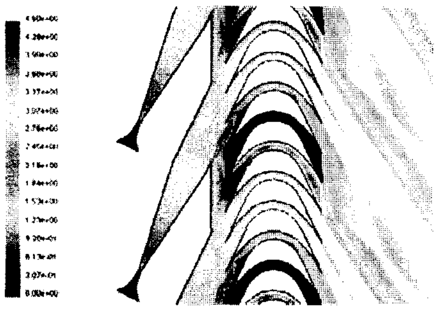


(a) Mach No. contour

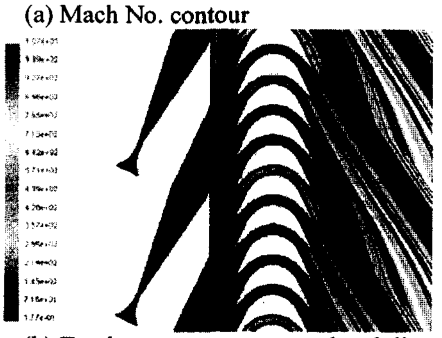


(b) Total pressure contour and path line
Fig. 10 Computational result (Ml:2.5, Mu:3.5)

Fig. 11 shows the computation result at Ml is 2.25 and Mu is 3.75. It is similar to the result at the case (i). In the channel, separation is not formed, and oblique shock is observed. And the loss of total pressure is due to the oblique shock, boundary layer.



(a) Mach No. contour



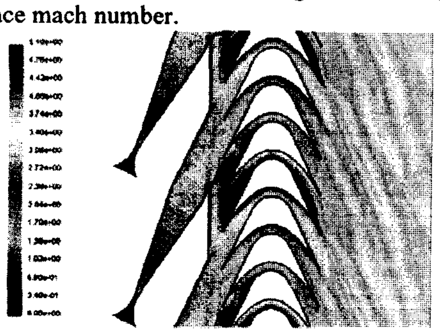
(b) Total pressure contour and path line

Fig. 11 Computational result (Ml:2.25, Mu:3.75)

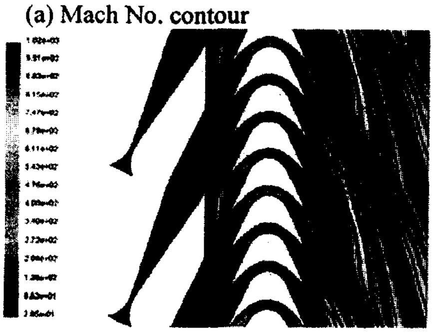
Fig. 12 (a) shows the Mach contour, the flow in the channel just turn along the cascade surface. But at suction surface 45 percent of the chord, a small separation is formed. This separation is due to the oblique shock generated in the nozzle. And this separation increases the loss of total pressure.

Fig. 12 (b) shows the total pressure contour. The loss of total pressure is due to the oblique shock and boundary layer and separation especially in this case.

In this chaper, the cascade is designed by method of characteristics. Three case is designed according to the surface mach number.



(a) Mach No. contour



(b) Total pressure contour and path line

Fig. 12 Computational result (Ml:2.0, Mu:4.0)

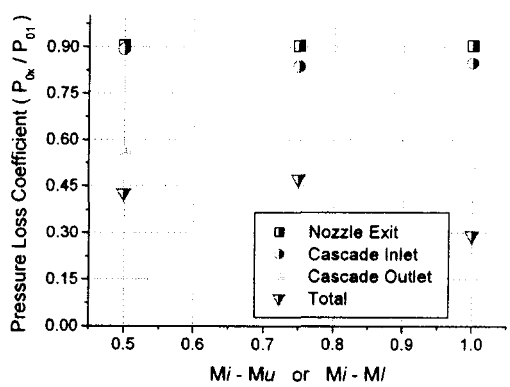


Fig. 13 Pressure loss coefficient for each part

Fig. 13 shows the loss coefficient of total pressure caused by nozzle, blade, and total turbine. Though the most of the total loss is caused at the cascade, 10 percent of that is caused at the nozzle and about 15 percent of that is occurred between the nozzle and cascade. The total loss is about 40 percent. The difference of total loss is due to the sum of boundary effect. The pitch is larger, the boundary layer is thicker, so the loss is increased. But the pitch is smaller, the number of pitch is increase, so the loss is increased too. So the optimum surface Mach no. is pressure surface Mach no. is 2.25 and suction surface Mach no. is 3.75.

The computations for three different edges, edges with zero-thickness, linear edges with 4mm thickness, and circular edge with 4mm radius, were performed to analyze the effect of the edge shape on the loss of the turbine. The boundary conditions are identical except the shape of edge. Fig. 14 shows the grid system of the computational domain.

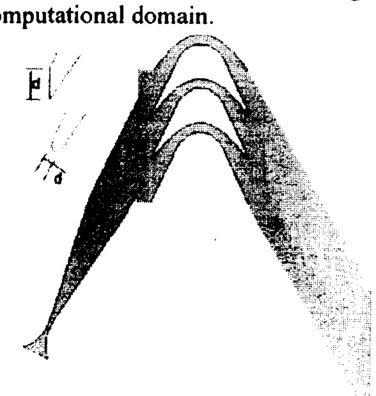
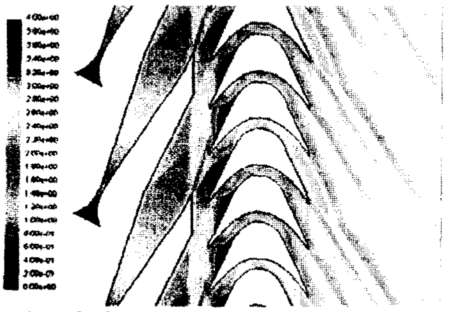
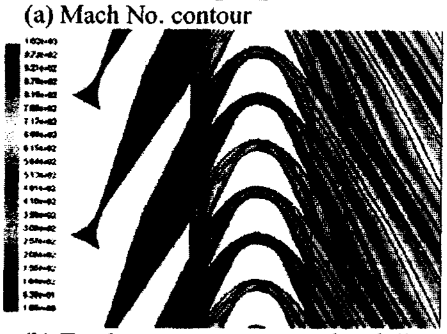


Fig. 14 Grid system of different edges

Fig. 15 (a) shows Mach number contours, The separation The flow within cascades is turned along the cascades without separation. The flow is decreased by the oblique shock propagated from the connection of a diverging part and a linear part. The shock is reflected from a wall of the blade. And oblique shocks are occurred at the leading edge due to zero-thickness of the edge and passed into the channel.



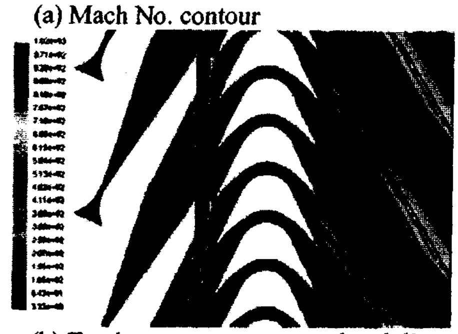
(a) Mach No. contour



(b) Total pressure contour and path line
Fig. 15 Computational result (straight edge)

And the shocks cause a large separation formed at 10% of the chord on the suction surface which is reattached at 40% of the chord on the surface.
Fig. 15 (b) shows total pressure contours. The losses caused by the oblique shocks propagated from the nozzle and the oblique shocks formed at the leading edge are observed. And the losses caused by the separation and the boundary layer are also observed.
Fig. 16 (a) shows Mach number contours of the result. The flow characteristic of the result is similar to that of the result computed with the linear edges.

(a) Mach No. contour



(b) Total pressure contour and path line
Fig. 16 Computational result (round edge)

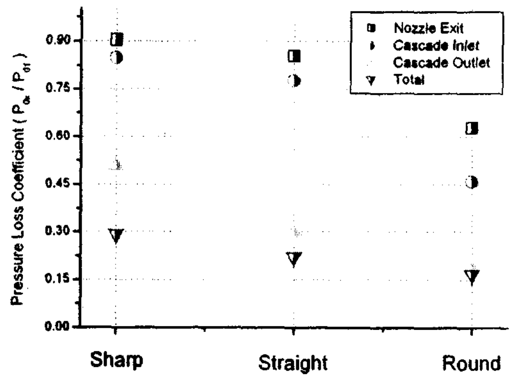


Fig. 17 Pressure loss coefficient for each part according to the edge shape

But, instead of the oblique shock, detached shocks are formed at the leading edge of blades by the bluntness of the blade. And the shocks cause a large separation formed at 10% of the chord on the suction surface which is reattached at 35% of the chord on the surface. Fig. 16 (b) shows total pressure contours. The losses caused by the shocks, the separation and the boundary layer are observed. Fig. 17 show the loss coefficient for each part of nozzle and cascade according to the shape of the edges The loss coefficient of the circular edges is greater than that of other shape of edges because the detached shock cause a high loss. The loss coefficient of the linear edges is greater thant that of the circular edges because the loss caused by the detached shock is greater than by the oblique shock

Conclusion

In order to investigate the nozzle–rotor interactions and the effect of partial admission, the flows in supersonic turbine cascades with a nozzle are computed using the commercial program FLUENT 6.0.
Many computations of partial admission supersonic turbines provide the shock structures and flow patterns in the nozzle and rotor.
It is found that the computation of the cascade and nozzle together can maintain the inlet condition though the expansion wave or shock affect the periodic condition.
In the supersonic turbine, the shock structure and shock magnitude affect the flow characteristics very much. Method of characteristics is useful for decreasing the loss. The performance of supersonic turbine cascade was considerably affected by the oblique shock or detached shock generated at the leading edge. It is necessary to decrease the loss, the shock generated at the leading edge or nozzle should be decreased or removed.

References

1) Louis. J. Goldman & Vincent J. Scullin, "ANALYTICAL INVESTIGATION OF SUPERSONIC TURBOMACHINERY BLADE I - COMPUTER PROGRAM FOR BLADING DESIGN", NASA TN D-4421.
2) R. F. Cuffel. H. Black and P. F. Master, "Transonic Flow Field in a Supersonic Nozzle with Small Throat Radius of Curvature", AIAA, J., 17, 1364, 1969.
3) Kiock, A., Lehthaus, F., Baines, N.C. and Sieverding, C.H., "The Transonic Flow through a Plane Turbine Cascade as Measured in Four European Wind Tunnels", Trans. of the ASME, J. of Eng. for Gas Turbines and Power. Vol. 108, pp. 277-85, 1986.
4) Bong-Gun Shin, Young-Sik Seong, Soo-in Jeong,Kui-Soon Kim, Eun-Seok Lee "A Study on The Characteristics of The Inlet Boundary Condition of a Supersonic Turbine Cascade", 2003 KSPE Fall Conference, pp. 99-103, 2003.