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Virtual Prototyping of Gas Turbine Components – Aerodynamic Redesign and Analysis of Academic Jet Engine

Booklet of PhD Theses

A Dissertation submitted by:

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Summary

Todays the gas turbines for jet engines are the only available power systems for commercial and military airplanes at subsonic and low-speed supersonic flow regimes. Although their thermodynamic cycles do not have high core thermal efficiency (~ 28 to 40 %), the jet engines have substantial advantages in overall power, power density, compactness, streamlining, simplicity and low maintenance cost demands. These power units are also less sensitive to overloads and produce less vibration due to the well balanceable and rather axisymmetric rotational components. The gas turbines have high availability (80-99 %), reliability, which can exceed 98 % and low emissions (there is no lubricant in the combustion chamber and no soot during transient loads). They have fewer moving parts and lower sensitivities to fuel composition. Additionally, gas turbines do not need a liquid-based cooling system, although the maximum allowable temperature (~ 1500 C°) at the turbine inlet section is limited for metallurgical reasons [1].

Beside the technical level of gas turbines today, there are many possible areas for potential improvements of their efficiency, power and emissions. Although the experiences and the know-how of the gas turbine manufacturers are increasing continuously, developing more accurate mathematical models for designs and analyses can significantly contribute to decrease cost, time and capacity in the early phases of design and developments. These are proven

i. by the fact of frontloading product development; it reconfigures the conventional design processes and several sampling phases can be omitted by the virtual prototyping[1] and

ii. by the scientific literature; there are many research publications deal with modelling, design, analyses and optimization.

[1] Based on our experiences via pilot project in the industrial pre-development field, the cost, time and capacity are decreased by 50 %, 60 % and 20 % respectively in case of simulation driven product development.

Hence, the goal of the present thesis is to develop, crosscheck and verify calculation processes, which can be used for modelling, design and analysing jet engines in aerodynamic point of view with especial care for a specific academic jet engine under consideration. The introduction of the new calculation approaches for improving accuracy and technical characteristics of the investigated propulsion systems are also the parts of the present work. Beside the available single-, dual-, and triple-spool jet engines with their characteristics found in the international open literature, the considered single spool engine is a variant of the TSz-21 starter gas turbine, which is used originally for MiG-23 and Szu-22 Russian fighters. The TSz-21 gas turbine was reconstructed to be an academic/research propulsion system by Dr. Beneda and Dr. Pásztor from 2005-2008 and it is still under development with especial care for control systems [2].

The introduced aerodynamic design process has four main steps as

1. improved thermo-dynamic cycle analyses for determining the design (operational) point,
2. mean line and 3D design of the engine for having geometrical sizes and CAD models,
3. Computational Fluid Dynamics (CFD) analysis for verification of the design and plausibility check by the available data and
4. optimization of the vanned diffuser with inverse design of the solid walls - vanned diffuser in the present case.

Following the market research, the determination of the customer requirements, the specifications and the selection of the cycle type of the turbomachinery layout, the first step of the jet engine design is the thermodynamic cycle analysis for determining the design point. A concentrated parameter distribution-type method has been developed, implemented and verified for analysing the characteristics of jet propulsion engines by means of a) available and b) expected specifications as follows.

a) Regarding the scenario about the available specifications, the following triple-, dual- and single-spool turbojet engines are considered in the analyses [3]:
   i. НК-32 and НК-25 triple spool mixed turbofan engines at take-off condition with afterburner,
   ii. НК-22 and НК-144А dual spool mixed turbofan engines at take-off and at flight conditions with and without afterburning respectively, НК-8-4 and НК-86А dual spool mixed turbofan engines at take-off condition without afterburning and
   iii. ВД-7 and КР7-300 singles spool turbojet engines without afterburner and РД-9Б and the АЛ-21Ф3 single spool turbojet engines with afterburning at take-off condition.

   The developed mathematical model for analysing the above-mentioned engines is based on the mass and energy balance together with the thermo-dynamic process equations including frictional (viscous) flow related losses. Constrained nonlinear optimisation is used for determining the unknown parameters as efficiencies, losses, power reduction rates of the auxiliary systems, bleed air ratios for technological reasons, air income ratios due to blade cooling and total temperatures in the afterburner (if these are relevant) by means of having parameter-state, which provides the closest results to the available thrusts and thrust specific fuel consumptions. The temperature and component mass fraction dependent gas properties as specific heat at constant pressure and ratio of specific heats are determined by iteration cycles. As a part of the model development, a new closed-form equation is derived for determining the critical pressure in choked flow condition at converging nozzle with considering losses and process-dependent gas properties. New explicit expression is derived also for calculating the optimum total pressure ratio of the compressor pertaining at maximum specific thrust at chocked and unchoked nozzle flow conditions. РД-9Б and the АЛ-21Ф3 single spool turbojet engines with afterburning at take-off condition are used for verification and plausibility check of the new equation about the optimum compressor total pressure ratio.

b) Concerning the case about the expected specifications or design scenario in other words, the already developed and verified thermodynamic cycle analysis mentioned in point a) above is used to redesign the TSz-21 gas turbine to have a 330 N thrust low-sized academic jet engine. The specific thrust and thrust specific fuel consumption distributions are determined in the function of the turbine inlet total temperature and total pressure by the thermodynamic analysis for determining the operating point of the engine.

After having the design point by the thermodynamic cycle analysis, the second step of the engine aerodynamic design is the determination of the geometrical sizes and Computer Aided Design (CAD) model preparation of the assembly. The mean line design and its 3D extension have been considered, realized for creating the compressor and turbine segments in line with the outcomes of the thermo-dynamical cycle analysis described in point b) above. The configuration and the sizing of the intake channel as well as the combustion chamber and exhaust nozzle are determined by using the dimensions of the compressor and turbine, beside guidelines, theoretical and practical solutions, suggestions and experiences for shaping. As the all geometrical dimensions become available at the end of this state, the 3D model generations are completed for the intake channel, compressor, combustion chamber, turbine and exhaust nozzle (see Figure 1.).

CFD analyses are completed in the third step of the design process to crosscheck the differences between the expected and the computed characteristics of the engine. Two different simulation approaches are applied: 1. separated engine components and 2. full engine model (all engine components are together). The results of the simulations are compared with the results of the i. thermodynamic cycle analysis, ii. mean line design and iii. available measured data for verification and plausibility check. Conclusions are drawn about the agreement and the effectivity of the used analytical and numerical methods.

Finally, a 2D inviscid inverse design method is adopted, implemented and used for improving the characteristics of the vanned diffuser in the compressor unit in order to increase the total pressure recovery factor, static pressure rises, flow turning in axial direction and the mass flow rate per unit length beside having the expectation to keep the original geometrical configuration and dimensions as much as possible. The results of the inverse design method are verified by a commercial CFD code at both inviscid and viscous flow conditions.

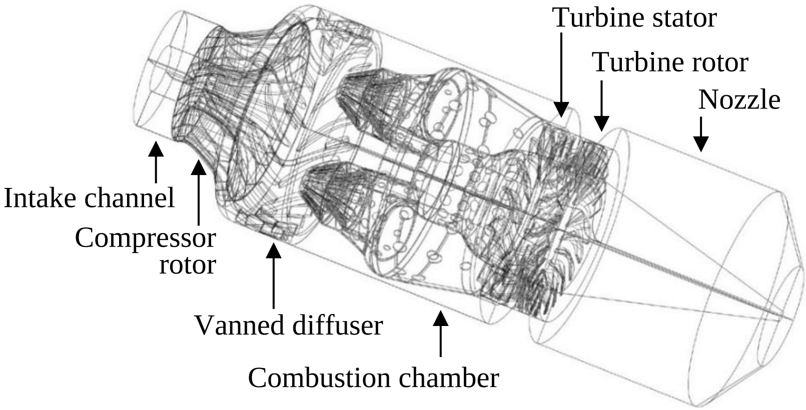

Figure 1. 3D wireframe model of the assembly and the main components of the investigated academic jet engine

Although the presently introduced aerodynamic design process is completed for a single spool academic turbojet engine, it can be applied for other configurations following the necessary adaptations, modifications and extensions.

The above mentioned 4 step design process focuses on the aerodynamic design only. The other contributions as mechanical (both static and dynamic), Noise Vibration and Harshness (NVH) (incl. eigenfrequencies, harmonic response, Power Spectral Density (PSD), vibro- and aeroacoustics), thermal, fatigue, creep, wearing, durability and ageing aspects of the solid components and joints including design loops together with the already mentioned aerodynamic design scenarios and extending that with control, electronics, tests, production related topics and investigating off-design performances are indispensable part of the design process but they are excluded from the present study.

New Scientific Results

Advanced aerodynamic design process has been developed and applied for a turbojet engine with especial care *i.* for deriving new equations and procedures to increase the accuracy of thermo-dynamic cycle analyses and *ii.* for introducing inverse design methodology for redesign vanned diffuser in order to optimize design specifications. Jet engines with available technical parameters in the open literature are used for the verification and plausibility check of the thermodynamic models, meanwhile an in-house academic turbo jet engine has been considered for design, design-verification and plausibility check.

Thesis 1:

The accuracy of the critical pressure at the exit of the converging nozzle of jet engines is increased by considering specific heats at constant pressure and ratios of specific heats are the functions of temperatures and fuel to air mass flow rates ratio. ***Hence, the following equation has to be used for determining critical static pressure at the outlet of the converging nozzle:***

$$p_C = p_{07}\left[1-\frac{1}{\eta_n}\left(1-\frac{2\bar{C}_{pmix}(T_{09},T_C,f)}{2\bar{C}_{pmix}(T_{09},T_C,f)+\gamma_{mix}(T_C,f)R_{mix}}\right)\frac{\bar{C}_{pmix}(T_{09},T_C,f)}{2\bar{C}_{pmix}(T_{09},T_C,f)}\right]^{\frac{\bar{\gamma}_{mix}(T_{09},T_{9s},f)}{\bar{\gamma}_{mix}(T_{09},T_{9s},f)-1}} \quad (1)$$

where p_{07} is the total pressure [Pa] at the upstream of the nozzle, η_n is the isentropic efficiency of the nozzle [-], $\bar{C}_{pmix}$ is the average specific heat [J/(kgK)], T_0 and T is the total and static temperatures respectively [K], f is the fuel to air mass flow rates ratio [-], R_{mix} is the specific gas constant of the gas mixture [J/(kgK)], $\bar{\gamma}_{mix}$ is the average ratio of the specific heats [-] and "C" in subscript means critical. The standalone parameters are valid at the given nozzle sections meanwhile the averaged ones are at the given temperature ranges and fuel to air ratios.

The Eq. *(1)* gives higher critical pressure by 9.3 % for the РД-9Б single spool turbojet engine than its original form shown by Eq. *(2)* with constant gas parameters (the ratio of specific heats for the gas is 1.33).

$$p_C = p_{07}\left(1-\frac{1}{\eta_n}\left(\frac{\gamma_{gas}-1}{\gamma_{gas}+1}\right)^{\frac{\gamma_{gas}}{\gamma_{gas}-1}}\right) \tag{2}$$

The thermodynamic analyses of the НК-32 and НК-25 triple spool jet engines are also completed including the determination of the critical pressure at the exit of the converging nozzle by using a variant of Eq. *(1)* and Eq. *(2)*. The average deviation between the resulted (by the parameter fitting) and the available thrust and thrust specific fuel consumption [3] is 0.119 % by using a variant of Eq. *(1)* and 3.13 % at the conventional equation Eq. *(2)* at the same other conditions. The "variant" in the present context means that the true specific heats at constant pressure and at the given temperatures are considered instead of the averaged one in the total enthalpy equation.

Related publications: [4, 5 and 8]

Thesis 2:

The accuracy of the optimum total pressure ratio of the turbojet engines pertaining at maximum specific thrust is improved by applying frictional (viscous) flow assumptions beside the temperature and mass fraction dependencies of the relevant properties of the gas mixture. ***Therefore, the following expression has to be used for determining the optimum total pressure ratio of turbojet engines at maximum specific thrust:***

$$\boldsymbol{\pi_{C\_opt}} = \sqrt[\beta]{\frac{\varepsilon(1+\phi)}{\phi(\varepsilon+\beta)}}, \tag{3}$$

where

$$\boldsymbol{\beta} = \frac{\bar{\gamma}_{mix}(T_{02},T_{03s},f=0)-1}{\bar{\gamma}_{mix}(T_{02},T_{03s},f=0)}, \tag{4}$$

$$\boldsymbol{\varepsilon} = \frac{\bar{\gamma}_{mix}(T_{04},T_{05s},f_T)-1}{\bar{\gamma}_{mix}(T_{04},T_{05s},f_T)}, \tag{5}$$

$$\boldsymbol{\phi} = \frac{\bar{C}_{pmix}(T_{02}T_{03},f=0)}{\bar{C}_{pmix}(T_{04}T_{03},f_{cc})}\frac{T_{02}}{T_{04}}\frac{1}{\eta_m\eta_{C,s}\eta_{T,s}(1-\delta_{tech})(1+\delta_{bc})(1+f_{cc})(1-\xi)}, \tag{6}$$

$\bar{\gamma}_{mix}$ is the average ratio of the specific heats of gas mixture [-], T_0 is the total temperature [K], f is the fuel to air mass flow rates ratio [-] ($_{cc}$: combustion chamber), $\bar{C}_{pmix}$ is the average specific heat at constant pressure for the gas mixture [J/(kgK)], η is the efficiency [-] ($_s$: isentropic, $_m$: mechanical, $_T$: turbine and $_C$: compressor), δ_{tech} is the

bleed air ratio for technological reasons [-], δ_{bc} is the ratio of the incoming air due to blade cooling [-] and ξ is the power reduction rate of the auxiliary systems [-]. The standalone parameters are valid at the specified engine sections meanwhile the averaged ones are at the given temperature ranges and fuel to air ratios.

In case of viscous (frictional) flow conditions and variable specific heats, which are included in Eq. *(3)*, the calculated optimum total pressure ratio is higher by 40 % and 53.3 % for the РД-9Б and АЛ-21Ф3 jet engines respectively than their total pressure ratios found in the technical specification [3]. These newly calculated optimum total pressure ratios correspond to 3.39 % and 3.63 % thrust increments consistently. In addition, besides keeping the viscous flow assumption, if the specific heats (and so the ratios of specific heats) are defined to be constant, there are 9.52 % and 13.04 % decrements in the total pressure ratios for РД-9Б and АЛ-21Ф3 jet engines respectively.

Related publication: [4]

Thesis 3:

The thermo-dynamical model for design and analysis jet engines including single-, dual-, and triple spool configurations has to be established by the following way:

i. ***Mass and energy balance have to be applied,***

ii. ***Frictional (viscous) flow assumption has to be considered in the processes of the thermo-dynamical cycles,***

iii. ***The specific heats and the ratio of specific heats are the functions of the temperature and the fuel to air mass flow rates ratios. Iteration cycles have to be used for evaluating the temperatures and component mass fractions if they are the variables of the gas parameters.***

iv. ***The non-available technical data as efficiencies (e.g.: mechanical, isentropic of compressor and turbine, burning and exhaust nozzle), total pressure recovery factors (e.g.: inlet diffuser, combustion chamber and afterburner or turbine exhaust pipe), total pressure ratio of the fan and intermediate***

pressure compressor, power reduction rate of the auxiliary systems, bleed air ratio for technological reasons, air income ratio due to blade cooling and total temperature of the afterburner, if these are relevant – have to be determined either by

a) ***well known and established (practical) experiences,***
b) ***measurements,***
c) ***simulations (e.g.: CFD simulations for efficiencies and total pressure recovery factors) and***
d) ***parameter identification, in which the goal function of the optimization is to minimize the differences between the calculated and available thrust and thrust specific fuel consumption.***

v. ***Eq.*** *(1)* ***has to be used for determining critical static pressure at the exit of the converging nozzle.***

vi. ***Eq.*** *(3)* ***has to be used for determining the optimum compressor total pressure ratio at maximum specific thrust for turbojet engines.***

4 single spool (at one operational condition), 4 dual spool (two is at two and two is at one operational conditions) and 2 triple spool (at one operational condition) jet engines are analysed by the mathematical models introduced in the present thesis. The resulted thrusts and thrust specific fuel consumptions by constraint nonlinear optimisation - in order to find the unknown parameters – are compared with available data in [3] for verification. The maximum and the average relative deviation between them are 1.46 % and 0.645 % for the thrust and 0.55 % and 0.32 % for the thrust specific fuel consumptions respectively, meanwhile the identified parameters are within the plausible range.

Related publications: [4-6], [8], [9], [12] and [13]

Thesis 4:

The combination of the Stratford's separation prediction method – for providing the maximum closed surface area of the pressure distribution – and inviscid inverse design method – for creating the blade geometry belongs to this pressure distribution – have to be used in design, re-design and developments of vanned diffusers in

centrifugal compressor units for providing the optimum vane configurations by means of having:

- ***i.*** ***the minimal design changes with respect to the baseline configuration, if it is applicable, and in the present case:***
  - ***a.*** ***the splitter vanes are removed, and***
  - ***b.*** ***only the vanes with higher chord-length are kept and used as initial geometry for the inverse design method,***
- ***ii.*** ***the maximum static pressure-rise,***
- ***iii.*** ***the highest flow turning in axial direction,***
- ***iv.*** ***the largest mass flow rate***
- ***v.*** ***and the greatest total pressure recovery factor*** $\left(\frac{p_{total\_downstream}}{p_{total\_upstream}}\right)$***,***

meanwhile the flow is close but certain safe distance far from the separation as it is guaranteed by the Stratford's separation prediction method.

Regarding the redesigned (optimized) blade configuration, the average relative deviation of the pressure distributions between the results of the CFX at viscous flow condition and the inviscid in-house code is 3.04 % at suction side and 0.3 % at pressure side.

The aerodynamic performance of the base line design and the optimized vanned diffuser has been compared with each other by means of viscous flow analyses. The optimized version has

i. higher total pressure recovery factor by 3.13 %,
ii. higher static pressure-rise by 1.81 %,
iii. flow angle decrement measured from the axial (x) direction by 15.38 %, and
iv. higher mass flow rate per unit length by 3.2 %

than the original vane configuration has. 2D planar approach has been considered in the present methodology.

Related publications: [6], [7] and [10]

Thesis 5:

The following process has to be used for aerodynamic design of jet engines:

- ***i.*** ***thermo-dynamic cycle analyses according to* Thesis 1. 2. *and* 3. *for determining the design (operational) point,***

ii. ***establishment of the main geometrical sizes and the necessary RPM, if they are relevant,***
iii. ***mean line and 3D design with geometry preparation and CAD modelling,***
iv. ***CFD analyses for verification of the design and plausibility check by the available data and***
v. ***inverse design of the solid walls, vanned diffuser in the present case according to*** **Thesis 4.**

After completing the aerodynamic redesign of the academic jet engine in line with the above defined process, excluding the last steps, the conclusions are next. The average relative deviation – using data at engine unit sections – between the results of the thermodynamic cycle analysis and the CFD analysis in case of the full model is 1.05 % for the total temperature and it is 4.71 % for the total pressure. The average relative deviation between the results of the mean line design and CFD analysis in case of the full model also is 1.13 % for the total temperature and it is 1.56 % for the static temperature. Similarly, it is 4.59 % for the total pressure and it is 5.01 % for the static pressure. Although its accuracy is not verified statistically, available measured data is used for plausibility checking of the computed result. The average relative deviation between the available measured data and the results of the CFD analysis in case of full model is 1.98 % for the total temperature and it is 2.11 % for the static temperature. Similarly, it is 10.56 % for the total pressure and it is 3.43 % for the static pressure. The thrust of the academic jet engine is 333.1 N by the mean line design and 325.2 by the full-model CFD analysis. The expected target thrust is 330 N. The maximum relative deviation between these computed and the expected value is 1.46 % and it is at the full-model CFD analysis.

Related publications: [4-15]

Berlin, 31. 12. 2020.

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