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SIMPLIFIED MATHEMATICAL MODEL FOR A SINGLE SPOOL AND NO BYPASS JET ENGINE[4][5]

Abstract

A concentrated parameter distribution type mathematical model has been developed and implemented in MATLAB environment for modeling thermo dynamical cycles and characteristics of the single spool no bypass jet engines. The governing equations of the model are based on the real thermo dynamical processes, in which the mechanical efficiency, isentropic efficiencies, burning efficiency, pressure losses and the power reduction rate of the auxiliary systems were considered. Beside the material properties and geometrical data, the altitude, the flight speed, the total pressure ratio of the compressor, the turbine inlet total temperature and the air consumption are considered as the input parameters of the analysis. The outputs of the simulation are the overall efficiency, specific thrust and thrust specific fuel consumption together with the points of thermo dynamical cycle of the system in T-s diagram. The results of the analysis, as the specific thrust, thrust specific fuel consumption and the outlet total temperature of the turbine are compared with available operational data of the Tumansky R-29 type turbojet engine at sea level static condition.

EGYTENGELYES ÉS EGYÁRAMÚ SUGÁRHAJTÓMŰ EGYSZERŰSÍTETT MATEMATIKAI MODELLJE

Összefoglalás

Az ebben a publikációban dokumentált kutatási tevékenység célja egy koncentrált paraméter-eloszlású számítási eljárás kidolgozása és implementálása MATLAB környezetben egytengelyes és egyáramú gázturbinás sugárhajtóművek termodinamikai körfolyamatainak és jellemző karakterisztikáinak meghatározása érdekében. A valóságos áramlást modellező hőtani és energetikai folyamatok alapegyenletei a rendszer mechanikai hatásfokát, a kompresszor és a turbina izentropikus hatásfokát, a tüzelési hatásfokot, a nyomásveszteségi tényezőket és a kiegészítő berendezések teljesítményfelvétel-hányadát foglalják magukba. A folyamatokban résztvevő anyagok tulajdonságai és a szükséges geometriai méretek mellett a magasság, a repülési sebesség, a kompresszor torlóponti nyomásviszonya, a turbina belépő torlóponti hőmérséklete és a hajtóműbe belépő levegő tömegárama került megadásra bemeneti feltételként. A számítás eredményei a termikus hatásfok, a propulziós hatásfok, a teljes hatásfok, a fajlagos tolóerő, a tolóerőre vonatkoztatott fajlagos tüzelőanyag-fogyasztás és a termikus torlóponti állapotjelzők a T-s diagramban ábrázolt körfolyamat minden pontjában. Az eredmények ellenőrzése céljából a MÍG-23-as vadászrepülő Tumansky R-29 típusú hajtóművének a szakirodalomban fellelhető specifikációi kerültek alkalmazásra. A számítási eredmények és

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a rendelkezésre álló adatok közül a fajlagos tolóerő-, a tolóerőre vonatkoztatott fajlagos tüzelőanyag-fogyasztás- és a turbina utáni torlóponti hőmérséklet-értékek lettek egymással összehasonlítva tengerszinten és starthelyzetben.

NOMENCLATURE

Latin letters and abbreviations:

A	Inlet area of the engine [m]
a	Speed of sound [m/s]
C_P	Specific heat at constant pressure [J/kg/K]
$C_{p\_air\_mean}$	Mean (T_{03}-T_{04}) specific heat of air at constant pressure [J/kg/K]
$C_{p\_air\_T04}$	Real specific heat at constant pressure of air at T_{04} [J/kg/K]
c_{Pcom}	Mean (T_{02}-T_{03}) specific heat of air at constant pressure [J/kg/K]
C_v	Specific heat at constant volume [J/kg/K]
com	compression
d_1	Inlet diameter of the engine [m]
d_9	Outlet diameter of the engine [m]
exp	expansion
f	Fuel to air ratio [-]
H	Altitude [m]
h	Specific enthalpy [J/kg]
L_0	Theoretical air mass required to burn 1 kg fuel perfectly
M	Mach number
M_{air}	Molar mass of the air [kg/kmol]
$\dot{m}_{air}$	Mass flow rate of air [kg/s]
$\dot{m}_{fuel}$	Mass flow rate of fuel [kg/s]
p	Pressure [Pa]
p_a, p_0	Ambient static pressure [Pa]
Q_R	Lower heating value of the fuel [MJ/kg]
R	Specific gas constant [J/kg/K]
R_M	Universal gas constant [J/kmol/K]
rd	Total pressure recovery factor in the intake duct (diffuser)
rtd	Pressure recovery factor in the turbine exhaust nozzle (in the afterburner liner)
s	Entropy [J/kg]
s_0	Reference entropy [J/kg/K]
T	Temperature [K] or Thrust [kN]
T_0	Ambient temperature [K]
TSFC	Thrust Specific Fuel Consumption [kg/kN/h]
v	Specific volume [m^3/kg]
V_{el0}	Flight speed [m/s]

Greek letters and abbreviations:

ρ	Density [kg/m^3]
π_C	Total pressure ratio of the compressor
γ	Ratio of specific heats
ζ	Power reduction rate for auxiliary systems
η_d	Diffuser efficiency
r_{cc}	Total pressure loss of combustion chamber
η_b	Burning efficiency
η_m	Mechanical efficiency
η_{Prop}	Propulsive efficiency
η_{izC}	Isentropic efficiency of compressor
η_{izcT}	Isentropic efficiency of turbine

1. INTRODUCTION

Nowadays, the only gas turbines have energetically and economical capability to be applied in the propulsion system of the commercial and military aircraft due to their high power density and low range factor[6] at high speed flight below Mach number 1. The other important feature of the gas turbines is the less sensitivity for fuel composition, and it makes them more suitable in case of limiting amount of fossil based fuels. Additionally, these engines have been used as highly reliable and safe power resources with less maintenance.

Although the augmentation of the three main factors as the compressor pressure ratio, the turbine inlet temperature and the component efficiencies can strongly contribute to increase the performance of the jet engine [1], the operation of gas turbines is strongly cost demanding due to the high fuel consumption. Due to this fact, the improvement of the overall efficiency of gas turbines is a key point in the research field of the propulsion systems and turbomachinery related sciences. Hence, as a first approach for analyzing and design the operation of jet engines, the developments and the validation of a simplified thermo dynamical model is indispensable to have.

Regarding the literature research of the present topic, three different articles are mentioned. Yarlagadda focused on performance analysis of a J85 turbojet engine with an inlet flow control mechanism to increase rpm for same thrust values by using 1-D non-linear unsteady equations [3]. Sforza presents a general jet engine cycle, which is discussed in the enthalpy–entropy plane, and the concept of an ideal jet engine cycle is introduced. Detailed analysis of the performance of various types of jet engines operating ideally under different flight conditions is presented. The turbine inlet temperature is treated as a parameter, and thrust and specific fuel consumption results are discussed. A conclusion of the effects of real engine operations on the ideal results is given

[6]The range factor, in this manner, is the ratio of the weight of the fuel and the engine to the engine net thrust decreased by the nacelle drag for a range and flight speed [9].

[10]. Sanjay deals with thermodynamic analysis of a basic gas turbine based gas-steam combined cycle. The article investigates the effect of variation of cycle parameters on rational efficiency and component-wise non-dimensionalised exergy destruction of the plant [11].

In the present study, a concentrated parameter distribution type method has been developed and implemented to analyze the main characteristics of a jet engine at sea level. The results of the simulations are compared with the available operational data of Tumansky R-29 type turbojet engine (see Figure 1.), which are found in [8].

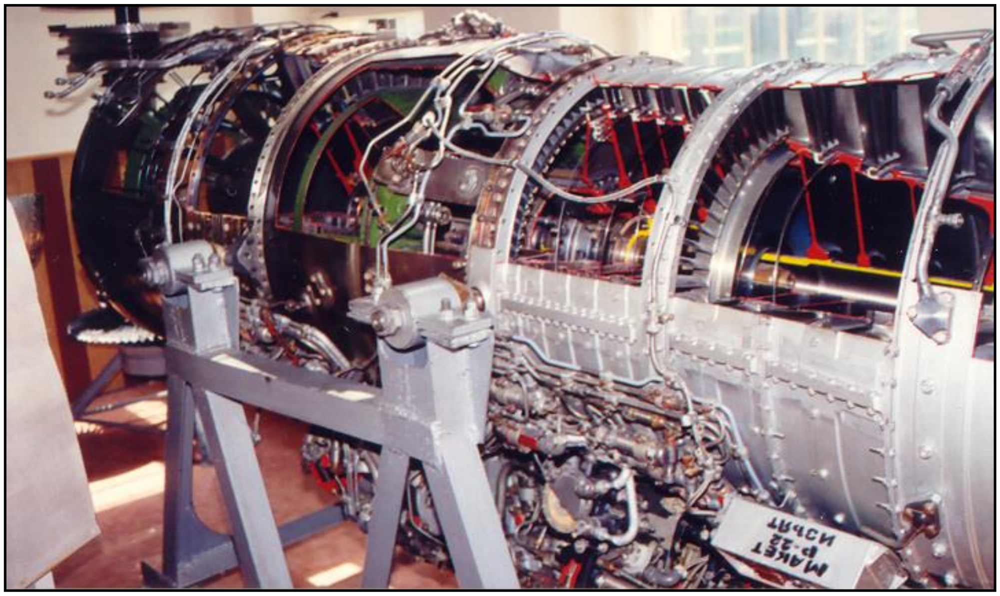

Figure 1. The opened real assembly of the Tumansky R-29 jet engine [12]

2. Governing Equations

As a first approach, steady thermo dynamical equations are used for calculating and plotting real cycle of the engine processes in T-s diagram. The turbine is divided by 8 main cross sections, which are the followings:

Sections	**Locations**
0	Free stream (ambient condition)
1	Engine intake
2	Compressor inlet
3	Compressor outlet
4	Turbine inlet
5	Turbine outlet
6	Exit of afterburner liner
9	Nozzle exit (convergent nozzle only)

The cross sections listed above are shown in Figure 2.

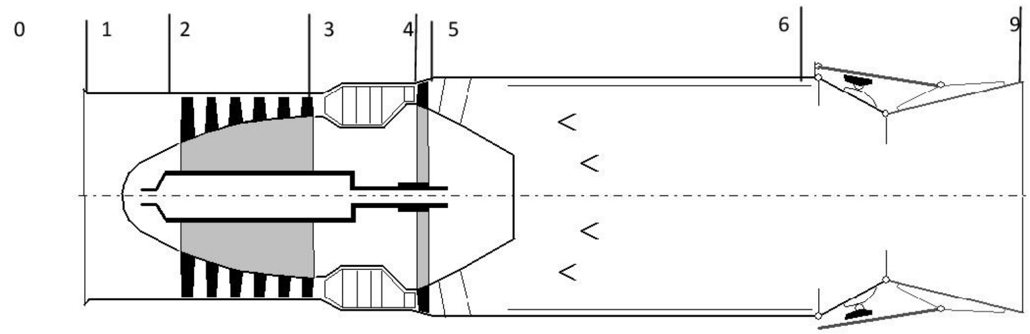

Figure 2. Sections of the turbojet engine for the analysis [7]

All the processes of the operational gas in the jet engine are frictional and so irreversible. Thus isentropic efficiencies of compressor, turbine and nozzle were applied beside the pressure loss in the intake and combustion chamber. The burning efficiency was also considered. With respect to Figure 2, the flow enters into gas turbine over section (0-2). The compressor delivers the air between section 2 and 3. The combustion processes develops from stage 3 to 4. The turbine is located among section 4 and 5. A non-operational afterburner is found between points 5 and 6. The last but not least part of the engine is the exhaust nozzle, which is located between sections 6 and 9 and only converging non-variable geometry nozzle has been considered in the present study.

The important components of aero-engine such as turbojet are the air intakes. The ambient air is coming into the duct where the flow is accelerated from free stream if the engine does not move. The compressor then increases both temperature (T) and pressure (p) of the gas. Work input is required to achieve the pressure ratio and the associated temperature rise depends on the efficiency level. After the air is compressed in the compressor, the fuel is added to the air and burned continuously in the combustion chamber to raise the exit gas temperature of fuel mixture from 1100 to 2600 °C approximately, depending on the engine level technology. The total pressure loss is developed in the diffuser and the combustor due to the flow losses as friction, collision and separation. Then the hot, high-pressure gas expands through the turbine where work is extracted to produce shaft power, which drives the compressor and auxiliary systems. Both the pressure and temperature are decreased. The exhaust pressure at the turbine is typically twice that of ambient [1]. Downstream of turbine the gas goes through the exhaust nozzle. The propelling nozzle is convergent that accelerated the flow to produce the high velocity jet to provide the thrust. According to the given data of the engine, the relevant different processes of the engine cycle can be calculated separately as follows.

2.1 Intake (0-2)

The ambient temperature and pressure values were taken from the standard atmosphere tables found in [1]. The intake total temperature can be calculated based on the equation (1).

$$\mathrm{T}_{01} = \mathrm{T}_0\left(1 + \frac{\gamma-1}{2}\mathrm{M}^2\right) \tag{1}$$

The total outlet pressure of the inlet diffuser can be expressed as follows:

$$p_{01} = p_0(1 + \frac{\gamma-1}{2}M^2)^{\frac{\gamma}{\gamma-1}} \qquad (2)$$

The intake velocity can be determined by an iteration cycle (from (5)-(8), until V_{el1}=V_{el1_new}), in which the first approximation is done by using ambient total parameters and incoming mass flow:

$$\rho_{01} = \frac{p_{01}}{(RT_{01})} = \rho_1 \qquad (3)$$

$$V_{el1"} = \frac{\dot{m}_{air}}{\rho_1 A} = \frac{4\dot{m}_{air}}{\rho_1 d_1^2 \pi} = V_{el1} \qquad (4)$$

The static inlet temperature, pressure and density are derived by equations (5)-(7).

$$T_1 = T_{01} - \frac{V_{el1}^2}{(2C_P)} \qquad (5)$$

$$p_1 = p_{01}(\frac{T_1}{T_{01}})^{\frac{\gamma}{\gamma-1}} \qquad (6)$$

$$\rho_1 = \frac{p_1}{(RT_1)} \qquad (7)$$

The velocity from the continuity equation at section 1 is determined by equation (8).

$$V_{el1\_new} = \frac{\dot{m}_{air}}{\rho_1 A}.\ (\text{If}\quad V_{el1\_new} = V_{el1} \rightarrow \text{ok, else}\ V_{el1} == V_{el1\_new}\ \text{and go to eq. (5)}) \qquad (8)$$

Due to the changing of static temperature of the intake, the sonic speed is calculated as follows:

$$a_1 = \sqrt{\gamma R T_1} \qquad (9)$$

The local Mach number within the intake duct is:

$$M_1 = \frac{V_{el1}}{a_1} \qquad (10)$$

The phenomenon in the intake can be approached by a constant total temperature process and the total pressure is determined by the pressure recovery factor of the intake:

$$p_{02} = p_{01} rd, \qquad (11)$$

however, the diffuser efficiency η_d can be expressed and calculated from the following equation:

$$p_{02} = p_1(1 + \eta_d \frac{\gamma-1}{2} M^2{}_1)^{\frac{\gamma}{\gamma-1}} \qquad (12)$$

The specific volume of the air at the inlet and outlet section of the intake is given by the ideal gas law.

$$v_0 = \frac{(RT_0)}{p_0} \qquad (13)$$

$$v_{02} = \frac{(RT_{02})}{p_{02}} \qquad (14)$$

The outlet entropy of the intake can be determined by using equation (15):

$$s_{02} = C_v \ln\frac{T_{02}}{T_0} + R\ln\frac{v_{02}}{v_0} + s_0, \tag{15}$$

where s_0 is the reference entropy.

2.2 Compressor (2-3)

The work needed for compression of the air before entering into the burner (combustion chamber) and it is provided by the compressor. The outlet total pressure of the compressor is calculated by the knowledge of the total pressure ratio of the compressor and the outlet total pressure of the intake:

$$p_{03} = \pi_C p_{02} \tag{16}$$

The outlet total temperature of the compressor can be calculated by knowing the isentropic efficiency η_{izC} (see equation (18)). The specific volume of the air in the outlet of the compressor is determined by equation 19.

$$T_{03s} = T_{02}\left(\frac{p_{03}}{p_{02}}\right)^{\frac{\gamma}{\gamma-1}} \tag{17}$$

$$T_{03} = \frac{T_{03s}-T_{02}}{\eta_{izC}} + T_{02} \tag{18}$$

$$v_{03} = \frac{(RT_{03})}{P_{03}} \tag{19}$$

The entropy variation is calculated by equation (20).

$$s_{03} = C_v \ln\frac{T_{03}}{T_{02}} + R\ln\frac{V_{03}}{V_{02}} + s_{02} \tag{20}$$

2.3 Combustion Chamber (3-4)

The stagnation pressure at the outlet of combustion chamber, state (4), is less than its value at the inlet, at state (3), because of real flow assumption. Thus, the outlet pressure of the combustion chamber is expressed as follows:

$$p_{04} = r_{cc} p_{03} \tag{21}$$

The fuel to air ratio is calculated by taking into consideration the efficiency of burners η_b (see equation (22)) in the function of combustion chamber and turbine inlet total temperature. The specific heat (C_p) of the air depends on its temperature [6]. Considering a temperature interval between 679.6 K and 1364 K in the combustion chamber, a temperature dependent mean value of the specific heat was considered as $C_{p\_air\_mean}$ and used in the calculation.

$$f = \frac{C_{p\_air\_mean}(T_{03}-T_{04})}{(1+L_0)(C_{Pgas}T_{04})-(L_0 C_{p\_air\_T04}T_{04})-(Q_R\eta_b)} \tag{22}$$

The $C_{P\_air\_T04}$ is the real specific heat of the air at turbine inlet total temperature and at constant pressure. The mass flow rate of the fuel comes from equation (23).

$$\dot{m}_{fuel} = f\,\dot{m}_{air} \tag{23}$$

The specific volume of the gas mixture at the outlet of the combustion chamber is calculated by (24).

$$v_{04} = \frac{(R_{gas}T_{04})}{p_{04}} \tag{24}$$

The entropy of the burner is determined by equation (25).

$$s_{04} = C_{Vgas}\ln\frac{T_{04}}{T_{03}} + R_{gas}\ln\frac{v_{04}}{v_{03}} + s_{03} \tag{25}$$

2.4 Turbine (4-5)

The gas flows through the turbine that provides shaft power input to the compressor besides the existing mechanical losses and energy content for auxiliary systems. By using the energy balance of the spool, the temperature at the section 5 is as follows:

$$T_{05} = T_{04} - \frac{C_{Pcom}(T_{03}-T_{02})}{(1+f)C_{Pgas}\eta_m\zeta} \tag{26}$$

The isentropic total temperature at the turbine outlet can be determined by (27).

$$T_{05s} = T_{04} - \frac{T_{04}-T_{05}}{\eta_{izT}} \tag{27}$$

The outlet total pressure of the turbine is calculated as follows:

$$p_{05} = \frac{p_{04}}{\left(T_{04}/T_{05s}\right)^{\frac{\gamma_{gas}}{\gamma_{gas-1}}}} \tag{28}$$

The specific volume of the gases at the turbine outlet is given by the ideal gas law:

$$v_{05} = \frac{(R_{gas}T_{05})}{p_{05}} \tag{29}$$

The outlet entropy of the turbine is defined by (30):

$$s_{05} = C_{Vgas}\ln\frac{T_{05}}{T_{04}} + R_{gas}\ln\frac{v_{05}}{v_{04}} + s_{04} \tag{30}$$

2.5 Afterburner (5-6)

The total pressure at the outlet of the afterburner will be less than its value at the inlet due to the pressure losses and it is considered by the given pressure recovery factor (rtd) as the following:

$$p_{06} = p_{05}rtd \tag{31}$$

The total temperature is supposed to be constant in the afterburner duct:

$$T_{06} = T_{05} \tag{32}$$

The specific volume of the gas mixture in the afterburner is then computed by (33).

$$v_{06} = \frac{(R_{gas}T_{06})}{p_{06}} \tag{33}$$

The outlet entropy of the afterburner is determined by (34).

$$s_{06} = C_{vgas} \ln \frac{T_{06}}{T_{05}} + R_{gas} \ln \frac{v_{06}}{v_{05}} + s_{05} \tag{34}$$

2.6 Convergent Exhaust Nozzle (6-9)

The thermodynamic process develops in the convergent nozzle is found in Figure 3. By knowing the inlet conditions of the nozzle (p_{06}, T_{06}) and its isentropic efficiency, it is possible to determine whether the converging nozzle is choked or unchoked. Additionally to this information, one can calculate the pressure, temperature and velocity at the exit of the exhaust nozzle. If the ambient pressure is higher than the critical pressure, the nozzle flow is unchoked, therefore the exhaust pressure of the exit is equal to the ambient pressure. At the other case, when the critical pressure is above the ambient pressure, the nozzle flow is choked, and then the exhaust pressure of the nozzle is equal to the critical pressure p_C.

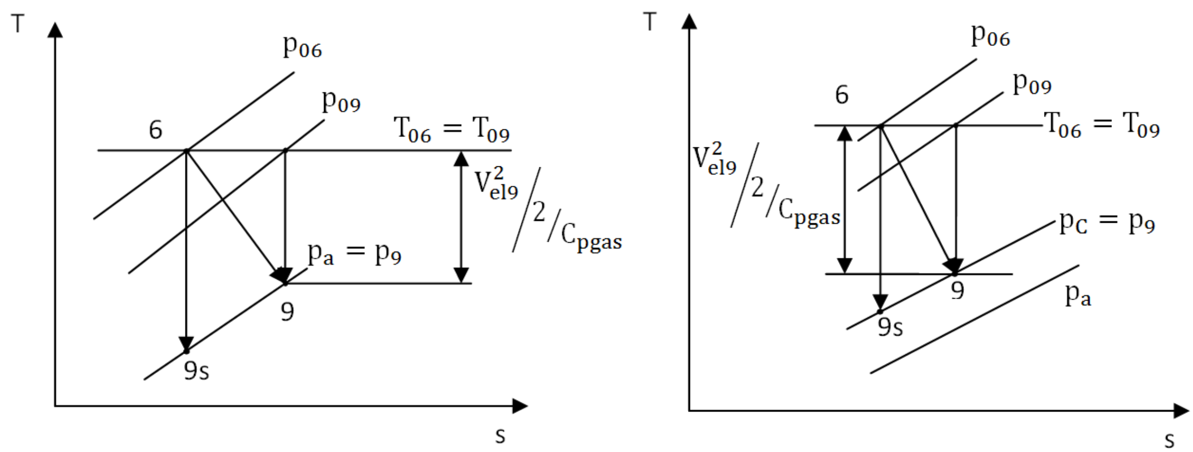

Figure 3. Nozzle flow with losses for unchoked (left hand side) and for choked (right hand side) flow conditions

The critical pressure can be calculated from the relation below (35).

$$p_C = p_{06}\left(1 - \frac{1}{\eta_n}\left(\frac{\gamma_{gas}-1}{\gamma_{gas}+1}\right)\right)^{\frac{\gamma_{gas}}{\gamma_{gas}-1}} \tag{35}$$

If $p_C < p_0$ then the nozzle is unchoked, $p_9 = p_0$, and the exhaust velocity of the nozzle is determined by equation (36).

$$V_{el9=}\sqrt{2C_{Pgas}T_{06}\eta_n(1 - (\frac{p_0}{p_{06}})^{\frac{\gamma_{gas}-1}{\gamma_{gas}}}} \tag{36}$$

In equation (38), the nozzle outlet static temperature is determined by the available nozzle efficiency η_n and isentropic static temperature of the nozzle T_{9s}, which comes from equation (37).

$$T_{9s} = T_{06}\left(\frac{p_0}{p_{06}}\right)^{\frac{\gamma_{gas}-1}{\gamma_{gas}}} \tag{37}$$

$$T_9 = T_{06} - \eta_n(T_{06} - T_{9s}) \tag{38}$$

While the critical pressure is higher than the ambient pressure, then the nozzle is choked. Thus, the outlet static pressure is determined by equation (35). The outlet static temperature is calculated as follows:

$$T_C = T_{06}\left(\frac{2}{(\gamma_{gas}+1)}\right) = T_9 \tag{39}$$

The exhaust velocity of the choked nozzle comes from the following relation:

$$V_{el9} = \sqrt{\gamma_{gas} R_{gas} T_9} \tag{40}$$

The specific volume and entropy is determined by the equations (41) and (42), respectively.

$$v_9 = \frac{(R_{gas} T_9)}{p_9} \tag{41}$$

$$S_9 = C_{vgas} \ln \frac{T_9}{T_{06}} + R \ln \frac{v_9}{V_{06}} + S_{06} \tag{42}$$

3. Engine Performance Parameters

The performance of an aircraft engine may be rated by its ability to provide the necessary thrust force in propelling an aircraft efficiently. The engine performance parameters are identified as:

Thrust

The thrust is expressed in equation (43) in which A_9 is the nozzle outlet area.

$$T = \dot{m}_{air}\big((1+f)V_{el9} - V_{el0}\big) + A_9(p_9 - p_0) \tag{43}$$

Thrust specific fuel consumption

Changing the mass flow rate of air has an effect on the specific fuel consumption (SFC) and thrust. The TSFC is given by (44).

$$TSFC = \frac{\dot{m}_{fuel}}{T} \tag{44}$$

Thermal Efficiency

Thermal efficiency is the ratio of all useful power imparted to engine airflow and the energy contained in the fuel burning:

$$\eta_{th} = \frac{TV_{el0} + 0.5\dot{m}_{air}(1+f)(V_{el9} - V_{el0})^2}{\eta_b \dot{m}_{fuel} Q_R} \tag{45}$$

Propulsion and Burning Efficiency

Propulsive efficiency (denoted by η_{Prop}) is the ratio of the thrust power and the power imparted to engine airflow.

$$\eta_{Prop} = \frac{TV_{el0}}{TV_{el0} + 0.5\dot{m}_{air}(1+f)(V_{el9} - V_{el0})^2} \tag{46}$$

The burning efficiency is denoted by η_b and it can be considered by equation (49).

$$\eta_b = \frac{\eta_b \dot{m}_{fuel} Q_R}{\dot{m}_{fuel} Q_R} \tag{47}$$

Overall Efficiency

The overall efficiency is the ratio of useful power and the energy supplied by the fuel (51) and it is the product of the propulsive, the thermal and the burning efficiency. The useful power is the power available for thrust generation.

$$\eta_O = \eta_b \eta_{th} \eta_{Prop} \tag{48}$$

$$\eta_O = \frac{T V_{el0}}{\dot{m}_{fuel} Q_R} \tag{49}$$

4. Comparison of the Results with Available Data

The aim of this part of the article is to check how close the results of the present method are to the reality. Generally, it is hard to find out any complex information about thermo dynamical characteristics and performance data of a jet engine due to confidentially issues. However, some of them for example about the Tumansky R-29 jet engine are available in the literature [8]. The data used for the simulation are presented in Table 1.

Tumansky R-29 turbojet engine	**aircraft: MiG-23**
Maximal thrust	$T_{max.}$ = 81.4 kN
Maximal turbine inlet temperature	$T_{04,max.}$ = 1091 °C
Maximal turbine outlet temperature	$T_{05,max.}$ = 840 °C
TSFC	TSFC = 0.0968 $kg.N^{-1}.h^{-1}$
Compressor total pressure ratio	π_C = 13
Maximal air mass flow rate	m_{air} = 110 $kg.s^{-1}$
Inlet diameter	d_1 = 846 mm

Table 1: Operational data of Tumansky R-29 turbojet engine [8]

The input data for the analysis is found in Table 2. The data can be divided into four main groups: environmental conditions and reference values, material properties and constants, geometry and operational data of the engine and finally the useful power reduction coefficients.

Environmental conditions and reference values	Material Properties and constants	Geometry and operational data of the engine	Power reduction coefficients
$s_0=1000\frac{J}{kgK}$	$M_{air}=28.97$ kg/kmol	$\pi_C=13$	$\eta_n=0.95$
$p_0=101325$ Pa	$L_0=14.72$	$T_{04}=1364$ K	$\eta_{izT}=0.88$, $\eta_{izC}=0.84$
$T_0=293.15$ K	$\gamma_{gas}=1.33$, $\gamma_{air}=1.4$	$m_{air}=110\frac{Kg}{s}$	$\eta_b=0.93$
$V_{el0}=0\frac{m}{s}$	$R_{gas}=282.8\frac{J}{kgK}$, $R_{air}=282.8\frac{J}{kgK}$	$d_1=0.846$ m	$\eta_m=0.98$
H=0 m	$Q_R=42.8\frac{MJ}{kg}$	$d_9=0.8$ m	$r_{cc}=0.95$
	$R_M=8314\frac{J}{kmolK}$	$A_9=d_9^2\pi/4$	$\zeta=0.92$
	$C_{Pcom}=1030$ J/kg/K $C_{p\_air\_mean}=1141$ J/kg/K $C_{p\_air\_T04}=1196$ J/kg/K		rd=0.98, rtd=0.96

Table 2: Input data of the simulation

Based on the previously mentioned input data, the developed Matlab code calculates the points of the T-s thermo dynamical cycle of the engine, together with thrust, thrust specific fuel consumption and overall efficiency. Considering the available data only the maximum thrust (T),

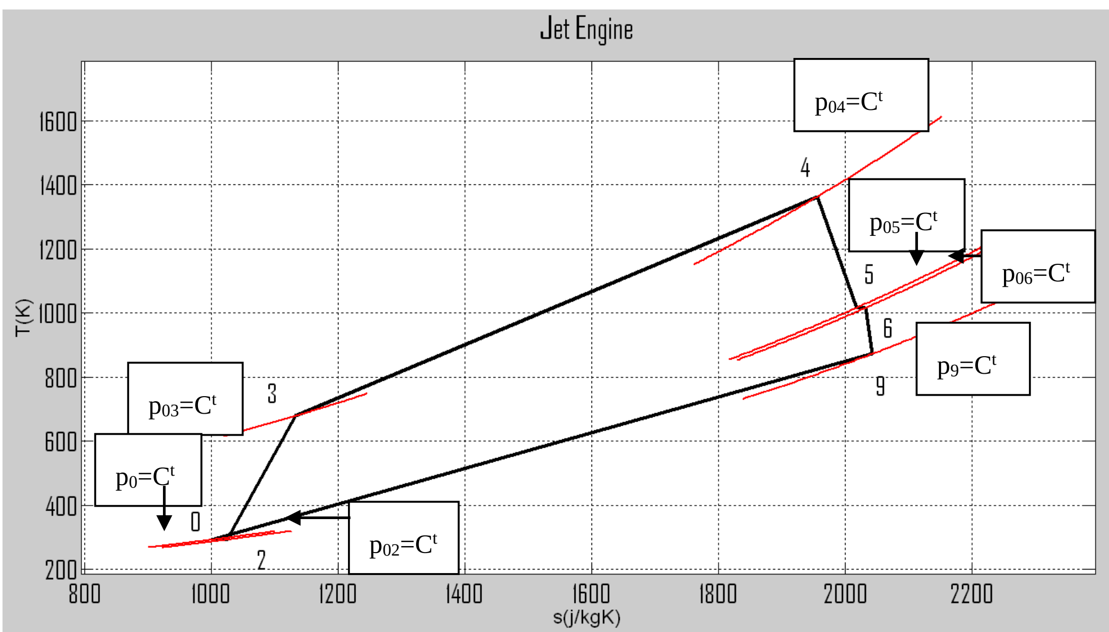

Figure 4. T-s diagram of the Tumansky R-29 turbojet engine

thrust specific fuel consumption (TSFC) and maximum outlet turbine temperature (T_{05}) has been used to check directly how far the calculated results of analysis from the data from the specification. The simulation resulted in a maximum thrust (T) of 85.5 kN, a maximum turbine outlet temperature of 1000 K and a thrust specific fuel consumption (TSFC) of 0.096 kg/N/h. The reference values for this engine are 81.4 kN for maximum thrust, 1113.15 K for the maximum turbine outlet temperature, and 0.0968 kg/N/h for the specific fuel consumption. The relative error for the simulated thrust was 5 %, for the maximum turbine outlet temperature was 10.2 % and for the thrust specific fuel consumption was 0.86%.

The real thermo dynamical cycle of the engine in T-s diagram is plotted in Figure 4.

In order to improve the accuracy of the simulation, the temperature dependence of specific heat should be considered in each segment of the analysis. Moreover, considering the material properties and amounts of each component of the gas mixture, it can also improve the accuracy of the analysis.

Additionally, in order to have more detailed information about the thermo dynamical, fluid dynamics and heat transfer process of the jet engine not only for understanding the physical behavior of different components, but able to design and develop the expected characteristics, more sophisticated calculation method has been implemented and used as the next step of the present research.

CONCLUSIONS

A concentrated parameter distribution type model of the thermo dynamical cycle of a single spool no bypass jet engine has been worked out and implemented in Matlab environment.

Mechanical efficiency, power rate for auxiliary systems, isentropic efficiencies, burning efficiency and pressure losses were considered in the governing equations of the model according to the real thermo dynamical processes.

The input parameters of the gas turbine program are the material properties of the operational fluids, geometrical data of the engine, altitude, flight speed, total pressure ratio of the compressor, total inlet turbine temperature and mass flow rate of the air, meanwhile the points of the thermo dynamical cycle, performances and the efficiencies of the jet engine were determined.

The results of the simulation were compared and checked with the available data in open literature regarding the Tumansky R-29 turbojet engine. Although the simulation results show less than or equal 10.2 percent difference between the simulated and the available real data, more different types of engine and detailed thermo dynamical and numerical analyses would be necessary for testing the theory and the correct operation of the mathematical model.

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