

## STUDY OF THE TURBOJET ENGINES AS PROPULSION SYSTEMS FOR THE UNMANNED AERIAL VEHICLES

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DOI: 10.19062/2247-3173.2016.18.1.15

**Abstract:** *The use of jet engines as propulsion systems for Unmanned Aerial Vehicles UAV's represents a actual challenge. For a large class of UAV's, providing more advantages with respect to the turboprop and turboshaft engines, as well as super-charged V12 piston engines, the small to medium size turbojet engines represent a more effective option. The research presented in this paper is focused on the thermo-dynamical analysis of a small sized turbojet. The single-spool turbojet engine model was used for the performance analysis; the one-dimensional engine cycle design was used to calculate the turbojet performances for a specified range of flight altitude and Mach number, as well as various engine operational regimes. From the numerical simulations have been expressed new parameter correlations, e.g. thrust - fuel flow, which will contribute to the deduction of a control law for the fuel flow. The objectives of the thermo-dynamical analysis have been fulfilled, with the calculated engine operating maps (i.e. variation of performances with altitude, flight Mach number and engine rotational speed), universal map of the engine; in addition, this analysis provides the values for significant engine parameters which are required by the engine's dynamic study.*

**Keywords:** *Unmanned Aerial Vehicle UAV, jet engines, turbojet, modeling, performances, engine maps*

### 1. INTRODUCTION

There is a large class of UAV and RPV [6] - [9] and a continuous demand for their use in both civil and military applications. Their main design goal is to satisfy the conditions imposed by the profile mission, in conjunction with other criteria such as simple construction and lower costs for manufacturing, operating and maintenance.

- **Why the turbojet engine?**

The turbojet engine it is a convenient solution, due to the simplicity of construction (single-spool), ease of operating and maintenance and lower costs.

Taking into account the vehicle flight mission profile, the thrust provided by the engine should range a good match. Thrust augmentation if necessary for the same family type of engine, which may be required by the demand of increasing the weight of the vehicle, can be obtained by increasing the turbine (stagnation) inlet temperature  $T_{3T}$ , as well as the increasing of the air flow rate and pressure ratio. Nevertheless, in case of the same family of engines (with the preservation of the same outer diameter, which means

the air flow rate remains unchanged) remain two options, that is to have higher T3T and pressure ratios. In case of a turbojet equipped with centrifugal compressor, the largest and convenient value for the pressure ratio is about 5, Babak [21]; centrifugal compressors with pressure ratios slightly higher than 6, (as it is the case of the TURMO IV C turboshaft engine), develop embedded supersonic areas on the impeller and rotor, with the consequent occurrence of shock waves, followed by separation and/or re-attachment of the boundary layer, all of these being potential sources for energy and pressure losses. In case of the axial compressor, still keeping the subsonic flow within the axial stage, the pressure ratio can be increased with the addition of axial stages. As proven in practice, [4]-[5], due to safety reasons, axial compressors designed for single-spool turbojet engine constructions are efficient (i.e. operate with lower loss levels) if the pressure ratio is higher than 6 but less than 12 up to 15. Twin spool constructions are used if pressure ratios range from 15 up to 25, since these do operate farther from the surge margin; twin spool constructions can be found at turbojet, turbofan and turboprop engines. For large pressure ratios (over 30-35) the triple spool constructions are more efficient

For UAV's and RPV's, the most convenient from the economy standpoint is to select a small or small to medium sized turbojet, propulsion system "on-the-shelf", which means already designed, build and manufactured. A small or small to medium sized turbojet is usually a single-spool construction, with centrifugal compressor and (one or two staged) axial turbine. A potential increase of thrust, engaging as few changes as possible and reduced costs, is the increasing of the turbine inlet temperature (e.g. with about 100 up to 300 degrees, generating the boost of thrust with about 300 [N]), while the values of airflow rate and pressure ratio are maintained unchanged, which from the construction standpoint mean the maintaining of the same cross section diameter and the same compressor. The increasing of turbine inlet temperature requires less modifications on the combustor level. The selection and/ or the purchasing of an engine built and manufactured in Europe, represents a better option to support the development of EU industry.

- **Which are the objectives of the analysis?**

1. The objectives of the analysis of the turbojet engine as a recommended solution, are the modeling and simulation;

2. The purpose of **modeling and simulation** of a jet engine (thus the turbojet being included) is to do the steady state analysis and further the transient analysis.

3. The methods for analyzing the performances of the turbojet engine and its dynamic control are grouped so that A.1// - *Steady state analysis* and A.2// - *Transient analysis* will be described and explored;

- A. The **steady state analysis** consists in the investigation of the operating, equilibrium states, thus the performances of the jet engine being determined.

- B. The **transient analysis** reveals the dynamic behavior of a jet engine and allows engine's control; the design and management of the aircraft engine controls are based on the results of transient analysis.

As principle, the construction of a TURBOJET ENGINE Model supposes the completion of the steady state analysis and the transient analysis.

The steps for completing the performance analysis (i.e. steady state analysis) /A/ are:

- A.1 - computation of the Brayton cycle at Sea Level Static SLS and ISA conditions;

- A.2 - calculation of turbojet performances, such that to build the engine's Operation

Maps:

- A.2.1 - Altitude Map,

A.2.2 - Velocity Map,

A.2.3 - Rotor Speed Map,

A.3 - calculation of engine's Universal Map.

This paper is focused on the turbojet performance analysis, with the application to the J85 single spool turbojet engine, as Test Case.

The steps for completing the dynamic analysis /B/ are:

- selecting the state variables and relevant first order differential equations
- selecting the output equations
- investigation of the operating states
- perform Taylor linearization to obtain linear models
- combine individual linear models (corresponding to the main parts of the engine) in order to form a composite model describing the turbojet engine operation all over the whole operating area
- constructing a simulation model (e.g. in Simulink ambient)
- constructing a design model
- flight path cycle analysis and further optimizations

For a applications such as experimental aircrafts, powered gliders and UAV, UCAV systems, RPV, the use of turbojet engines as propulsion systems is an increasingly prevalent option; its convenience is due to the lower costs in purchasing, operation and maintenance, as well as inter-changeable features. In most cases, small sized turbojet engines are used, which are characterized by simplicity in construction, such as single-shaft engine with single-stage axial or radial compressor, annular combustion chamber, single-stage axial turbine, and stationary exhaust jet. In the compressor intake there is a brushless starter generator, which enables the starting from the board network and power generating in course of the engine operation. In this paper will be presented the first part, that is the steady-state analysis, providing the turbojet engine's performance analysis.

## 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.

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.

Since not all the engine's design parameters are given, it comes out that at first hand, one must identify the missing parameters. In case of the turbojet engine, this is done by repeating the calculation of the Brayton cycle, until there is obtained a match with a specified parameter, usually the turbojet engine's thrust.

## 2.2 IDENTIFICATION OF MISSING THERMODYNAMIC ENGINE PARAMETERS

The main engine parameters which must be known (being given or determined) before performing the thermodynamic analysis are given below; J85 turbojet engine is the Test Case.

**Compressor Pressure Ratio**  $\pi_c^*$  (1) defined as the ratio of stagnation pressures at compressor exit versus inlet.

$$\pi_c^* = \frac{P_2^*}{P_1^*} \quad (1)$$

$$\pi_c^* = 8.3 \quad (2)$$

**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) (the bleed and auxiliary flow rates can be neglected, as less than 2% of engine air flow rate); 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  $C_p$ , 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 :

$$l_T^* = i_3^* - i_4^* \quad (3)$$

$$l_T^* = l_c^* \quad (4)$$

$$l_c^* = i_2^* - i_1^* = i_1^* \cdot \frac{(\pi_c^*)^{\frac{k-1}{k}} - 1}{\eta_c^*} \quad (5)$$

$$C_p = 1.005 \left[ \frac{kJ}{kgK} \right] \quad (6.1)$$

$$C_{pg} = 1.165 \left[ \frac{kJ}{kgK} \right] \quad (6.2)$$

$$l_T^* = i_3^* - i_4^* = C_{pg} \cdot (T_3^* - T_4^*) \quad (7)$$

$$T_3^* = T_4^* - \frac{l_T^*}{C_{pg}} \quad (8)$$

$$T_3^* = 1250 [K] \quad (9)$$

**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 = 100, Nominal = 97, Cruising = 91, Cruising lowered = 84, Idle ground = 50.

**Airflow rate**  $\dot{M}_\alpha$  [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  $C_{sp}$   $\left[ \frac{kg}{Nh} \right]$ ),

$$\dot{M}_\alpha = 20.43 \left[ \frac{kg}{s} \right] \quad (10)$$

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**  $\eta_c^*$  (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  $\eta_c^* = 0.85$  .

$$\eta_c^* = \frac{l_{c, id}^*}{l_c^*} \quad (11)$$

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

$$\eta_t^* = \frac{l_t^*}{l_{t, id}^*} \quad (12)$$

- **mechanical (shaft) efficiency**  $\eta_m$  (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.

$$\eta_m = \frac{l_c^*}{l_t^*} \quad (13)$$

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

$$\sigma_{da}^* = \frac{p_1^*}{p_H^*} \quad (14)$$

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

$$\sigma_{ca}^* = \frac{p_3^*}{p_2^*} \quad (15)$$

- **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];
- c) in addition, **Brayton cycle** (the variation of specific enthalpy [kJ/kg] versus specific entropy [kJ/kgK], at SLS, ISA conditions, Fig. 1 ;

### 2.4 THE BRAYTON CYCLE AND ENGINE PERFORMANCES

For the turbojet engine operating at **Sea Level Static SLS** and **International Standard Atmosphere ISA conditions** (i.e altitude  $H = 0$  [km] and flight velocity  $V = 0$  [m/s]), is calculated the Brayton cycle, Fig. 1.

### 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):

$$P_{CI} = 43500 \left[ \frac{\text{kJ}}{\text{kg}} \right],$$

- ratio of specific heat  $k = \frac{C_p}{C_v}$ , see Table 1.
- constant pressure specific heat  $C_p \left[ \frac{\text{kJ}}{\text{kgK}} \right]$ :
- gas constant  $R \left[ \frac{\text{kJ}}{\text{kgK}} \right]$ ; the relation between  $R$  and  $C_p$  is (16):

$$C_p = R \cdot \frac{k}{k-1} \quad (16)$$

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:  $p_0 = 1.01325 \text{ [bar]}$  (17.1),  $T_0 = 288 \text{ [K]}$  (17.2) and  $i_0 = C_p \cdot T_0 \text{ [kJ/kg]}$ , (18)

- conditions at engine inlet (intake) - station 0 (SLS) or H (flight) :

- if  $H = 0 \text{ [km]}$  then  $p_1^* = \sigma_{da}^* \cdot p_0 \text{ [bar]}$ , (19.1),  $T_1^* = T_0 \text{ [K]}$  (19.2), and  $i_1^* = C_p \cdot T_1^* \text{ [kJ/kg]}$  (20),

- if  $H > 0$  then  $p_1^* = p_H^* \cdot p_0 \text{ [bar]}$  (21.1),  $T_1^* = T_H^* \text{ [K]}$  (21.2) and  $i_1^* = C_p \cdot T_1^* \text{ [kJ/kg]}$  (22),

where  $T_H = T_0 - 6.5 \cdot H \text{ [km]}$ , [K] (23) and  $p_H = p_0 \cdot \left( \frac{T_H}{T_0} \right)^{5.2553}$  (24) and

$$T_H^* = T_H + \frac{v^2}{2 \cdot C_p} \quad (25) \quad \text{or} \quad T_H^* = T_H \cdot \left( 1 + \frac{(k-1)}{2} \cdot Mach^2 \right) \quad (26) \quad \text{and}$$

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

- dynamic pressure ratio:

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

- 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\*:

$$p_2^* = \pi_c^* \cdot p_1^* \quad (31), \quad i_2^* = i_1^* \cdot \left( 1 + \frac{(\pi_c^*)^{\frac{k-1}{k}} - 1}{\eta_c^*} \right) \quad (32), \quad T_2^* = \frac{i_2^*}{c_p} \quad (33)$$

- conditions at turbine inlet - station 3\*:

$$p_3^* = \sigma_{ca}^* \cdot p_2^* \quad (34), \quad T_3^* \text{ from equation (8)}, \quad i_3^* = c_{pg} \cdot T_3^* \quad (35)$$

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

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

- burned gas flow coefficient (from mass balance eqn. in combustor):

$$m_g = 1 + m_c \quad (37)$$

- 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\*:

$$i_4^* = i_3^* - l_t^* \quad (40), \quad T_4^* = \frac{i_4^*}{c_{pg}} \quad (41); \quad p_4^* = \frac{\delta_t^*}{p_3^*} \quad (42),$$

- where  $\delta_t^*$  is the pressure ratio in turbine, and it comes out from the expression of specific work in turbine.  $\delta_t^* = \left( 1 - \frac{l_t^*}{i_3^*} \right)^{-\frac{(kg)}{(kg-1)}}$  (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:

$$p_5 = p_{cr} < p_H \quad (44.2), \quad p_{cr} = \left( \frac{2}{kg+1} \right)^{\frac{(kg)}{(kg-1)}} \cdot p_4^* \quad (45)$$

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

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

### 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**  $C_{sp} = \frac{3600 \cdot m_c}{F_{sp}}, \left[ \frac{kg}{Nh} \right]$  (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 at altitude H [km] versus SLS:

$$\dot{M}_\alpha = \dot{M}_{\alpha 0} \cdot \frac{\pi_c^*}{\pi_{c0}^*} \cdot \pi_d^* \cdot \frac{P_H}{P_0} \quad (50)$$

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

$$l_c^* = l_{c0}^* \cdot \bar{n}^2 \quad (51)$$

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

$$\bar{n} = \frac{n}{n_{NOMinal}} \quad (52)$$

- 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:

$$l_{c0}^* = i_0 \cdot \left( \frac{(\pi_{c0}^*)^{\frac{k-1}{k}} - 1}{\eta_{c0}^*} \right) \quad (53)$$

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

- 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):

$$\pi_c^* = \left[ 1 + \left( (\pi_{c0}^*)^{\frac{k-1}{k}} - 1 \right) \cdot \frac{i_0}{i_H^*} \cdot \bar{n}^2 \cdot \frac{\eta_c^*}{\eta_{c0}^*} \right]^{\left( \frac{k}{k-1} \right)} \quad (55)$$

$$\pi_c^* = \left[ 1 + \left( (\pi_{c0}^*)^{\frac{k-1}{k}} - 1 \right) \cdot \frac{i_0}{i_H^*} \cdot \bar{n}^2 \right]^{\left( \frac{k}{k-1} \right)} \quad (56)$$

$$\pi_c^* = \left[ 1 + \left( (\pi_{c0}^*)^{\frac{k-1}{k}} - 1 \right) \cdot \frac{i_0}{i_H^*} \cdot \bar{n}^2 \right]^{\left( \frac{k}{k-1} \right)} \quad (57)$$

$$\pi_c^* = \left[ 1 + \frac{i_c \cdot i_d}{i_H^*} \cdot \bar{n}^2 \right]^{\left( \frac{k}{k-1} \right)} \quad (58)$$



### 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**, Fig. 1; 2/ **VELOCITY MAP**, Fig. 2 and 3/ **SPEED MAP**, Fig. 3.

**(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

<b>Thrust</b>	$F = f(H) \Big _{\substack{V=0 \\ n\% = 1}}$
<b>Specific thrust</b>	$F_{sp} = f(H) \Big _{\substack{V=0 \\ n\% = 1}}$
<b>Specific fuel consumption</b>	$C_{sp} = f(H) \Big _{\substack{V=0 \\ n\% = 1}}$
<b>Altitude Map</b>	

**(2) - VELOCITY 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 flight velocity  $V$  [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

<b>Thrust</b>	$F = f(V) \Big _{\substack{H=0 \\ n\% = 1}}$	<b>Thrust</b>	$F = f(M) \Big _{\substack{H=0 \\ n\% = 1}}$
<b>Specific thrust</b>	$F_{sp} = f(V) \Big _{\substack{H=0 \\ n\% = 1}}$	<b>Specific thrust</b>	$F_{sp} = f(M) \Big _{\substack{H=0 \\ n\% = 1}}$
<b>Specific fuel consumption</b>	$C_{sp} = f(V) \Big _{\substack{H=0 \\ n\% = 1}}$	<b>Specific fuel consumption</b>	$C_{sp} = f(M) \Big _{\substack{H=0 \\ n\% = 1}}$
<b>Velocity Map</b>		<b>Velocity Map</b>	

**(3) - SPEED 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 rotor speed [rpm] or its equivalent engine operational regime [%rpm], while the altitude  $H$  [km] and flight velocity  $V$  [m/s] are constant, their values being usually taken for SLS, ISA conditions

<b>Thrust</b>	$F = f(V) \Big _{\substack{H=0 \\ n\% = 1}}$	<b>Thrust</b>	$F = f(M) \Big _{\substack{H=0 \\ n\% = 1}}$
<b>Specific thrust</b>	$F_{sp} = f(V) \Big _{\substack{H=0 \\ n\% = 1}}$	<b>Specific thrust</b>	$F_{sp} = f(M) \Big _{\substack{H=0 \\ n\% = 1}}$
<b>Specific fuel consumption</b>	$C_{sp} = f(V) \Big _{\substack{H=0 \\ n\% = 1}}$	<b>Specific fuel consumption</b>	$C_{sp} = f(M) \Big _{\substack{H=0 \\ n\% = 1}}$
<b>Speed Map</b>		<b>Speed Map</b>	

### 3.6. THE TURBOJET ENGINE'S UNIVERSAL MAP

**Definitions of UNIVERSAL ENGINE MAP**, (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)

**UNIVERSAL ENGINE MAPS** are represented in coordinates **THRUST PARAMETER** and **SPECIFIC FUEL CONSUMPTION PARAMETER** versus either **flight Mach number** or **SPEED PARAMETER**.

The **UNIVERSAL ENGINE MAP** 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 :

<b>UNIVERSAL ENGINE MAP</b> - definition # 1		<b>UNIVERSAL ENGINE MAP</b> - definition # 2	
$\frac{F}{p_1^*} = f(M) \Big _{\frac{n}{\sqrt{T_1^*}} = const.}$	<b>Thrust parameter</b>	$\frac{F}{p_1^*} = f\left(\frac{n}{\sqrt{T_1^*}}\right)$	<b>Thrust parameter</b>
$\frac{C_{sp}}{\sqrt{T_1^*}} = f(M) \Big _{\frac{n}{\sqrt{T_1^*}} = const.}$	<b>Specific fuel consumption parameter</b>	$\frac{C_{sp}}{\sqrt{T_1^*}} = f\left(\frac{n}{\sqrt{T_1^*}}\right) \Big _{M=const.}$	<b>Specific fuel consumption parameter</b>
$\frac{n}{\sqrt{T_1^*}} = constant$	<b>Speed parameter</b>	$M = constant$	<b>Flight Mach number</b>

Universal Map

#### 4. RESULTS

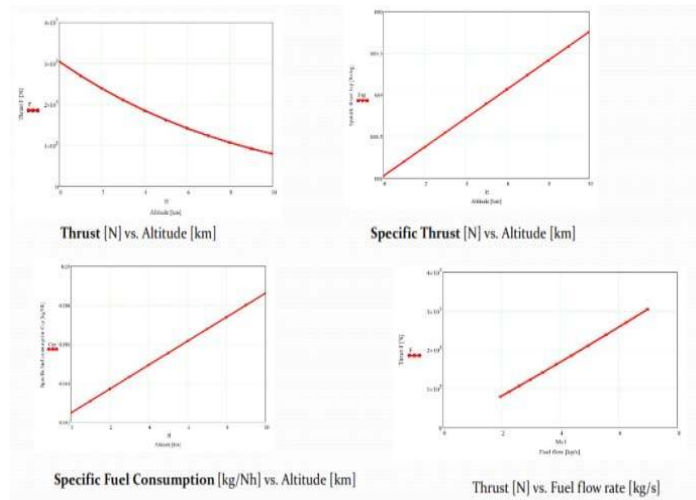


FIG. 1 - ALTITUDE MAP

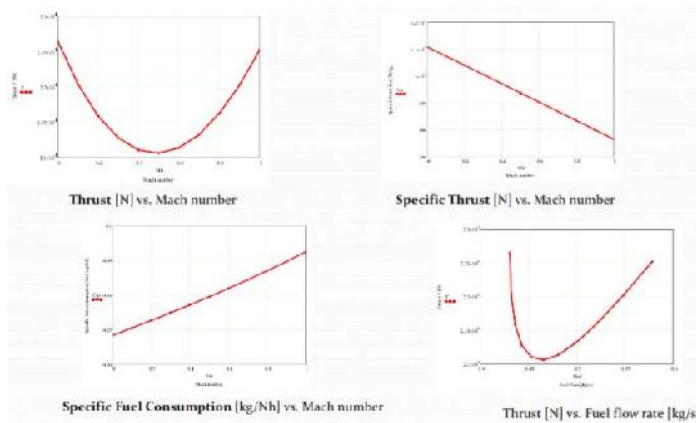


FIG. 2 - VELOCITY MAP

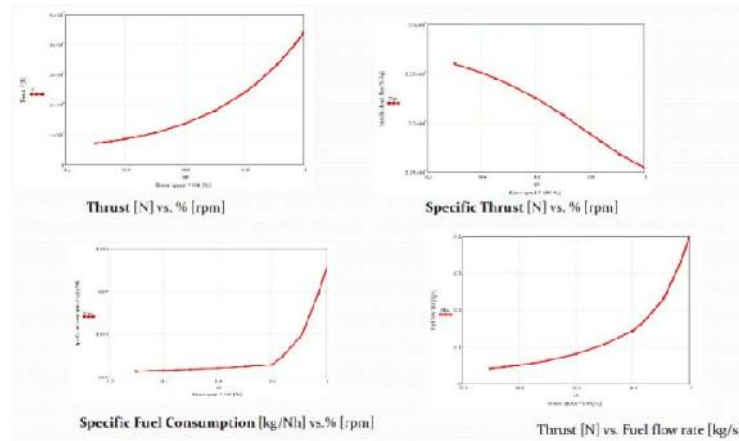


FIG. 3 - SPEED MAP

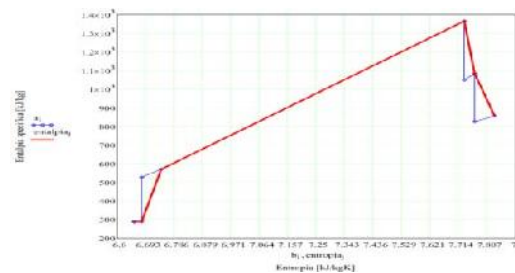
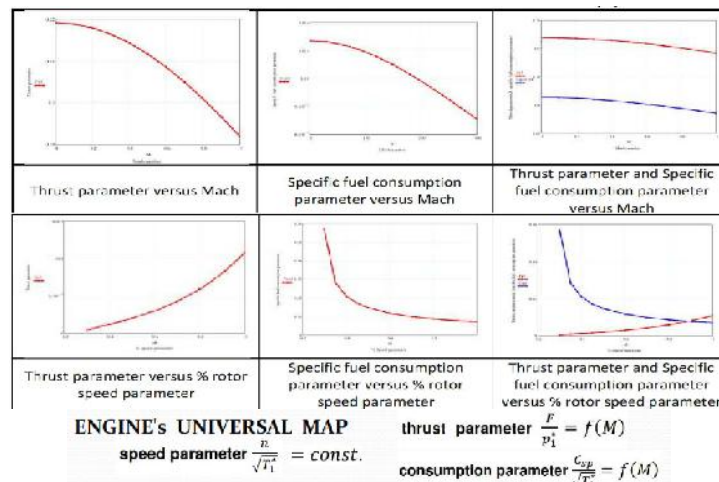


FIG. 4 - J 85 turbojet Brayton cycle (at SLS, ISA conditions)



The turbojet engine's **OPERATING MAPS** are: 1/ **ALTITUDE MAP**, 2/ **VELOCITY MAP** and 3/ **SPEED MAP**, followed by 4/ **UNIVERSAL MAP**, with the appropriate calculation of the thrust parameter and specific fuel consumption parameter.

### 3. CONCLUSIONS & ACKNOWLEDGMENT

The focus of this study is the performance analysis for a single spool turbojet, which was completed within a thorough research; its results were expressed graphically, as the turbojet engine's operational and universal maps. The engine's performances calculated from the in-house codes match the engine data from catalogues.

The authors would like to express their gratitude to the INCAS - National Institute of Aerospace Research "Elie Carafoli" Bucharest for the support while carrying on research activities within the Flow Physics Department, Numerical Simulations Unit, as well as the

documentary research. The results presented in this paper are correlated to the current research activities regarding the jet propulsion systems, carried on within INCAS.

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