NUMERICAL INVESTIGATION OF TURBOJET ENGINE THRUST CORRELATED WITH THE COMBUSTION CHAMBER'S PARAMETERS

Irina-Carmen ANDREI[*], Constantin ROTARU[**], Maria-Cristina FADGYAS[*], Gabriela STROE[***], Mihai Leonida NICULESCU[*]

[*]National Institute for Aerospace Research "Elie Carafoli" (andrei.irina@incas.ro, niculescu.mihai@incas.ro)
[**]"Henri Coanda" Air Force Academy, Brasov, Romania (rotaru.constantin@yahoo.com)
[***]"POLITEHNICA" University of Bucharest, Faculty of Aerospace Engineering, (ing.Stroe@yaoo.com,)

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Abstract: *The purpose of this paper is to highlight the correlation between the thrust as the main performance of a turbojet engine and the parameters of the combustion chamber, e. g. the pressure losses and the influence of the perturbations in the turbine inlet temperature T3T. The performances of the turbojet engine have been predicted by following the numerical simulation of the engine's operation, with an in-house developed code, based on a comprehensive mathematical model of the turbojet engine. The investigation is carried on at the engine's design point, which means 100% rotational speed, at sea level static standard atmosphere conditions. The most significant parameters of the combustion chamber, namely the combustion chamber's pressure losses and the perturbations in the turbine inlet temperature T3T have been chosen such that to match the standpoint of the fighter pilot; the most relevant parameters is the turbine inlet temperature T3T. The numerical results are summarized as graphs and charts, from which one can express new correlations and further, a new command and control law for the turbojet engine's operation can be concluded. The contributions of this study may prove to have practical applications, for being used both for training the pilot students and during flight operation, for contributing to a significant improvement in flight safety, which can be of real help for fighter pilots.*

Keywords: *turbojet engine, performance prediction, combustion chamber, numerical simulation, engine maps*

1. INTRODUCTION

The parameters of the combustion chamber, from the standpoint of a turbojet engine performances, are the turbine inlet temperature T3T [K], the pressure loss coefficient, compressor exit temperature and the properties of the fuel, which for most of the cases are represented by the fuel specific power P_{ci}; taking into account the following assumptions: 1/ that the combustion chamber is designed such that the pressure losses should be minimized, and therefore the pressure loss coefficient remains quasi constant, its magnitude being around 0.98; 2/ that the fuel specific power is with two orders of magnitude higher than the largest value of the stagnation specific enthalpy at turbine inlet, no matter that the fuel specific power is ranging from 40000 up to 45000 [kJ/kg];

3/ the variation of the compressor exit temperature is not so large and does not influence the turbojet performances, because the variation of the compressor pressure ratio (as shown by the compressor universal map) is limited; therefore, the investigation presented in this paper can be focused on the influence of the variation of the turbine inlet temperature, altitude and Mach number as flight parameters and speed as engine operating regime.

The study case is considered the VIPER 631 turbojet engine. For a large class of applications, the turbojet engine it is a optimized solution, due to the simplicity of construction (single-spool), ease of operating and maintenance and lower costs.

The **objectives** of the analysis of the turbojet engine are represented by modeling and simulation, in purpose to obtain the predicted engine performances, for different flight regimes and various operating regimes. Basically, the TURBOJET ENGINE Model supposes the completion of the steady state analysis, which supposes the calculation of the Brayton cycle at Sea Level Static SLS and ISA conditions, followed by the calculation of turbojet performances (thrust, specific thrust and fuel specific consumption, such that to build the engine's Operation Maps, which are: 1/ the Altitude Map, 2/ the Velocity Map, 3/ the Rotor Speed Map and 4/ the engine's Universal Map.

2. PERFORMANCE ANALYSIS OF TURBOJET ENGINE

2.1 Problem statement and framework

Performance Analysis of Turbojet Engine, also referred as **Thermodynamic Analysis** supposes the completion of three phases; thorough details are given in literature, Mattingly [1], as well as authors other researches, Andrei [8-15, 17-18], Rotaru [21-23] and Prisacariu [16].

The first phase consists in calculating at SLS, ISA conditions (i.e. "*fixed point*", which usually means altitude H = 0 [km] and flight velocity V = 0 [m/s]) of the engine's performances (*Thrust*, *Specific Thrust*, *Specific Fuel Consumption*) and the determination of the engine's thermodynamic cycle (i.e. Brayton cycle).

The second phase consists in calculating the engine's performances at different flight regimes and rotor speed, which usually are expressed by **ENGINE's OPERATING MAPS** (i.e. **ALTITUDE MAP, VELOCITY MAP, SPEED MAP**.

The third phase, which consists in calculating of the **ENGINE's UNIVERSAL MAP**, completes the performance analysis. The results obtained following the performance analysis of the engine allow to study the dynamic behaviour of the engine and to do the numerical simulations.

2.2 Identification of missing thermodynamic engine parameters

The <u>main engine parameters</u> of the VIPER 631 turbojet engine as the study case, must be input data (being given or determined) for the thermodynamic analysis.

<u>**Compressor Pressure Ratio**</u> π_c^* (1) defined as the ratio of stagnation pressures at compressor exit versus inlet.

$$\pi_c^* = \frac{p_2^*}{p_1^*} \tag{1}$$

$$\pi_c^* = 5.9 \tag{2}$$

Turbine inlet temperature T3T (also referred as T_3^*) is determined from the relation (8) expressing the specific work of turbine (3) as the specific enthalpy drop between turbine exit and turbine inlet; the operating law for the single-spool turbojet is (4), with the meaning that the specific work produced by the turbine is used to produce specific work on compression (5);

Compressor pressure ratio is defined as the ratio of stagnation pressures at compressor exit versus inlet (4); specific enthalpy is proportional with the temperature and constant pressure specific heat *Cp* , with different values for air (6.1) and mixture of burned gas (6.2); expressing the turbine specific work as a function (7) of the stagnation temperatures at turbine inlet T_3^* and turbine exit T_4^*, then relation (8) is deduced :

$$l_T^* = i_3^* - i_4^* \tag{3}$$
$$l_T^* = l_c^* \tag{4}$$
$$l_c^* = i_2^* - i_1^* = \frac{i_1^* \cdot \left((\pi_c^*)^{\frac{k-1}{k}} - 1\right)}{\eta_c^*} \tag{5}$$
$$C_p = 1.005 \left[\frac{kJ}{kgK}\right] \tag{6.1}$$
$$C_{pg} = 1.165 \left[\frac{kJ}{kgK}\right] \tag{6.2}$$
$$l_T^* = i_3^* - i_4^* = C_{pg} \cdot (T_3^* - T_4^*) \tag{7}$$
$$T_3^* = T_4^* - \frac{l_T^*}{C_{pg}} \tag{8}$$
$$T_3^* = 1280\ [K] \tag{9}$$

Note that for the specified operating regimes one can deduce also the **T3T operating control law**, expressing its variation with the % of the rotor speed regimes [%rpm]; next, the automatic controls of the engine/ fuel systems can be designed. E.g. Max starting = 105, Nominal -100, Cruising = 90, Cruising lowered = 85, Idle ground = 40.

Airflow rate $\dot{M}_a$ [kg/s] is determined after a performing a number of iterations for the Brayton cycle, with the consequent calculation of the turbojet engine performances (i.e. specific thrust F_{sp} $\left[\frac{Ns}{kg}\right]$, thrust F $[N]$ and specific fuel consumption C_{sp} $\left[\frac{kg}{Nh}\right]$),

$$\dot{M}_a = 26.332 \left[\frac{kg}{s}\right] \tag{10}$$

Other engine parameters which must be determined before performing the thermodynamic analysis are listed downwards. These values have been trimmed such that to match the given thrust and specific fuel flow.

- **adiabatic efficiency on compression** η_c^* (11) defined as the ratio of specific work on compression and ideal specific work . From experience, centrifugal compressors have slightly lower values of adiabatic efficiencies with respect to axial flow compressor. In this case $\eta_c^* = 0.85$.

$$\eta_c^* = \frac{l_{c_{id}}^*}{l_c^*} \tag{11}$$

- **adiabatic efficiency on turbine expansion** η_t^* (12) defined as the ratio of ideal specific work of turbine and its specific work. In this case $\eta_t^* = 0.89$.

$$\eta_t^* = \frac{l_t^*}{l_{t_{id}}^*} \tag{12}$$

- **mechanical (shaft) efficiency** η_m (13) defined as the ratio of specific work consumed by compressor and specific work produced by turbine; in case of a single spool turbojet engine, $\eta_m = 1$. In, since there are no mechanical losses between compressor and turbine.

$$\eta_m = \frac{l_c^*}{l_t^*} \tag{13}$$

- **pressure loss at engine intake** σ_{da}^* (14) defined as the ratio of the stagnation pressures at intake exit versus inlet; $\sigma_{da}^* = 0.92$.

$$\sigma_{da}^* = \frac{p_1^*}{p_H^*} \tag{14}$$

- **pressure loss in combustor** σ_{ca}^* (15) defined as the ratio of the stagnation pressures at compressor exit versus combustor exit; $\sigma_{ca}^* = 0.98$.

$$\sigma_{ca}^* = \frac{p_3^*}{p_2^*} \tag{15}$$

- **combustion efficiency** $\xi_{ca} = 0.998$
- **exhaust nozzle velocity** $\varphi_{ar} = 0.940$

2.3 Purpose of calculations

a) the determination of **turbojet engines performances** (i.e. specific thrust F_{sp} $\left[\frac{Ns}{kg}\right]$ (47), thrust F $[N]$ (48), specific fuel consumption C_{sp} $\left[\frac{kg}{Nh}\right]$ (49), for all flight envelope and engine operating regimes;

b) the influence of altitude, flight Mach number and rotor speed [rpm] on inlet air flow rate (50) and compressor pressure ratio (55) - (58), was taken into account when calculating the turbojet engine performances, when operating at altitude, at specified flight velocity and engine regime [%rpm];

3. MATHEMATICAL MODEL

3.1 Hypothesis

The mathematical model of a turbojet engine describing its behavior as close to reality is based on the following **HYPOTHESIS**:

- the working fluid is considered perfect gas,
- two species:

A. // **air** // - from intake to compressor,

B. // **burned gas** // - within combustor, turbine and exhaust unit,

- fuel specific power, for JET A, JET A1 and/or JET B (aviation kerosene): $P_{CI} = 43500 \left[\frac{kJ}{kg}\right]$,
- ratio of specific heat $k = \frac{C_p}{C_v}$, see Table 1.
- constant pressure specific heat C_p $\left[\frac{kJ}{kgK}\right]$;
- gas constant R $\left[\frac{kJ}{kgK}\right]$; the relation between R and C_p is (16):

$$C_p = R \cdot \frac{k}{k-1} \tag{16}$$

Table 1 - Properties of the working fluids

Fluid	k	C_p [kJ/kg/K]	R [J/kg/K]
Air	1.4	1.005	287.3
Burned Gas	1.33	1.165	288.4

3.2. Basic equations

Basic equations (17) - (46) for computing the **turbojet engine performances** (algorithm defined by equations / relations ordered as entries in work flow):

- SLS, ISA conditions: $p_0 = 1.01325\ [bar]$ (17.1), $T_0 = 288\ [K]$ (17.2) and

$$i_0 = C_p \cdot T_0 \quad \left[\frac{kJ}{kg}\right] \tag{18}$$

- conditions at engine inlet (intake) - station 0 (SLS) or H (flight) :
- *if* H = 0 [km] *then* $p_1^* = \sigma_{da}^* \cdot p_0\ [bar]$, (19.1), $T_1^* = T_0\ [K]$ (19.2), and

$$i_1^* = C_p \cdot T_1^* \quad \left[\frac{kJ}{kg}\right] \tag{20}$$

if H > 0 *then* $p_1^* = p_H^* \cdot p_0\ [bar]$ (21.1), $T_1^* = T_H^*\ [K]$ (21.2) and

$$i_1^* = C_p \cdot T_1^* \quad \left[\frac{kJ}{kg}\right] \tag{22}$$

where $T_H = T_0 - 6.5 \cdot H\ [km], \qquad [K]$ (23)

and $p_H = p_0 \cdot \left(\frac{T_H}{T_0}\right)^{5.2553}$ (24)

and $T_H^* = T_H + \frac{V^2}{2 \cdot C_p}$ (25)

or $T_H^* = T_H \cdot \left(1 + \frac{(k-1)}{2} \cdot Mach^2\right)$ (26)

and $p_H^* = p_H \cdot \left(1 + \frac{(k-1)}{2} \cdot Mach^2\right)^{\left(\frac{k-1}{k}\right)}$ (27)

- dynamic pressure ratio:

$$\pi_d^* = \frac{p_H^*}{p_H} = \left(\frac{T_H^*}{T_H}\right)^{\left(\frac{k-1}{k}\right)} = \left(1 + \frac{(k-1)}{2} \cdot Mach^2\right)^{\left(\frac{k-1}{k}\right)} = (\Theta(Mach))^{\left(\frac{k-1}{k}\right)} \tag{28}$$

- conditions at compressor inlet - station 1^*: $p_1^* = \sigma_{da}^* \cdot p_H^*$ (29.1), $T_1^* = T_H^*$ (29.2), $i_1^* = i_H^*$ (30)
- conditions at combustor inlet - station 2^*: $p_2^* = \pi_c^* \cdot p_1^*$ (31)

$$i_2^* = i_1^* \cdot \left(1 + \frac{\left((\pi_c^*)^{\frac{k-1}{k}} - 1\right)}{\eta_c^*}\right) \tag{32}$$

$$T_2^* = \frac{i_2^*}{C_p} \tag{33}$$

- conditions at turbine inlet - station 3^* : $p_3^* = \sigma_{ca}^* \cdot p_2^*$ (34)

• T_3^* from equation (8), $i_3^* = C_{pg} \cdot T_3^*$ (35)

- fuel flow coefficient (from energy balance eqn. in combustor):

$$m_c = \frac{(i_3^* - i_2^*)}{(\xi_{ca} \cdot P_{ci} - i_3^*)} \tag{36}$$

- burned gas flow coefficient (from mass balance eqn. in combustor): $m_g = 1 + m_c$ (37)
- fuel flow coefficient: $m_c = \frac{\dot{M}_c}{\dot{M}_a}$ (38)
- burned gas flow coefficient: $m_g = \frac{\dot{M}_g}{\dot{M}_a}$ (39)
- conditions at turbine exit - station 4^*: $i_4^* = i_3^* - l_t^*$ (40)

$$T_4^* = \frac{i_4^*}{C_{pg}} \quad (41)$$

$$p_4^* = \frac{\delta_t^*}{p_3^*} \quad (42)$$

- where δ_t^* is the pressure ratio in turbine, and it comes out from the expression of specific work in turbine. $\delta_t^* = \left(1 - \frac{l_{t_{id}}^*}{i_3^*}\right)^{-\left(\frac{kg}{kg-1}\right)}$ (43)
- conditions at nozzle exit - station 5 :

case: full exhaust nozzle expansion: $p_5 = p_H$ (44.1) , then the thrust obtained is maximum

case: partial exhaust nozzle expansion: $p_5 = p_{cr} < p_H$ (44.2),

$$p_{cr} = \left(\frac{2}{kg+1}\right)^{\left(\frac{kg}{kg-1}\right)} \cdot p_4^* \quad (45)$$

- velocity of expelled gas c_5 [m/s], (46):

$$c_5 = \varphi_{ar} \cdot \sqrt{2 \cdot \left\{ \left[i_3^* \cdot \left(1 - \pi_d^* \cdot \sigma_{da}^* \cdot \pi_c^* \cdot \sigma_{ca}^*\right)^{-\left(\frac{kg-1}{kg}\right)} \right] - \atop -i_1^* \cdot \left[\frac{(\pi_c^*)^{\frac{k-1}{k}} - 1}{\eta_c^* \cdot \eta_t^* \cdot \eta_m} \right] \right\}} \quad (46)$$

3.3. Definitions of turbojet engine performances

Relations (47) - (49) define the **turbojet engine performances** (i.e. thrust F [N],, specific thrust F_{sp} [Ns/kg] and specific fuel consumption C_{sp} [kg/Nh] or **TFC**,

- **specific thrust** $F_{sp} = m_g \cdot c_5 - V, \left[\frac{Ns}{kg}\right]$ (47)
- **thrust** $F = F_{sp} \cdot \dot{M}_a, [N]$ (48)
- **specific fuel consumption** $C_{sp} = \frac{3600 \cdot m_c}{F_{sp}}, \quad \left[\frac{kg}{Nh}\right]$ (49)

3.4. The influence of altitude, flight mach number and rotor speed on airflow rate and compressor pressure ratio

Equations (50) - (58) express the **influence of altitude, flight Mach number** and **rotor speed** on inlet **air flow rate** (50) and **compressor pressure ratio** (55) - (58):

- Airflow rate (50) is influenced by the change of altitude and flight Mach number, by the means of the variation of compressor pressure ratio, dynamic pressure ratio and the ratio of static pressures al altitude H [km] versus SLS:

$$\dot{M}_a = \dot{M}_{a_0} \cdot \frac{\pi_c^*}{\pi_{c_0}^*} \cdot \pi_d^* \cdot \frac{p_H}{p_0} \quad (50)$$

- specific work on compression (51) changes with the square of rotor speed (52)

$$l_c^* = l_{c_0}^* \cdot \bar{n}^2 \tag{51}$$

- rotor speed % (52) represents the ratio of speeds at operating versus nominal engine regime:

$$\bar{n} = \frac{n}{n_{NOMinal}} \tag{52}$$

- the relations between specific work of compressor, compressor pressure ratio, intake enthalpy and rotor speed, are (53) for SLS, ISA conditions and (54) for the flight at altitude:

$$l_{c_0}^* = i_0 \cdot \left(\frac{(\pi_{c_0}^*)^{\frac{k-1}{k}} - 1}{\eta_{c_0}^*} \right) \tag{53}$$

$$l_c^* = i_H^* \cdot \left(1 + \frac{\left((\pi_c^*)^{\frac{k-1}{k}} - 1 \right)}{\eta_c^*} \right) \tag{54}$$

- the influence of altitude, flight Mach number and rotor speed on compressor pressure ratio (55) - (58) is deduced from relations (53), (54) and (51); the ratio of compressor efficiencies at operating regime versus nominal can be taken about 1.0 (as initial approximation or in case that the universal compressor map is not available):

$$\pi_c^* = \left[1 + \left((\pi_{c_0}^*)^{\frac{k-1}{k}} - 1 \right) \cdot \frac{i_0}{i_H^*} \cdot \bar{n}^2 \cdot \frac{\eta_c^*}{\eta_{c_0}^*} \right]^{\left(\frac{k}{k-1}\right)} \tag{55}$$

$$\pi_c^* = \left[1 + \left((\pi_{c_0}^*)^{\frac{k-1}{k}} - 1 \right) \cdot \frac{i_0}{i_H^*} \cdot \bar{n}^2 \right]^{\left(\frac{k}{k-1}\right)} \tag{56}$$

$$\pi_c^* = \left[1 + \left((\pi_{c_0}^*)^{\frac{k-1}{k}} - 1 \right) \cdot \frac{i_0}{i_H^*} \cdot \bar{n}^2 \right]^{\left(\frac{k}{k-1}\right)} \tag{57}$$

$$\pi_c^* = \left[1 + \frac{l_{c_{id}}}{i_H^*} \cdot \bar{n}^2 \right]^{\left(\frac{k}{k-1}\right)} \tag{58}$$

3.5. The turbojet engine's operating maps

Definitions of turbojet engine's OPERATING MAPS, (i.e. the variation of the jet engine performances: thrust F [N], specific thrust F_{sp} [Ns/kg] and specific fuel consumption C_{sp} [kg/Nh] or TSFC, with altitude, flight velocity and engine rotational regime)

ENGINE OPERATING MAPS are represented by: 1/ **ALTITUDE MAP**, 2/ **VELOCITY MAP** and 3/ **SPEED MAP**

(1) - ALTITUDE MAP is defined as the variation of the jet engine's performances (i.e. thrust F [N],, specific thrust F_{sp} [Ns/kg] and specific fuel consumption C_{sp} [kg/Nh] or TSFC) with respect to altitude H [km], while the flight velocity and rotor speed [%rpm] are constant, their values being usually taken for SLS, ISA conditions

Thrust	$F = f(H)\|_{V=0,\ n\%=1}$
Specific thrust	$F_{sp} = f(H)\|_{V=0,\ n\%=1}$
Specific fuel consumption	$C_{sp} = f(H)\|_{V=0,\ n\%=1}$
Altitude Map	

<u>(2) - VELOCITY MAP</u> is defined as the variation of the jet engine's performances (i.e. thrust F [N],, specific thrust F_{sp} [Ns/kg] and specific fuel consumption C_{sp} [kg/Nh] or TSFC) with respect to flight velocity $\boldsymbol{V}$ [m/s] or its equivalent, flight Mach number, while the altitude H [km] and rotor speed [%rpm] are constant, their values being usually taken for SLS, ISA conditions

Thrust	$F = f(V)\|_{H=0,\ n\%=1}$	**Thrust**	$F = f(M)\|_{H=0,\ n\%=1}$
Specific thrust	$F_{sp} = f(V)\|_{H=0,\ n\%=1}$	**Specific thrust**	$F_{sp} = f(M)\|_{H=0,\ n\%=1}$
Specific fuel consumption	$C_{sp} = f(V)\|_{H=0,\ n\%=1}$	**Specific fuel consumption**	$C_{sp} = f(M)\|_{H=0,\ n\%=1}$
Velocity Map		**Velocity Map**	

<u>(3) - SPEED MAP</u> is defined as the variation of the jet engine's performances (i.e. thrust F [N],, specific thrust F_{sp} [Ns/kg] and specific fuel consumption C_{sp} [kg/Nh] or TSFC) with respect to rotor speed [rpm] or its equivalent engine operational regime [%rpm], while the altitude $\boldsymbol{H}$ [km] and flight velocity $\boldsymbol{V}$ [m/s] are constant, their values being usually taken for SLS, ISA conditions

Thrust	$F = f(V)\|_{H=0,\ n\%=1}$	**Thrust**	$F = f(M)\|_{H=0,\ n\%=1}$
Specific thrust	$F_{sp} = f(V)\|_{H=0,\ n\%=1}$	**Specific thrust**	$F_{sp} = f(M)\|_{H=0,\ n\%=1}$
Specific fuel consumption	$C_{sp} = f(V)\|_{H=0,\ n\%=1}$	**Specific fuel consumption**	$C_{sp} = f(M)\|_{H=0,\ n\%=1}$
Speed Map		**Speed Map**	

3.6. The turbojet engine's universal map

<u>Definitions of UNIVERSAL ENGINE MAP</u>, (i.e. the variation of non-dimensional parameters of thrust F [N],, specific thrust F_{sp} [Ns/kg] and specific fuel consumption C_{sp} [kg/Nh] or TSFC, in case of a jet engine)

<u>UNIVERSAL ENGINE MAPS</u> are represented in coordinates **THRUST PARAMETER** and **SPECIFIC FUEL CONSUMPTION PARAMETER** versus either **flight Mach number** or **SPEED PARAMETER**. The **<u>UNIVERSAL ENGINE MAP</u>** is defined in two equivalent ways, expressing the variation of Thrust parameter and Specific fuel consumption parameter with respect to Mach number, for a constant Speed parameter, or vice-versa:

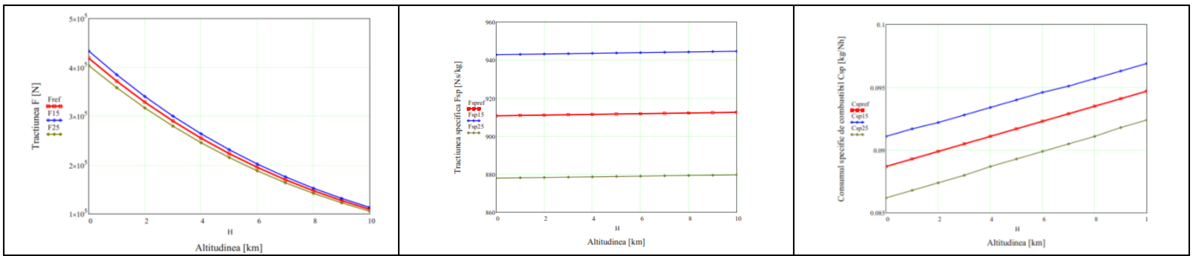
UNIVERSAL ENGINE MAP - definition # **1**	
$\frac{F}{p_1^*} = f(M)\Big\|_{\frac{n}{\sqrt{T_1^*}}=const.}$	Thrust parameter
$\frac{C_{sp}}{\sqrt{T_1^*}} = f(M)\Big\|_{\frac{n}{\sqrt{T_1^*}}=const.}$	Specific fuel consumption parameter
$\frac{n}{\sqrt{T_1^*}} = constant$	Speed parameter

UNIVERSAL ENGINE MAP - definition # **2**	
$\frac{F}{p_1^*} = f\left(\frac{n}{\sqrt{T_1^*}}\right)\Big\|_{M=const.}$	Thrust parameter
$\frac{C_{sp}}{\sqrt{T_1^*}} = f\left(\frac{n}{\sqrt{T_1^*}}\right)\Big\|_{M=const.}$	Specific fuel consumption parameter
$M = constant$	Flight Mach number

Universal Map

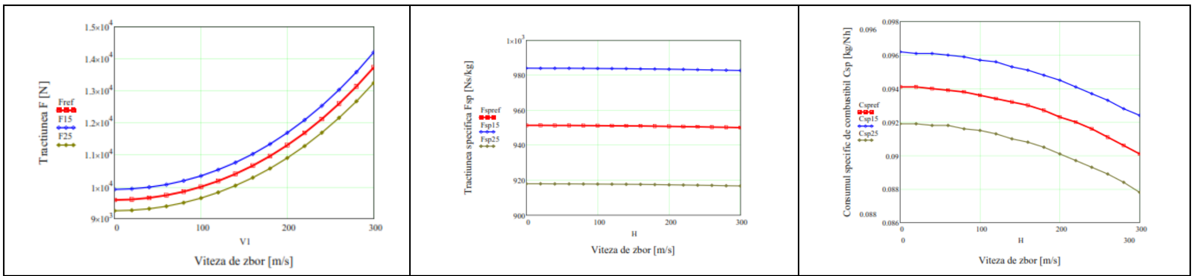
4. RESULTS

This investigation is focused on highlighting the influence of the turbine inlet temperature, as the most significant parameters of the combustion chamber, upon the turbojet engine's performances. With respect to the engine's design point parameters, i.e. the turbine inlet temperature being 1280 [K], it was considered a ranging interval from 1230 [K] up to 1330 [K], which corresponds to a perturbation of temperature with 50 [K], up and down with respect to the reference value.



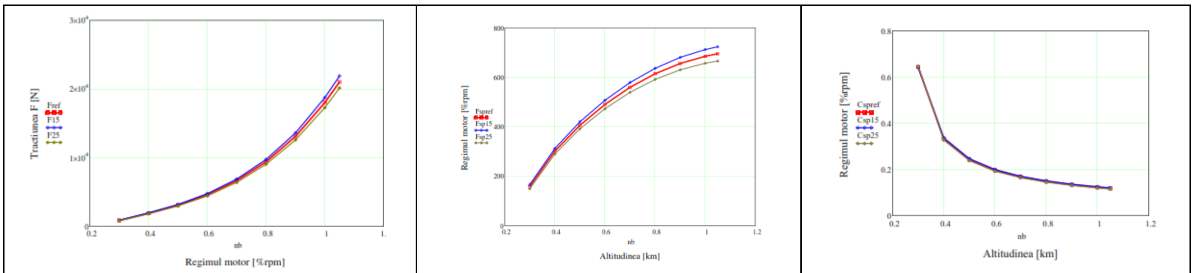
a)- Thrust [N] | b)- Specific Thrust [Ns/kg] | c)- Fuel Specific Cons. [kg/Nh]

FIG. 1 - Engine Performances from Altitude Map



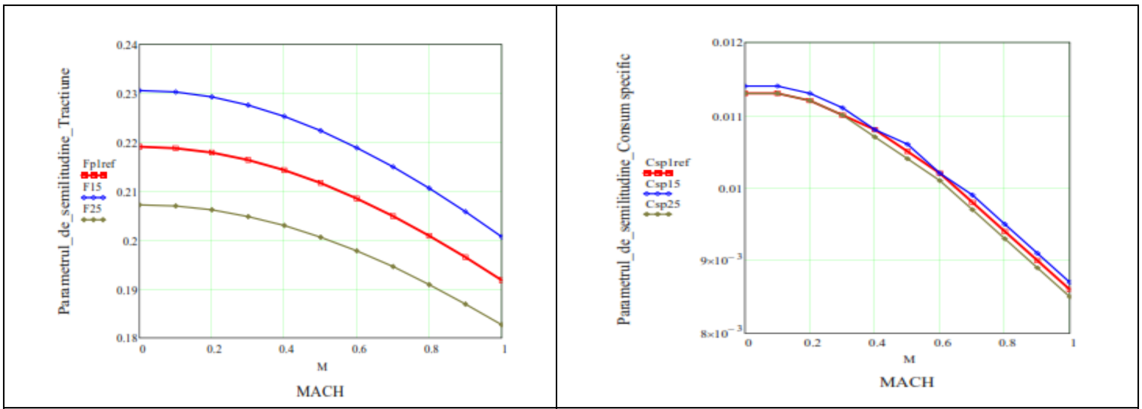
a)- Thrust [N] | b)- Specific Thrust [Ns/kg] | c)- Fuel Specific Cons. [kg/Nh]

FIG. 2 - Engine Performances from Velocity Map



a)- Thrust [N] | b)- Specific Thrust [Ns/kg] | c)- Fuel Specific Cons. [kg/Nh]

FIG. 3 - Engine Performances from Speed Map



a)- Thrust [N]	b)- Fuel Specific Consumption [kg/Nh]

FIG. 4 - Engine Performances from Engine's Universal Map

5. CONCLUSIONS & ACKNOWLEDGMENT

The investigation presented in this paper is focused on the influence of the variation of the turbine inlet temperature, as perturbation of the engine's design parameter and altitude Mach number and speed as perturbations of the flight and engine operating regimes.

The objective of this study is to highlight the variation of engine's thrust for a given perturbation of the turbine inlet temperature. The turbine inlet temperature ranges between 1230 and 1330 [K], which corresponds to a perturbation of +/- 50 [K] with respect to the reference value, that is considered at the engine's design point.

The predicted performances of the turbojet engine (i.e. thrust, specific thrust and fuel specific consumption) are summarized in Fig. 1 ÷ Fig. 4.

Taking into account the variation of thrust (case a)- in Fig. 1 ÷ Fig. 3), one can note that the effect of the perturbation of turbine inlet temperature T3T on thrust is higher when the flight velocity changes Fig. 2-a; unlike this case, the cases when the flight altitude or engine speed change, the influence on thrust of the perturbations of T3T can be neglected, Fig. 1-a and Fig. 3-a. From the Universal Map, the thrust parameter is much more influenced by the perturbation of the turbine inlet temperature T3T.

Future developments of this study can aim to establish other correlations and further, a new command and control law for the turbojet engine's operation that might be concluded. The contributions of this study may prove to have practical applications, for being used both for training the pilot students and during flight operation, for contributing to a significant improvement in flight safety, which can be of real help for fighter pilots.

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