A Digital Simulation and its Experimental Investigation for the Response of Gas–turbine Engines to Intake Flow Distortion

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ABSTRACT

This paper presents a digital technique which is used to study the response of gas–turbine engines to intake flow distortion. It will also calculate the performance characteristic of the axial flow compressor and the variation of the stall margin due to the circumferential distortion produced by the 180° – extent screen. The main purpose of this paper is to develop a mathematical model of the engine compressor performance with intake flow distortion. This will allow both the effects of intake flow distortion on the aerodynamic stability of gas–turbine engines and the variation of engine compressor performance to be predicted by the use of the digital technique described in this paper. The tests have been conducted on a 9–stage with first stage being transonic, single spool engine with intake flow distortion on the test–bed. Both the test results and the predicted results are also presented. The predicted results based on both the measured inlet distortion map and the uniform inlet compressor performance have been compared with the measured results. It can be seen from the comparison that the agreement is good.

I. Introduction

A lot of work has been done,in which the effects of intake flow distortion on engine stability have been investigated and many predicting models have been developed. The parallel compressor model is one of them.This model is used to adequately predict the effects of circumferential steady–state distortion of the total pressure and the temperature on the compressor stall margin.As inputs,the inlet distortion map,the uniform compressor characteristics and the compressor stall line are required in calculation. The model has been used by many investigators to predict the loss in the compressor stall margin of both the single–spool and the dual–spool distorted engine,but there exists a basic lack of agreement with a large amount of experimental data on distorted compressor performance characteristic and surge line loss. This is due to ignorance of some factors (such as the dynamic stall deldy effect).

This paper considers the circumference of the compressor with distorted inlet flow to be divided into multiple segments,and the effect of the dynamic stall delay and the inlet flow redistribution to be induced by distortion on the base of parallel compressor model.This paper emphasizes the study on the stability of the engine compressor with distorted inlet flow (not the individual compressor component).Hence,we must solve how to force the engine into surge.Since the method to reduce the exit area of the nozzle to force the engine into surge will make the turbine inlet temperature too high,the method to inject water into mixing zone in combustor in the paper is used. When this is done,the turbine inlet temperature will drop with the movement of engine operating point towards surge line along the corrected speed line.In the calculation,overall compressor performance is used,and the effective angle of attack is used as the stability criterion.An single–spool turbojet engine with 9–statge compressor,the first stage being transonic was calculated as an example.

To test the reliability of calculation and estimate the accuracy,the line of the constant corrected speed of engine compressor and its surge point, which have been measured from the ground test bed, are also presented in the paper. Experiments of engine have been conducted with the distorted inlet flowfield generated by a 180° –extant,36 mesh screen. The predicted results based on both the measured inlet distorted map and the "clean" inlet compressor performance have been compared with the test results. It can be seen from the comparisons that both agrees well with each other.

II. Multiple Segment Parallel Compressor Model with the Dynamic Stall Delay Effect

The current model expands the basic parallel compressor theory by using multiple parallel segments to provide a detailed definition of the circumferential flow field. These segments are pseudo–stream tubes passing through the compressor from inlet to exit. The flow rate in each segment is determined from its boundary conditions (inlet total pressure and total tomperature and exit static pressure) and the compressor performance within each segment is determined in a manner quite similar to classic parallel compressor.

Classic parallel compressor model takes account of the occurrence of instability for the overall distorted compressor when the pressure ratio of any segment of compressor reaches stall pressure ratio on "clean" inlet flow. In fact, because the angle of attack of compressor rotor blade airfoil is subjected to unsteady variations along the circumference, there many be an angle of attack larger than that of steady state stall when the rotor operates near the stall line.The occurrence of instability for the overall distorted compressor is considered by the current model when the effective angle of attack of any srgment reaches the steady seall angle of attack.

Ⅲ. Distortion Induced Inlet Flow Redistribution

The compressor operating with uniform inlet flow has axial flow at inlet. When there exsit circumferential total pressure distortion at compressor inlet, the flow redistribution will take place upstream of the compressor , hence it causes a variation of airflow angle at inlet. Without inlet guide vane, the incidence on the first rotor blade will also vary. Because the incidence is related to the flow rate, the predicting model should consider the variation in the incidence along the circumference.

The procedure for calculating the upstream flow redistribution is based on the use of a distribution of sources and sinks at the compressor inlet plane to represent the effect of the compressor on the upstream flow [1] .The formula for calculating the inlet circumferential velocity of each segment is deduced and give as follows.

$$C_{\theta L} = \frac{\sum_{n=1}^{L-1} \frac{Q_n}{2h} \frac{sin\varphi}{1-cos\varphi} + \sum_{n=L+1}^{m} \frac{Q_n}{2h} \frac{sin\varphi}{1-cos\varphi}}{\sqrt{1-\overline{M}_1^2}} \qquad (L=1,\cdots,m)$$

where: $\varphi = 2\pi b(L-n)/h$

h = period, $h = 2\pi r$

r = inlet average radius of the compressor

M_1 = intake flow average Mach number of the compressor

b = the circumferential width of each segment $b = h/m$

m = the number of all segments

Q_n = the strength of source (or sink)

Ⅳ. The Dynamic Stall Delay Effect

The angle of attack of a compressor rotor blade is related to circumferential location because of the circumferential distortion. When the rotor enters into the region of low inlet total pressure, the angle of attack is decreased. When the rotor goes out of the low pressure region, it is increased. Hence, the angle of attack is subjected to a kind of sudden periodic variations, which will cause the lag in lift response. If the lag in lift response appears, the rotor may be subjected to a kind of incidence variations beyond the steady state stall limit, but stall does still not occur. It may bee hypothesized that this kind of incidence variation is only of a very short duration. This phenomenon is verified by the test results of the current engine at 90 percent of the designed speed (Fig. 1). It is clearly seen from Fig. 1 that pressure ratio of the segment on distorted inlet flow is beyond steady state stall value.

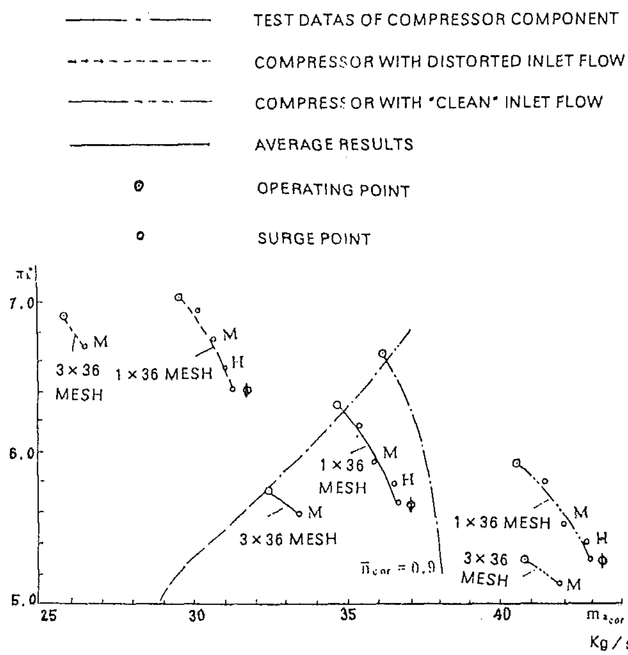

Fig. 1 compressor performance map with distorted segment and "clean" segment

In dynamic process, airflow incidence continuously changes. The variations in angle of attack not only require the adjustment of flow about an airfoil, but also make the change in circulation. Because of the difference of circulation , it causes the trailing edge vortex shed in order to adapt the operation under new incidence. It is known that the pressure ratio of the compressor is related to lift on the airfoils of blades which the compressor is composed of.

In the case of unsteady flow, the lift on the airfoil is not only depend on the instantaneous angle of attack, but also on the following two factores: (1) the inertia or acceleration of the mass of air in proximity of the airfoil and; (2) the shedding of the trailing edge vortex which acts as a dissipative force [2] . Hence, when the instantaneous angle of attack increases, the lift on the airfoil can not instantly response because airflow adjustment, shed vorticity and the boundary layer development require a finite amount of time, then the pressure ratio of the compressor is increased in a time lag. From above, it is seen that the compressor may be operated at the state where the incidence is larger than the steady state stall value but stall does not occur when the incidence continuously changes. During unsteady flow stall will not occur untill the effective angle of attack induced by the lag in lift response is equal to the steady state stall value.

The effect angle of attack is computed as follows:

$$\alpha_{eff}(\theta) - \alpha_0 = \sum_{n=1}^{\infty} [f(nk)\, a_n\, cos(n\theta - \varphi(nk)) + f(nk)\, b_n\, sin(n\theta - \varphi(nk))]$$

where: $f(nk) = \frac{1}{\sqrt{[1+(2nk\tau_1)^2][1+(2nk\tau_2)^2]}}$

$\varphi(nk) = tg^{-1}(2nk\tau_1) + tg^{-1}(2nk\tau_2)$

n = the hormonic number

a_n, b_n = Fourier Coefficients

τ_1, τ_2 = non–dimensional time constants

k = reduced frequency

α_0 = average angle of attack

α_{eff} = effective angle of attack

V. Test on the ground test–bed

The experimental arrangement of the test engine is showned in Fig. 2,which is composed of a 9–stage,axial flow compressor and a 2–stage turbine.An inlet pipe was installed in front of the engine face to measure the flow rate. The airflow enters the compressor inlet through the pipe, in which a rotatable distortion screen is installed to produce intake flow diatortion. The different level of intake flow distortion can be obtained by changing the screen mesh.The screen was located at the place about half the pipe diameter in front of the compressor inlet. Its circumferential coverage is 180 ° .There is a removable cone at the nozzle exit. When it moves forward into the nozzle,the nozzle exit area can be reduced. The maximum blocking area is about 30 percent of the rated nozzle exit area.

In the test, the corrected speed is kept constant. We use two methods to force the engine into surge: (1) gradully reducing the nozzle exit area by moving the cone toward the inside of nozzle; (2) injecting water into the combustor.

Fig. 3 shows the measured circumferential total pressure profile with 180 ° –extent,36 mesh screen when the speed is 87 percent of the designed one. Fig.5 shows the measured engine compressor characteristics for the same engine with distortion and "clean" inlet.

VI. Data Analysisand and Comparisons

Comparison of two menthods to force the engine into surge–The engine speed is 87 percent of the designed one. The front of the engine face has a 180 ° –extent,one deck of 36 mesh screen. Under these conditions,we obtained pre-

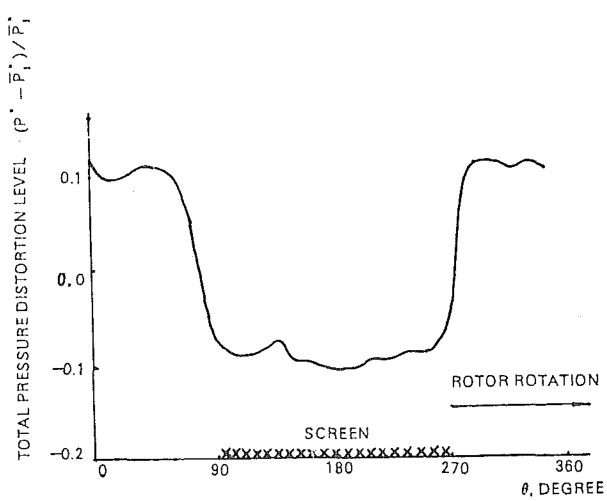

Figure 3. The circumferential total pressure profile

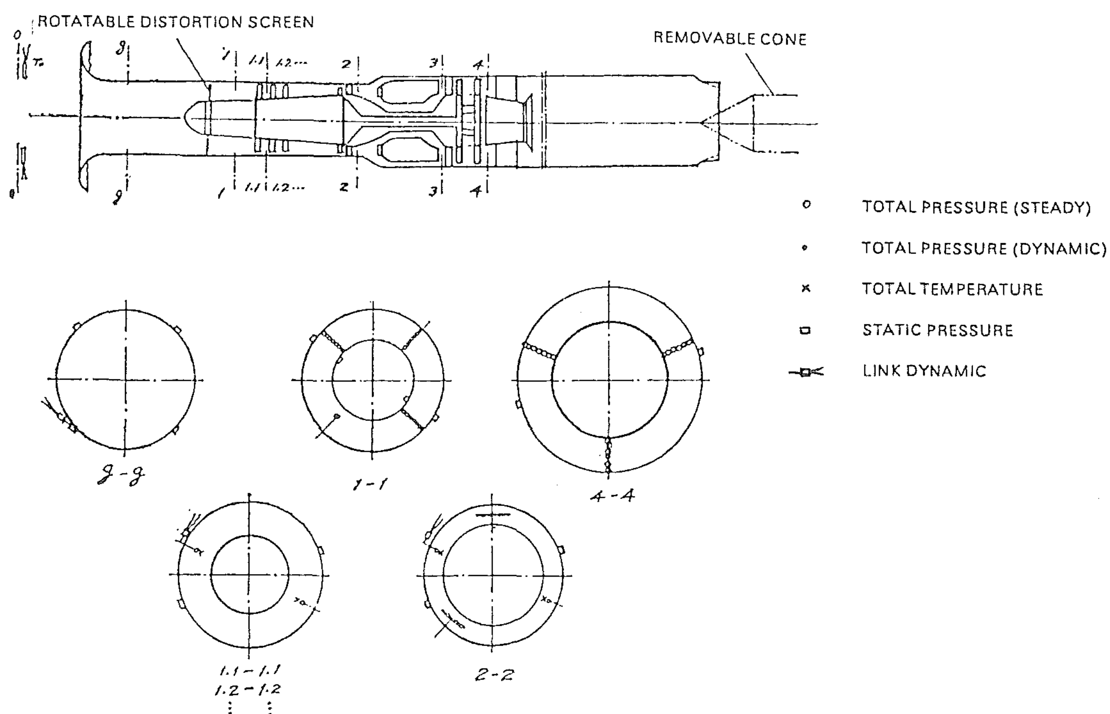

Figure 2. Test engine and the arrangement scheme

dicted results by use of above metioned two methods to force the engine into surge.They are given in Table 1,and 2.

G(Kg / s)	0	1	2	3	4
π_k^*	5.444	5.475	5.512	5.540	5.557
$m_{a_{cor}}$ (Kg/s)	32.41	32.04	31.76	31.23	30.99
T_3^* (K)	875.5	823.8	782.2	737.9	701.9

Table 1. The effect of injecting water into the mixing zone in combustor to the engine operating points

Ae(cm^2)	2034.8	1808.7	1520.5	1368.5
π_k^*	5.398	5.438	5.509	5.553
$m_{a_{cor}}$	32.80	32.42	31.60	30.72
T_3^* (K)	845.05	874.34	937.99	1002.6

Table 2. The effect of changing the nozzle exit area to the engine operating points

where: G = the quantity of injecting water

π_k^* = the pressure ratio of the compressor

$m_{a_{cor}}$ = the corrected flow rate of the engine

T_3^* = the temperature at turbine inlet

Ae = the area at exit of engine nozzle

As seen from the data of Table 1,and 2, by the use of the method to changing the nozzle exit area to force the engine into surge, the temperature at turbine inlet increases when the engine operating point moves towards stall line along the constant corrected speed line, but by the use of the method to inject water into combustor,the temperature decreases. This is because the water vapour formed by injecting water takes up some amount of space in the main airflow passage, which correspondingly decreases the area of the turbine guide vane. The pressure ratio increases and the compressor work per unit flow rate increases.But because injecting water makes the engine flow rate increase and turbine work per unit flow rate for turbine decreases,T_3^* decreases. While without injecting water, T_3^* increases.

Distorted Inlet Flow of the Compressor– When the circumferential total pressure distortion exists at the compressor inlet, airflow does not flow axially, but in resultant way (axially and circumferentially). Considering this phenomenon, we calculated the distribution of the rotor incidence along circumference(Fig. 4).

As seen from Fig. 4, because the function of compressor pump makes the streamline curve upstream of the compressor on non–uniform, the rotor blade incidence suddenly decreases when the rotor enters into the region of distortion, and it suddenly increases when the rotor comes out.

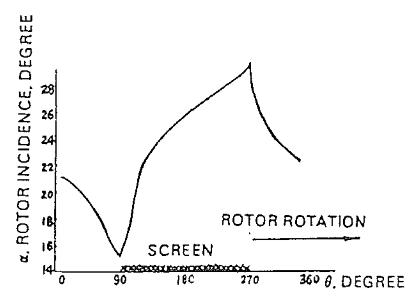

Figure 4. The circumferential distribution of the rotor incidence at the compressor inlet

Comparison of predicted data with test data for surge point of inlet distorted engine– The engine used for computation has the same operating environments as test engine, that is, under the standstill state on the ground,distorted inlet flow field induced by 180 ° –extent,36 mesh screen. The results are shown in Fig. 5.

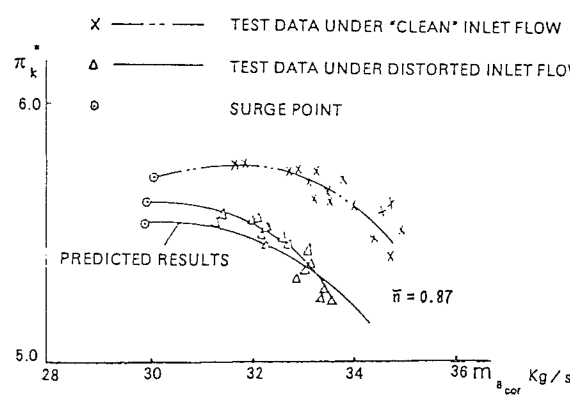

Figure 5. The effect of distorted inlet flow to the compressor characteristic

	distorted level	G(Kg / s)	π_k^*	$m_{a_{cor}}$ (Kg / s)
predicted value	12.2%	5.2	5.527	29.96
test value	12.2%		5.615	29.89

Table 3. The surge point parameters of the distorted engine ($\bar{n}$ = 0.87)

The predicted value and test value of the surge point are shown in Table 3. The predicted value is the result considering dynamic stall delay effect and the inlet flow redistribution effect. It can be seen from Table 3 that both results is agreeable to each other. If we do not consider dynamic stall delay effect (that is, we use the maximum instantaneous angle of attack as stability criterion. When that of any segment reachs steady stall value $\alpha_{sss} = 0.44904$, $\pi_k^* = 5.437$, and $m_{a_{cor}} = 32.42$), stall will occur in advance.

References

1. Mazzawy,R.S.,Multiple Segment Parallel Compressor Model for Circumferential Flow Distortion, AGARD, CP–177.
2. Melick,H.C.,Analysis of Inlet Flow Distortion and Turbulence Effects on Compressor Stability, NASA,CR–114577, March 1973.