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Abstract

The engine manufacturers adopt new measures in order to further improve the characteristics of a turbine engine. They pose new challenges to reduce a fuel consumption and an emission of pollution to the environment (including noise), but also keeping the highest level of reliability. Based on those considerations, current research in propulsion is conducted.

Modern turbines are characterised by high inlet temperature. This has implications for engine efficiency, which is expressed with a change of mass, cross-section and fuel consumption. In this article, main trends in the development of turbine engines are presented. This analysis was carried out on the basis of Rolls-Royce engine data.

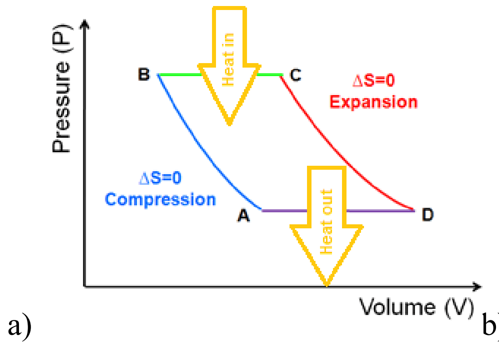
The article presents literature review concerning the analytical methods of high-pressure turbines preliminary design. The aerodynamic design process is highly iterative, multidisciplinary and complex. Due to this, modern gas turbines need sophisticated tools in terms of aerodynamics, mechanical properties and materials.

The article depicts simplified model of real turbine engine. As showed in the article, this model gives only a 10% error level in engine thrust value. The calculations may be used for preliminary engine analyses.

Keywords: *preliminary design, turbomachinery, gas turbines, high-pressure turbine, HPT*

1. Introduction

Gas turbines are consisted of three sections mounted on the same shaft: the compressor, the combustion chamber (or combustor) and the turbine. The thermodynamic process used in gas turbines is the Brayton cycle (Fig. 1a). Compressed air is mixed with fuel, and then burned under constant pressure conditions. The resulting hot gas is allowed to expand through a turbine and to perform work, also propelling the compressor. Operating principle of gas turbine is given in Fig. 1b).



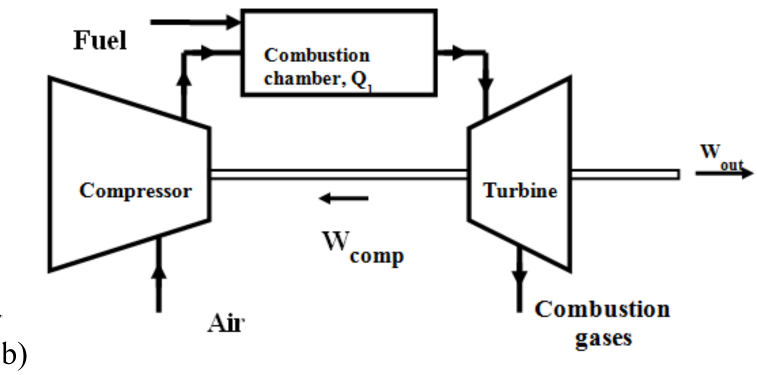


Fig. 1. a) The Brayton cycle and b) diagram of a simple gas turbine

Current turbine engines work with high inlet temperature of gas. This has implications for engine efficiency, which is expressed with a change of mass, cross-section and fuel consumption. If the inlet temperature is higher, then the greater specific volume of combustion gases, the more energy is received from expansion. Fig. 2 shows the change of turbine inlet temperature (TIT) over the years of the Rolls-Royce engines development [13]. Fig. 3 presents dependence between TIT and specific fuel consumption [13]. It can be noted that increasing inlet temperature considerably reduces the usage of fuel.

An overall increase of the allowable temperature for stator and rotor blades of high-pressure turbines was mainly possible because of introduction of cooling turbine components and improvement in metallurgy. The development of gas turbines has to be parallel with the development of aerodynamics, the theory of combustion process and material sciences. In connection with all this, modern advanced engines has a pressure ratio 35 to 1, achieving an overall efficiency of up to 85-90%, and the temperature of their combustion chamber exhaust gases is above 1650 K [13]. These include, in particular, jet turbine engines or by-pass turbojet and turbo-prop engines.

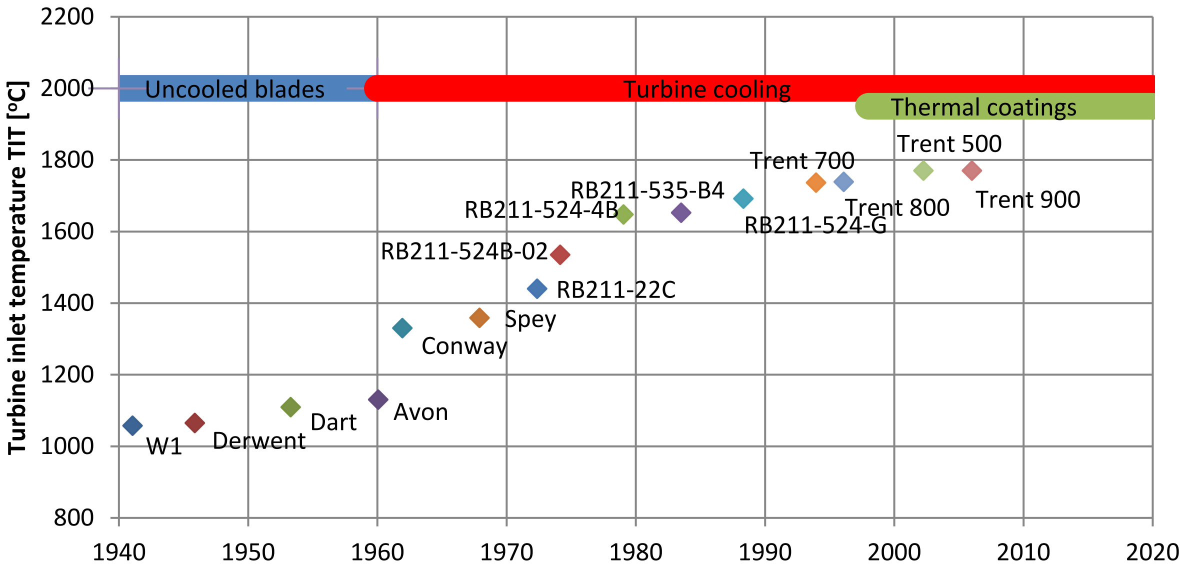


Fig. 2. The development turbine inlet temperature (TIT) over the years for the Rolls-Royce engines [13]

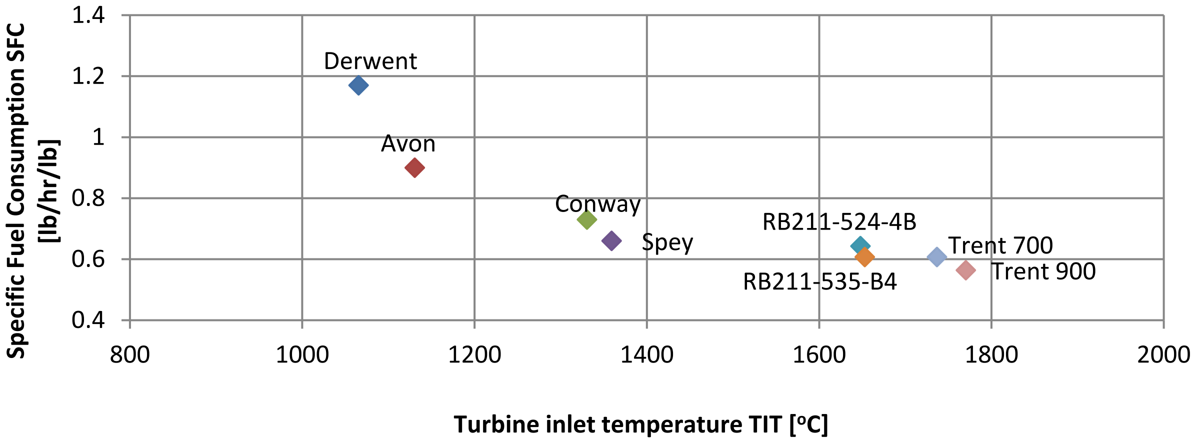


Fig. 3. Specific fuel consumption versus turbine inlet temperature [13]

The calculation procedure for the preliminary design of axial flow turbine consists of six major steps, as follows:

– estimation of the number of stages, their efficiency, pressure and temperature ratio,
– determination of the velocity triangles,
– calculation of stage static temperature and pressure,

– determination of stage pressure loss,
– geometry of blade calculation for each stage,
– geometry of stage calculations (number of blades).

Figure 4 shows an algorithm preliminary design of HPT.

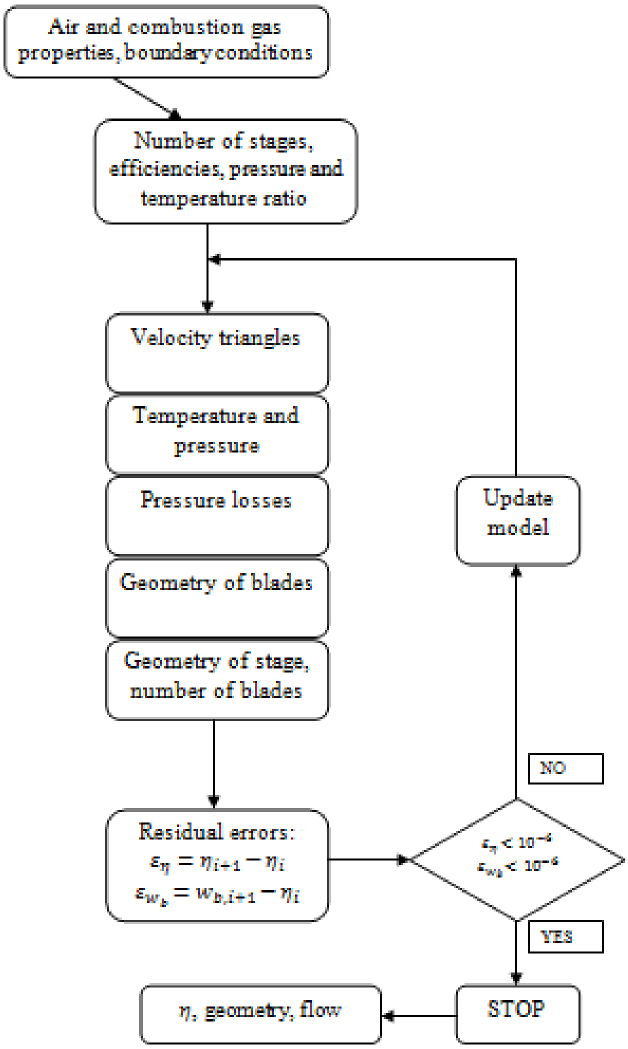


Fig. 4. An algorithm preliminary design of HPT

Design process of a turbine requires both engineering judgment and knowledge about the typical design values. Tab. 1 provides information on range of axial-flow turbine design parameters that can be used for guidance.

Tab. 1. Range of axial-flow turbine design parameters [11]

Parameter	Design range
Stage loading coefficient	1.4-2
Exit Mach number	0.4-0.7
Exit swirl angle	0-20 deg

The article presents a literature review concerning analytical methods of preliminary design high-pressure turbines. Each point of the algorithm was discussed. The aerodynamic design process is highly iterative, multidisciplinary and complex. Due to this, the modern gas turbines need a sophisticated design tools in terms of aerodynamics, mechanical properties and materials.

2. Literature review

Number of turbine stages

The first step is to determine number of turbine stages. Number of stages, at assumed enthalpy loss, depends on:

- the values of allowable work at one stage,
- the indicated value of effectiveness of turbine,
- the maximum permissible dimensions and mass of turbine.

The maximum value of allowable work at stage is limited by stage loading coefficient. The turbine must provide the required shaft work (Δh_0) and run at the same speed as the compressor (U). The equation that describes the stage loading (ψ) coefficient is as follows:

$$\psi = \frac{\Delta h_0}{2U^2}$$

Stage loading coefficient may achieve values from 1.7 to 1.85 for all stages except the last stage, for which the stage-loading coefficient should be between 1.4 and 1.5. Tab. 2 summarizes typical multi-stage turbine inlet and outlet parameters.

Tab. 2. Typical multi-stage turbine inlet and outlet parameters [11]

Parameters	Front stages (HP)	Last stages (LP)
α2	75^0-70^0	65^0-60^0
Degree of reaction	0.20-0.25	0.35-0.45
Exit Mach number (M3)	0.25-0.35	0.5 for turbojet and turbofan engines 0.65 – 0.70 for turbo-prop engine
Exit swirl angle (α3)		0-10^0

Efficiency

The next step is to determine efficiency for all stages. In [12] Smith developed a widely used efficiency correlation. Fig. 5 shows the data points and efficiency curves found by Smith. Data was

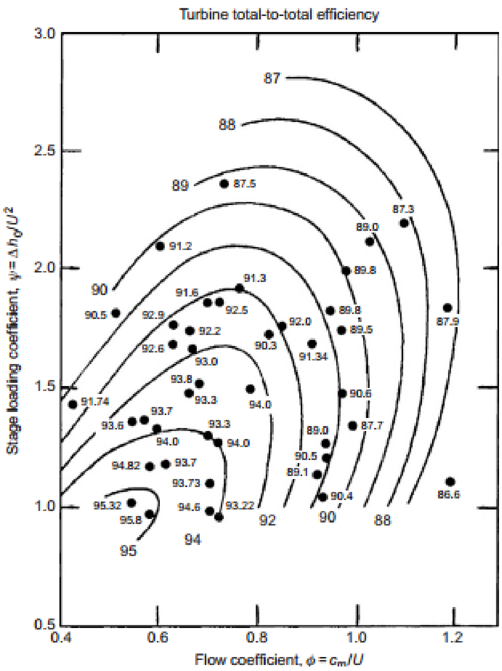


Fig. 5. Smith chart for turbine stage efficiency [12]

elaborated on the basis of 70 Rolls-Royce aircraft gas turbines. During experimental tests all stages were tested with constant axial velocity, the reaction degree was between 0.2 and 0.6 and the blade aspect ratio (blade height to chord ratio) has been assumed between 3 and 4. All efficiency values were corrected to eliminate tip leakage loss so that, in actual operation, the efficiency would be higher than this expected for the equivalent, real turbines.

Velocity triangles

Velocities of the working fluid in a turbomachine can be presented in the form of a graph known as a “velocity triangle” or a “velocity diagram”. Two main ways of presenting velocity triangles could be found in the literature. These two types of diagrams differ from each other by direction from which the angle is measured (from the axial or the tangential direction). The appropriate velocity triangles are shown in Fig. 6. The angle in this article is measured from the axial direction. Gas with an absolute velocity V_1 making an angle α_1, enters the nozzle or stator blades (in impulse or in reaction turbine respectively). Gas leaves the nozzles or stator blades with an absolute velocity V_2, which makes an angle α_2. The rotor-blade inlet angle will be chosen to suit the direction β_2 of the gas velocity V_{2R} relative to the blade at inlet. β_2 and V_{2R} are found by subtracting the blade velocity vector U from the absolute velocity V_2.

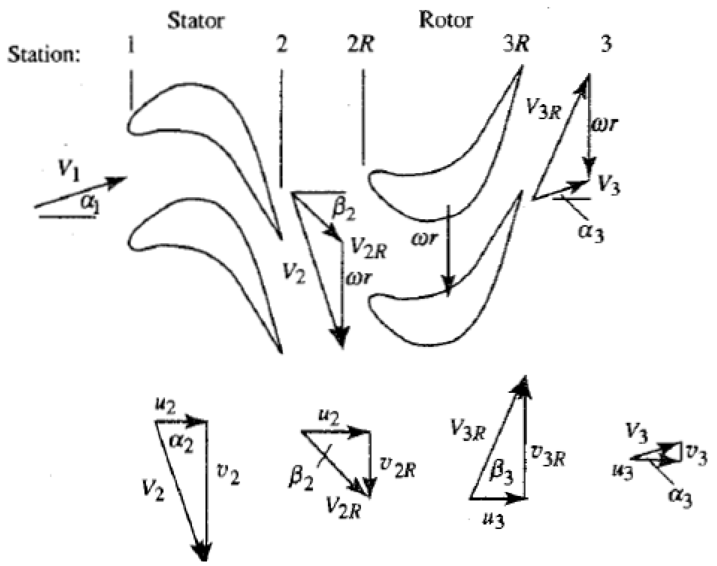


Fig. 6. Velocity triangles for a typical one-stage turbine [11]

Pressure losses

The complexity of flow hinders the general description of energy losses, so the losses are divided into groups that differ in physical and geometric determinants. Topic of pressure losses is voluminous and cannot be reproduced in detail in one short review article. Due to this, only a general description is presented below.

The aerodynamic losses in the flow through the turbine stage are non-stationary. They reduce the efficiency of energy conversion. They are caused by the strong impact of the rotor blades on the stationary vanes and by the turbulent phenomena, which are mainly responsible for blade losses, which include:

- profile losses,
- vortex (secondary flow) losses,
- tip clearance losses,
- shock loss,

- trailing edge losses,
- leakage bypass losses,
- disc friction losses,
- moisture losses.

The energy dissipation, occurring in the turbine stage, is increasingly analysed using empirical tests [1-6, 9].

With empirical correlation, it is possible to model pressure loss. This correlation was determined based on experimental data obtained by testing the airfoil cascades.

Geometry of blades

The next step is to determine an axial chord, a spacing between stator and rotor and a height of turbine blades (Fig. 7). The height of turbine blades shall be set on the basis of the continuity equation. In preliminary design both, the axial chord and the axial spacing can be estimated based on calculated height.

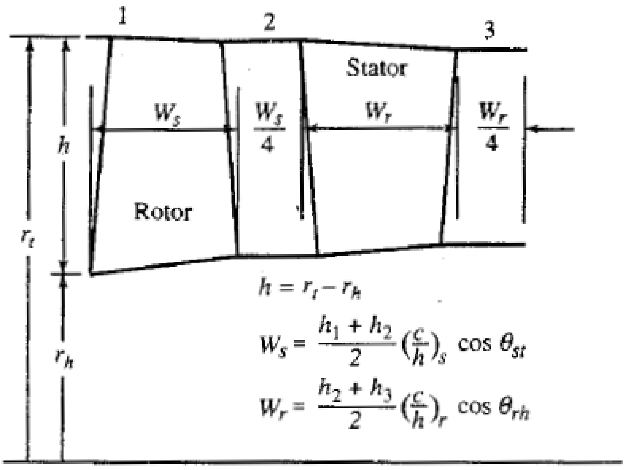


Fig. 7. The axial chord, the spacing between stator and rotor and the height of turbine blades [11]

Geometry of a stage

The choice of the number of blades can be based on empirical models, which determine the optimal pitch to chord ratio. One of the most popular estimates is the Ainley-Mathieson method [1] based on the minimum profile loss coefficients. The optimum pitch to chord ratio $(\frac{s}{c})$ for simple nozzle, blades can be approximated by:

$$\left(\frac{s}{c}\right)_0 = 0.427 + \frac{90-\alpha_2}{58} - \left(\frac{90-\alpha_2}{93}\right)^2.$$

Similarly, for the impulse blades:

$$\left(\frac{s}{c}\right)_1 = 0.224 + (1.575 - \frac{90-\alpha_2}{90})\frac{90-\alpha_2}{90}.$$

Defining:

$$\xi = \frac{(\beta_2)}{(\alpha_2)}.$$

The optimum pitch to chord ratio:

$$\left(\frac{s}{c}\right)_{opt} = \left(\frac{s}{c}\right)_0 + \left[\left(\frac{s}{c}\right)_1 - \left(\frac{s}{c}\right)_0\right] |\xi|\xi,$$

where: α_2, β_2 – see Fig. 6.

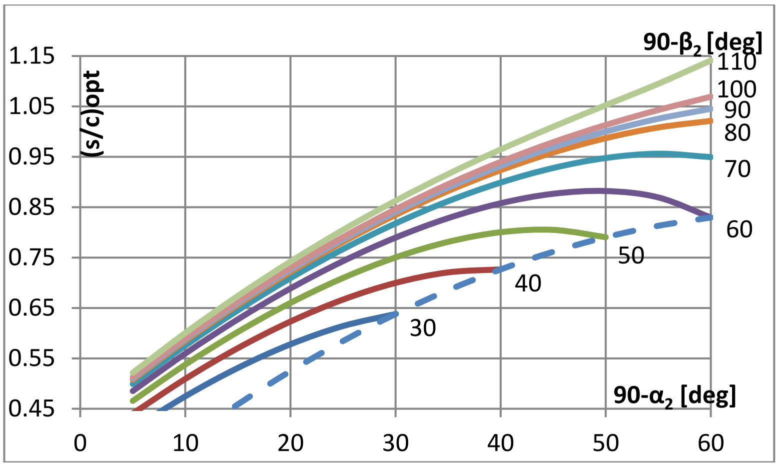


Fig. 8. The Ainley-Mathienson optimum pitch to chord ratio

Zweifel [16] developed an empirical method based on observing the effect of the lift coefficient. The Zweifel lift coefficient is given by the formula:

$$Z = 2\left(\frac{s}{c_x}\right)\cos(\alpha_{ex})^2\left[\tan(\alpha_{in}) + \frac{u_{in}}{u_{ex}}\tan(\alpha_{ex})\right]\left(\frac{u_{in}}{u_{ex}}\right)^2,$$

thereby:

$$\left(\frac{s}{c_x}\right)_{opt} = \frac{(Z)_{optym}}{2\cos(\alpha_{ex})^2\left[\tan(\alpha_{in})+\frac{u_{in}}{u_{ex}}\tan(\alpha_{ex})\right]\left(\frac{u_{in}}{u_{ex}}\right)^2},$$

where:
subscript: „in” mean inlet value, „ex” exit value,
$c_x = c \cdot \cos(\theta)$ – axial chord blade,
θ –stagger angle.

3. Real jet engine model

Major data source for estimating the temperature and pressure at the turbine inlet and exit was thermodynamics parameters for particular engine sections. For this purpose, a real jet engine model was prepared. The assumptions on which the model is based are as follows:

- air and combustion gases properties are independent from temperature and are described by Clapeyron relation,
- air leaks are ignored and mass flow of combustion gases shall be treated as sum of fuel and air mass flow,
- processes, occurring in specified engine sections, shall be treated as adiabatic and characterized by their efficiency.

Using this model the characteristics of an EJ200 engine (Fig. 9) have been developed.

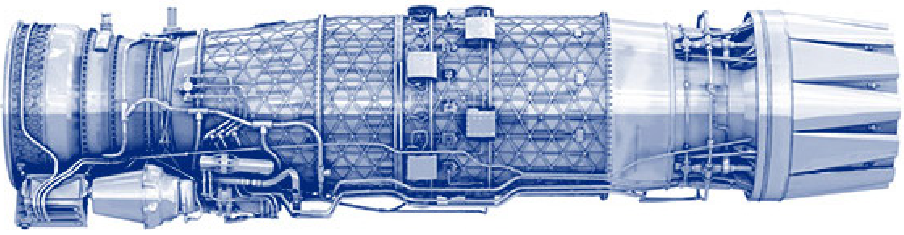


Fig. 9. EJ200 engine [7]

The technical specification provided by the engine manufacturers are reported in Tab. 3. The results of the calculations, pursuing those values with this model are reported in Tab. 4.

Tab. 3. Engine specification [7]

Thrust	13250 lbf =58 kN
Pressure ratio	26
TIT	1230 K
Mass flow	76 kg/s

Tab. 4. EJ200

EJ200	**Value**	**Unit**
Ambient pressure	101325	Pa
Ambient temperature	288	K
Inlet Pressure loss factor	0.97	-
Combustion chamber pressure loss coefficient	0.97	-
Nozzle Pressure loss factor	0.98	-
Compressor isentropic efficiency	0.88	-
Turbine isentropic efficiency	0.9	-
Thermal efficiency of the combustion chamber	0.98	-
Mechanical efficiency of turbine (compressor connection)	0.99	-
Specific heat of air	1000	J/kg/K
Heat capacity ratio of air	1.4	-
Individual gas constant	287	J/kg/K
Heat capacity ratio of flue gas	1160	J/kg/K
Heat capacity ratio of exhaust gases	1.33	-
Calorific value of fuel	43000000	J/kg
Vh=0 H=0 m		
Pressure before the compressor	98	kPa
Temperature before the compressor	288	K
Entropy change	8.7	J/kgK
Pressure behind the compressor	2.6	MPa
Temperature behind the compressor	791	K
Compression work	503	kJ/kg
Entropy change	75	J/kg/K
Compressor power	38	MW
Pressure at the outlet of the combustion chamber	2.5	MPa
Temperature at the outlet of the combustion chamber	1230	K
Heat	636	kJ/kg
Relative fuel consumption	0.015	-
Entropy change	450	J/kg/K
Fuel mass flow	1.15	kg/s
Exhaust gas temperature behind the turbine	799	K
Total exhaust gas pressure behind the turbine	0.3	MPa
Turbine expansion	7.3	-
Turbine work	500	kJ/kg
Entropy change	70	J/kg/K
Turbine power	39	MW
Outlet static pressure	101325	Pa
Outlet total pressure	0.3	MPa
Nozzle Total temperature	799	K
Outlet static temperature	595	K
Flow rate of the exhaust stream	687	m/s
Nozzle expansion	3.3	-
Entropy change	5.8	J/kgK
Specific thrust	698	Ns/kg
Specific Fuel Consumption	0.00002	kg/Ns
Thrust	53	kN

The analysis indicates that this model gives ales than 10% error in the engine thrust value. The calculation method, which is presented here, may be used for preliminary engine analysis. However, it must be recognised, that a number of simplifications was implemented.

At present, in the design of aircraft turbine engines, the specialized computer codes are commonly used. Available software includes Numerical Propulsion System Simulation (NPSS), GasTurb, TURBN and T-AXI.

Numerical Propulsion System Simulation (NPSS) [8] was developed in close cooperation between the US industry and NASA. This is a development environment based on object modelling, written in C++. The main areas of application of NPSS are the aerial engine systems. The advantage of this software is its full flexibility in modelling the entire system as well as the individual components.

One of the several programs that can be used at the initial stage of aviation engine design is GasTurb [10] written by a former employee of MTU. It benefits from a small hardware requirements and intuitive operation. The program's interface allows easily and quickly creating models of different types of the engines.

TURBN [11] is a program for the initial design of single or multi-stage turbines. This code is based on the curvature of the mean line, available with the book in which it was described.

The T-AXI [14-15] program enables the design of compressors and includes the turbine design code.

4. Conclusions

Current turbine engines are characterised by the high inlet gas temperature. This has implications for an engine efficiency, which is expressed with a change of mass, cross-section and fuel consumption.

Design process of a turbine requires both engineering judgment and knowledge of typical design values. The article presents the literature review concerning analytical methods of preliminary design high-pressure turbines. Each point of the algorithm was discussed. The aero design process is highly iterative, multidisciplinary and complex.

Major data source for estimation of the temperature and pressure at the turbine inlet and exit was a set of thermodynamics parameters for particular engine sections of known, existing engine. Due to this, an effective tool to develop the engine characteristic is needed. For this purpose, a real jet engine model was prepared. The assumptions, on which the model is based:

- air and combustion gases properties are independent from temperature and are described by Clapeyron relation;
- air leaks are ignored and mass flow of combustion gases shall be treated as sum of fuel and air mass flow;
- processes occurring in specific engine sections shall be treated as adiabatic and characterized by their efficiency.

As it has been shown in the article, this model gives an error level in engine thrust value of about 10%. The calculation method may be used for preliminary engine analysis. However, it must be recognised that a number of simplifications was implemented.

At present, in the design, manufacturing and operation of aircraft turbine engines, a set of specialized computer programs are commonly used. Available software includes Numerical Propulsion System Simulation (NPSS), GasTurb, TURBN and T-AXI.

References

[1] Ainley, D. G., Mathieson, G. C. R., *A Method of Performance Estimation for Axial-Flow Turbines*, Tech. rept. Aeronautical Research Council, London 1951.

[2] Craig, H. B. M., Cox, H. J. A., *Performance estimation of axial flow turbines*, Proc. Inst. Mech. Engrs., 71, Vol. 185 32/71, 1970.
[3] Craig, H. R. M., Cox, H. J. A., *Performance estimation of axial flow turbines*, Proc. Inst. Mech. Eng., 185, 32/71, 407-424 , 1971.
[4] Denton, J. D., *Loss mechanisms in turbomachines*, Trans. ASME J. Turbomachinery, 115, 621-656, 1993.
[5] Denton, J. D., *Loss mechanisms in turbomachines, Part I – Entropy creation in fluid flows, Part II – Loss generation in turbomachines*, VKI LS 1999-02, 1999.
[6] Dunham, J., Came, P. M., *Improvements, to the Ainley-Mathieson method of turbine performance prediction*, Trans. ASME, J. Eng. for Power, Series A, No. 3, 252, 1970.
[7] https://www.rolls-royce.com/products-and-services/defence-aerospace/products/combat-jets/ej200.aspx#engine-specifications.
[8] Jones, S. M., *An Introduction to Thermodynamic Performance Analysis of Aircraft Gas Turbine Engine Cycles Using the Numerical Propulsion System Simulation Code*, NASA/TM—2007-214690, 2007.
[9] Kocker, S. C., Okapuu, V., *A mean line prediction method for axial flow turbine efficiency*, Trans. ASME, J. Ens. for Power 1, 1982.
[10] Kurzke, J., *How to get component maps for aircraft gas turbine performance calculations*, Gas Turbine and Aeroengine Congress & Exhibition, Birmingham 1996.
[11] Mattingly, J. D., *Elements of propulsion: Gas turbines and rockets*, Reston, Va: American Institute of Aeronautics and Astronautics, 2006.
[12] Smith, M. H., *A simple correlation of turbine efficiency*, Journal of Royal Aeronautical Society, 1965.
[13] The Jet Engine, Rolls-Royce, 2005.
[14] Turner, M. G., Merchant, A., Bruna, D., *A Turbomachinery Design Tool for Teaching Design Concepts for Axial-Flow Fans, Compressor and Turbines*, ASME Turbo Expo GT2006-90105, May 8-11, Barcelona, Spain 2006.
[15] Turner, M., G., Merchant, A., Bruna D., *Turbomachinery Compressor Design (Part of the T-AXI suite of codes)*, Version 1.1, User manual, 2006.
[16] Zweifel, O., *Optimum Blade Pitch for Turbo-Machines with Special Reference to Blades of Great Curvature*, Brown Boveri Review, The Engineer's Digest, 1946.