

Calculus of Turbojet Performances for Ideal Case

S. Bennoud, S. Hocine, H. Slme

Abstract—Developments in turbine cooling technology play an important role in increasing the thermal efficiency and the power output of recent gas turbines, in particular the turbojets.

Advanced turbojets operate at high temperatures to improve thermal efficiency and power output. These temperatures are far above the permissible metal temperatures. Therefore, there is a critical need to cool the blades in order to give them a maximum life period for safe operation.

The focused objective of this work is to calculate the turbojet performances, as well as the calculation of turbine blades cooling.

The developed application able the calculation of turbojet performances to different altitudes in order to find a point of optimal use making possible to maintain the turbine blades at an acceptable maximum temperature and to limit the local variations in temperatures in order to guarantee their integrity during all the lifespan of the engine.

Keywords—Brayton cycle, Turbine Blades Cooling, Turbojet Cycle, turbojet performances.

I. INTRODUCTION

GAS turbines play a fundamental role in the today's industrialized applications. Gas turbines have undergone a considerable evolution in the past few years in order to satisfy the power increase, the power output and thermal efficiency of gas turbines engine [1], [2].

The manufacturers are strongly interested by the development of the performances of the aircraft gas turbine in particular the double flow turbojets that have been widely used in all the important achievements in the aviation industry including those concerning the recent design in civilian and military aircrafts [3], [4].

One method of increasing both the engine performances (thrust, power, and engine performances) is to increase the temperature of the gas entering the turbine.

In the recent turbojets, the turbine inlet temperature can be as high as 1800°C; however, this temperature exceeds the melting temperature of the metal airfoils, the temperatures reached at the end of the combustion cannot be supported by used materials (order of use: 1100°C). Therefore, it is imperative that the turbine blades must be cooled, so they can withstand these extreme temperatures [5], [6].

The augmentation of the temperature of the inlet gas of turbine will always produce an augmentation of thrust, thus the improving of overall performances of the turbojet. This augmentation will benefit the performance, but this effect has drawbacks because of the effects produced on the turbine blades, such as the thermal fatigue, the corrosion and also the

creep caused by the centrifugal force caused by the rotation of elements.

Under these conditions, the turbine blades must be cooled to maintain their integrity and achieve their maximum life period [7], [8].

Various cooling concepts are used in different combinations to adequately cool the turbine blades. The cooling by the air was used (the air comes from the top floor of high-pressure compressor (cooling air around 650°C), it is fed into the turbine blades by their roots, the temperature of the blades can be lowered to approximately 1000°C), this system is based on the use of the principle of thermal convection [2], [9]-[12], [18].

Using the heat transfer equations and the thermodynamics turbojet cycle formulas [13]-[18], cooling of turbine blades is studied.

The principal objective of this work is to calculate the turbojet performances, as well as the calculation of turbine blades cooling.

The developed application makes the calculation of the turbine blades cooling to different altitudes in order to find a point of optimal use making it possible to maintain the turbine blades at an acceptable maximum temperature and to limit the local variations in temperatures in order to guarantee their integrity during all the lifespan of the engine.

II THEORETICAL APPROACHES

The improvement of the turbojet performances led to the design of components equipped with better outputs able to increase the temperatures of exit of High Pressure Compressor (HPC) and temperature of exit of Combustion Chamber (CC).

The cooling system must meet design criteria that are dictated by various constraints which are: creep, thermal fatigue and oxidation and corrosion.

The performances of aircraft engine are described through two important parameters: the Specific Thrust (ST) and the Thrust Specific Fuel Consumption (TSFC).

It is important to dimensioning a system by finding the best compromise between a maximum specific push and a minimal specific consumption. For that, two variables characteristic of the engine must give: The Turbine Inlet Temperature (TIT) which represents the temperature of combustion gases at the entry of the High Pressure Turbine. And the Compressor Pressures Ratio (CPR).

The improvement of engine performances thus passes by the increase in the temperature at the turbine entry and the compression ratio. The principal objective is to find a point of optimal use making it possible to maintain the turbine blades at an acceptable maximum temperature and to limit the local

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variations in temperatures in order to guarantee their integrity during all the lifespan of the engine.

In this context, the turbojet is heat engine that converts thermal energy into useful work. This work is based on the production of compressed air and combustion products which are then accelerated to provide reaction propulsion. The thermodynamic analysis of this problem can be understood using the Brayton cycle.

The Brayton cycle is the fundamental constant pressure gas heating cycle used by turbojets (Brayton cycle for ideal turbojets is presented in Fig. 1). It consists of: 0-2: isentropic compression. 2-3: constant pressure heating through combustion chamber. 3-4: isentropic expansion through turbine and nozzle. 4-0: constant pressure cooling (absent in open cycle gas turbines).

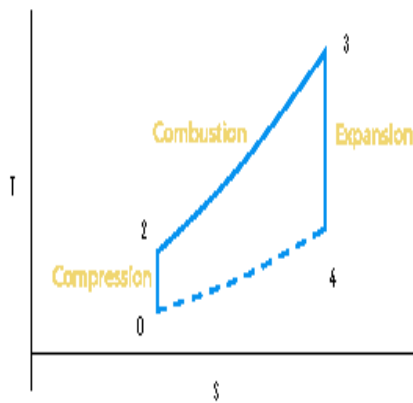


Fig. 1 The T-S diagram of ideal Brayton cycle for turbojets

The stations at which velocities and thermodynamic states were computed are presented in Fig. 2.

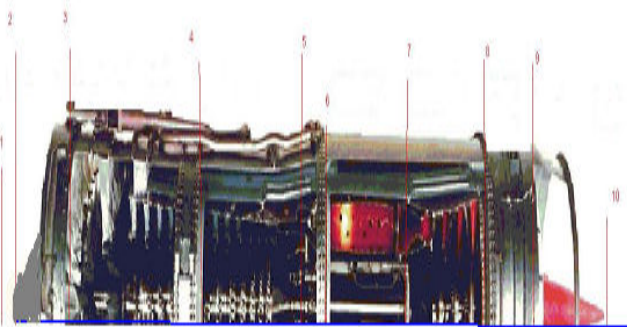


Fig. 2 Stations of turbojet cycle calculation.

1 to 2: Air inlet or diffuser.

2 to 3: Fan, which gives at air a compression beforehand and then divides the airflow into two flows, the primary airflow through the reactor and the secondary airflow through the annular gap between the hull and the stators of the low pressure turbine.

3 to 4: Low Pressure Compressor (LPC), in which the primary flow undergoes a first adiabatic compression.

4 to 5: High Pressure Compressor (HPC), in which the compressed air being partially penetrates and reaches the pressure and temperature of ignition.

5 to 6: Diffuser upstream to the Combustion Chamber.

6 to 7: Combustion Chamber, in which the flow acquires a quantity of energy delivered by the combustion, the temperature increases dramatically as the pressure is almost constant.

7 to 8: High Pressure Turbine (HPT), in which gases leaving the combustion chamber with raised pressure and temperature, undergo a first stage of adiabatic relaxation, which creates a fall of high pressure, recovered in a kinetic energy.

8 to 9: Low Pressure Turbine (LPT), who is the continuation of the relaxation of the gas to escape.

9 to 10: Nozzle, in which the pressure energy is converted into kinetic energy. The exhaust gases continue to relax until the atmospheric pressure.

III. METHODS

The principal objective of this work is to calculate the performances of a turbojet by calculus and optimizing of the cooling in the High Pressure Turbine blades.

The turbojet was divided on ten different stations (see Fig. 2). In each of these ten stations, aerodynamic and thermodynamic characteristics of the airflow of turbojet were determined (using the equations of the standard atmosphere and the thermodynamics), for example the static temperature, the total temperature, the static pressure, the total pressure, the velocity of flow and the Mach number,

Fig. 3 presents the flowchart that including the principal steps used to calculate different characteristics in order to obtain turbojet performances.

Obtained values such as: static and total temperature, static pressure at the exit of the 'high pressure compressor' and the static and total temperature, static pressure at the exit of the combustion chamber, can be used later in the calculation of cooling.

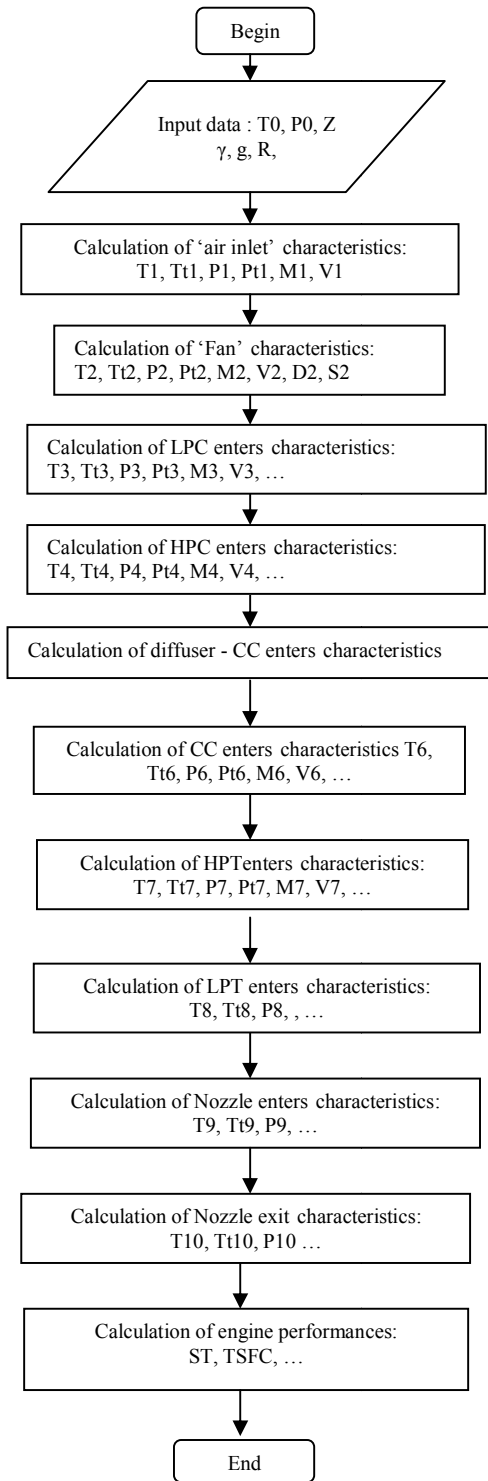


Fig. 3 Flowchart of developed application

IV. RESULTS

In this section, case study of (JT8-D15) turbojet is presented.

Stating by the introduction of flight data that presented in Table I, attached values to various turbojet sections can be detailed (see Table II).

TABLE I
 INPUT FLIGHT DATA

Z (m)	Ambient pressure (Pa)	Ambient temperature (K)	R gas constant (J.Kg ⁻¹ .K ⁻¹)	γ
5000	101325	300	287.053	1.4

The obtained results presented in Table II reveal the characteristics and all the parameters of air through the various stations of the turbojet.

TABLE II
 PERFORMANCES AT DIFFERENT STATIONS (Z=5000m)

	T [K]	Tt [K]	P [KPa]	Pt [KPa]	M	V [m/s]
1	274.0	279.3	62.90	67.34	0.31	104.05
2	273.6	279.3	61.72	66.33	0.32	106.98
3	332.0	337.9	117.27	126.64	0.35	130.49
4	458.9	438.5	245.71	280.57	0.27	202.40
5	680.9	691.9	1074.7	1128.6	0.48	148.73
6	685.8	691.9	1078.0	1111.7	0.21	110.25
7	914.0	925.8	1121.2	1180.8	0.28	175.95
8	879.5	921.8	536.27	590.44	0.32	203.14
9	776.2	910.0	66.19	73.80	0.35	203.14
10	723.2	910.0	46.40	73.80	0.86	480.72

After several executions carried out by changing the setting of altitude, evolution of the parameters of air through the turbojet stations are presented in the following section.

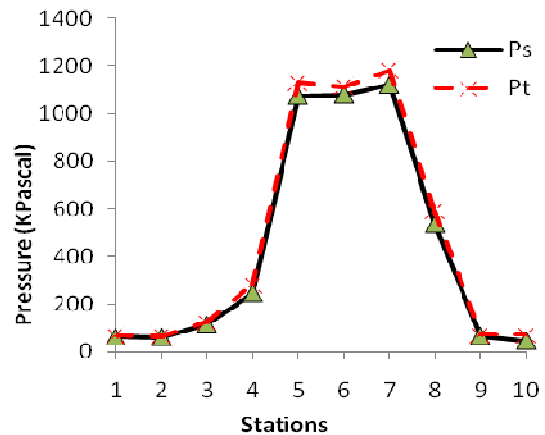


Fig. 4 Pressure evolution at Z=5000 m (Static pressure and total pressure)

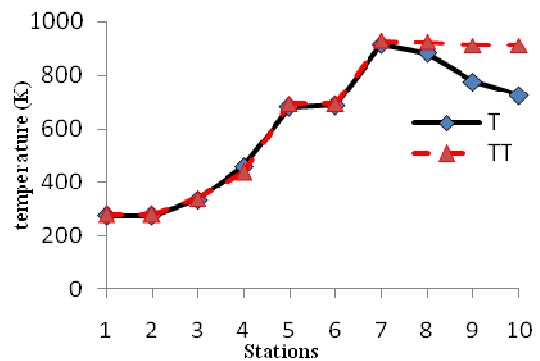


Fig. 5 Temperature evolution at Z=5000 m (Static temperature and total temperature)

The static temperature (and/ or total temperature) increases from the air inlet to the exit of the combustion chamber due to compression in compressors, and combustion in the combustion chamber, and thereafter it will be reduced due to the easing of turbines (Fig. 5). The same variation of the static pressure (and/ or total pressure) in the turbojet stations can be observed (Fig. 4).

In Figs. 6 and 7, temperature and pressure values in the fan enter at various altitudes is presented.

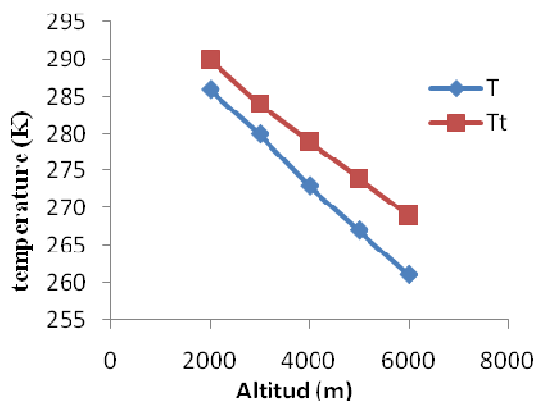


Fig. 6 Temperature at the station 2 (fan) at various altitudes

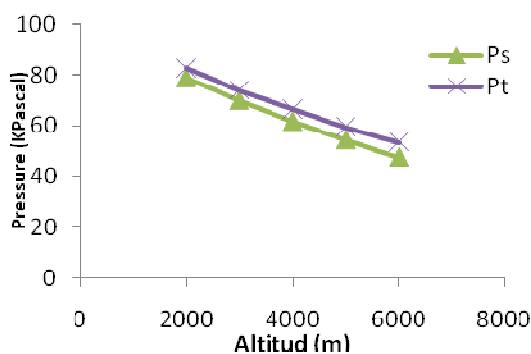


Fig. 7 Pressure at the station 2 (fan) at various altitudes

The comparison between the static temperature at the station 2 (fan), enabling to say that temperatures decrease if altitude is increased, and the same observation is true for the total temperature, because the calculations are done in a recursive manner, this which allows to say that the results obtained are very logical.

The static pressure (and/ or total pressure) decreased depending on the altitude, which confirms the principles of the standard atmosphere.

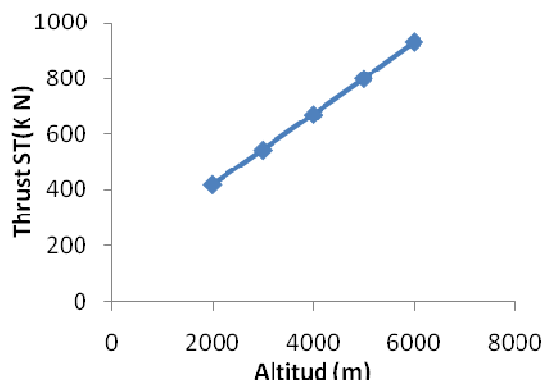


Fig. 8 Variation of ST according to Z

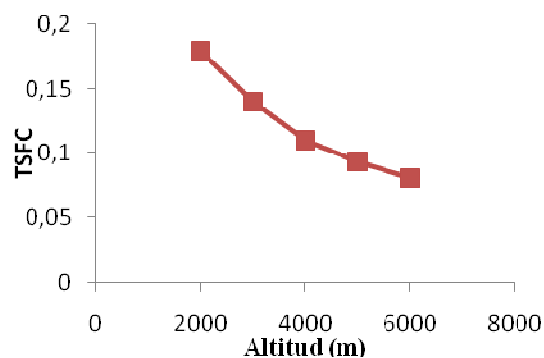


Fig. 9 Variation of TSFC according to Z

The total thrust ST increases and the specific fuel consumption TSFC decreases whenever the altitude increases (Figs. 8 and 9). This phenomenon is explained by the reduced air resistance as a function of altitude (i.e. that the increase in altitude causes a decrease in air resistance, implies that the consumption rate is decreased).

V. CONCLUSION

The high temperatures at the exit of the combustion chamber are necessary to meet design requirement (increase of turbojet performances), but these temperatures have drawbacks due to the effects produced on the turbine blades, such as the thermal fatigue, the corrosion, and also the creep.

These problems influence the thermal and mechanical characteristics of material, also the aerodynamic and thermodynamic parameters.

So, the turbine blades must be cooled to keep their integrity and to reach one lifespan maximum.

For evaluating the performance of various turbojets, it is appropriate to considering at first the thermodynamic cycle at selected flight conditions. For that double flow turbojets are selected for thermodynamic studies yielding numbers for specific fuel consumption and specific thrust.

A computer program has been developed to study ideal jet engines thermodynamic cycle.

This code works in an interactive option by input of relevant data and giving various results concerning aerodynamic and thermodynamic parameters.

The code run enables to get the following results: the total and static pressures, also the static and total temperatures have decreased, the thrust has increased and the specific consumption of fuel has decreased whenever the altitude increases.

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