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ISSUES ABOUT AIRCRAFT TURBOJET ENGINES CONTROL LAWS

INTRODUCTION

The mathematical pattern that describes the functioning of the parts of a turbojet engine, from the thermogasodynamic point of view, is represented by a system of equations at which the number of the unknown parts are bigger than the number of the equations. So, solving the system of equations without imposed additional terms is not possible. The additional conditions imposed to the parameters of the engine at the changing of the flight conditions are called control laws, and the ones imposed to the parameters of the engine at the changing of the functioning regimes (for the same flight conditions) are called control programmes. Choosing a certain control law or program represents a very important issue, with implications on the height and speed characteristics.
In order to analyse the laws and control programmes of the turbojet engines, it is necessary to find the functioning equation of the set compressor – combustion chamber – turbine.
From the equation of mass flow, written for the section of entrance into the turbine, results:

$$\sqrt{\frac{k}{R}\left(\frac{2}{k+1}\right)^{\frac{k+1}{k-1}}}\cdot S_1\cdot q(\lambda_1)\cdot\frac{p_1^*}{\sqrt{T_1^*}}(1+q_c)\cdot(1-\delta_1)= \\ =\sqrt{\frac{k'}{R'}\left(\frac{2}{k'+1}\right)^{\frac{k'+1}{k'-1}}}\cdot S_3\cdot q(\lambda_3)\cdot\frac{p_3^*}{\sqrt{T_3^*}} \qquad (1)$$

where:
k – the adiabatic exponent;
R – the air constant parameter;
$\delta_1=\frac{G_{ar}}{G_a}$; G_{ar} is the air flow that comes from the compressor for the set climatisation – pressurisation and cooling of the turbine blades, and G_a is the air flow of the engine;

$q_c = \dfrac{G_c}{G_a}$ where G_c is the fuel flow introduced in the combustion chamber;
M , p , T — represent the Mach number, the pressure and the temperature;
The indices "1" si "3" refer to the entrance section in the compressor and the turbine;

$$q(\lambda_1) = \lambda_1 \cdot \left[1 - \frac{k-1}{k+1}\lambda_1^2\right]^{\frac{1}{k-1}} \cdot \left(\frac{k+1}{2}\right)^{\frac{1}{k-1}}; \qquad \lambda_1^2 = \frac{\frac{k-1}{2}M_1^2}{1 + \frac{k-1}{2}M_1^2};$$

Considering that $p_3^* / p_1^* = \pi_c^* \cdot \Gamma_{ca}$, the equation above can be expressed like this:

$$\pi_c^* = q(\lambda_1)\sqrt{\frac{T_3^* \cdot T_0}{T_1^*}} \cdot C_1 \cdot \frac{S_1}{S_3} \tag{2}$$

where $T_0 = 288^\circ K$, and C_1 is a constant.
If the surfaces S_1 (compressor inlet) and S_3 (turbine inlet) are constant, then the equation becomes

$$\pi_c^* = q(\lambda_1)\sqrt{T_{3r}^*} \cdot C_2 \tag{3}$$

The equation (3) represents in co-ordinates $(\pi_c^*, q(\lambda_1))$ a line with the slope $\sqrt{T_{3r}^*} \cdot C_2$.
The constant C_2 is determined by knowing the thermogasodynamic parameters at the calculation regime.
On the other hand, from the equality of the compressor power and of the turbine $N_c = N_t \cdot \eta_m$, results:

$$\left(\pi_c^{*\frac{k-1}{k}} - 1\right)\frac{1}{\eta_c^*} = T_{3r}^* \left(1 - \frac{1}{\pi_t^{*\frac{k'-1}{k'}}}\right) \cdot \text{const.} \tag{5}$$

Replacing the parameter T_{3r}^* from the equations (3) and (5) we obtain the functioning equation of the set compressor – combustion chamber - turbine:

$$\frac{q(\lambda_1)^2\left(\pi_c^{*\frac{k-1}{k}}-1\right)}{\pi_c^{*2}\left(1-\frac{1}{\pi_t^{*\frac{k'-1}{k'}}}\right)\cdot\eta_c^*}=C_3 \qquad (6)$$

This equation can be modified by replacing the turbine pressure ratio, π_t^*, obtained from the equation of the mass flow, written for the inlet section in the stator of the first turbine step and the critical section of the jet nozzle, S_{5crt}:
Considering

$$\pi_t^* = p_3^* / p_4^* \text{ si } T_4^* = \left[1-\left(1-\frac{1}{\pi_t^{*\frac{k'-1}{k'}}}\right)\cdot\eta_t^*\right] \qquad (8)$$

results

$$\pi_t^* \sqrt{1-\left(1-\frac{1}{\pi_t^{*\frac{k'-1}{k'}}}\right)\cdot\eta_t^*} = \left[S_{5crt.}\cdot q(\lambda_{5crt.})/S_3\right]\cdot \text{const.} \qquad (9)$$

From the equation above the turbine pressure ratio π_t^* (that depends on $S_{5crt.}$, S_3 and $q(\lambda_{5crt.})$) is obtained. If the engine has the sections $S_{5crt.}$ and S_3 invariable and it works at a supracritical regime $\left(p_4^*/p_H > \beta_{crt.}\right)$, then $q(\lambda_{5crt.})=1$ and so π_t^* is constant.

CONTROL LAWS

Control law $n=\text{const}$.

For a turbojet engine with $S_{3'}$ (the exit section from the stator of the first turbine step) and S_{5cr} (the critical exit section of the engine) constant, after the control law $n=\text{const.}$, the control factor consists of the fuel flow $G_c=\text{var}$. The automatic control system of the engine ensures the modification of the fuel flow so that at the modification of the inlet parameters in the engine (that is accomplished by the modification of the flight conditions V_H and *H*) to ensure continuously the condition $n = \text{const.}$ In this way it is obvious that the following conditions should be obeyed:

$$T_3^* \le T_{3\ \max}^* \ \text{§ i}\ \Delta K_y \ge \Delta K_{y\min}$$

where

$$K_y = \frac{\left[\pi_c^* / q(\lambda_1)\right]_{line\ pomp.}}{\left[\pi_c^* / q(\lambda_1)\right]_{\text{line funct.}}}, \text{ and } \Delta K_y = (K_y - 1)\cdot 100\,\%$$

In order to determine the engine parameters, under the conditions of the control rule that is given, as an independent variable can be considered: the equivalent rotation $n_r = n\sqrt{T_o / T_1^*}$, the equivalent relative rotation $\overline{n_r} = \overline{n}\sqrt{T_o / T_1^*}$, or the temperature factor $\sqrt{T_o / T_1^*}$, where $T_0 = 288^\circ \text{K}$, $\overline{n} = n / n_0$, n_0 = the maximal rotation of the engine in the conditions $M_H = 0$ *and* $H = 0$.

The independent variable is linked to the flight regime by the temperature T_1^* which is defined by the equation

$$T_1^* = T_H\left(1 + \frac{k-1}{2} M_H^2\right) \tag{10}$$

For $H = \text{const.}$ (namely $T_H = \text{const.}$) along with the growing of M_H the T_1^* grows, too, and so $\overline{n_r}$ decreases and for $M_H = \text{const.}$, with the growing of *H* until 11 km, T_1^* decreases because of the decreasing of the temperature T_H and, so, the $\overline{n_r}$ grows.

Control law $T_3^* = \text{const}$.

Maintaining a temperature T_3^* constant, in front of the turbine of an engine with the geometry of the flow sections, invariable, in the conditions of the modification of the flight regime, it can be achieved by the modification of the fuel flow, so that $T_3^* / T_4^* = \text{const.}$ In this away, the engine rotation will modify because of the temperature T_1^* at the compressor inlet. The line of functioning regimes for a turbojet with uncontroled sections are traced with the help of equation (6), so for the two control laws ($n = \text{const.}$ and $T_3^* = \text{const.}$) the position of this line in the characteristic of the compressor is the same, but for the same value of $\overline{n_r}$, corespond different values of the engine rotation.

For this control rule, the conditions $n \le n_{\max}$ şi $\Delta K_y \ge \Delta K_{y\min}$ must be achieved.

Control law $n_r = \text{const.}$

By this control rule, the automaticall control system modifies the fuel flow, at the modification of the flight conditions, depending on T_1^*, so that the rotation may be proportional with the modification of the parameter $\sqrt{T_o / T_1^*}$, because $n_r = n\sqrt{T_o / T_1^*}$.

The limitations imposed in this case will be: $n \le n_{\max}$ and $T_3^* \le T_{3\max}^*$.

The line of the functioning regimes, which verifies the equation (6) from the compressor characteristic goes to a single point, which coresponds to the design regime, so for all the flight regimes, the values of the parameters π_c^*, $q(\lambda_1)$, and ΔK_y will be constant and equal with the values from the design regime. So, through the design point in the compressor characteristic only a line will cross, coresponding to $T_{3r}^* = \text{const.}$.

Obviously, the control of the engines can be done after one or more parameters. To obtain better characteristics, the combined control rules are used, namely certain control laws for certain flight regimes.

In Fig.1 is shown the variation of $\bar{n}$, $n_{\bar{r}}$ si T_3^* depending on $\sqrt{T_0 / T_1^*}$ for a combined control law. For such a combined control rule, the temperature T_{11}^* from which we pass to a new control law, is chosen to ensure the imposed parameters of the engine at all flight regimes, obeying the limitative conditions: $n \le n_{\max}, T_3^* \le T_{3\max}^*$ *and* $\Delta K_y \ge \Delta K_{y\min}$.

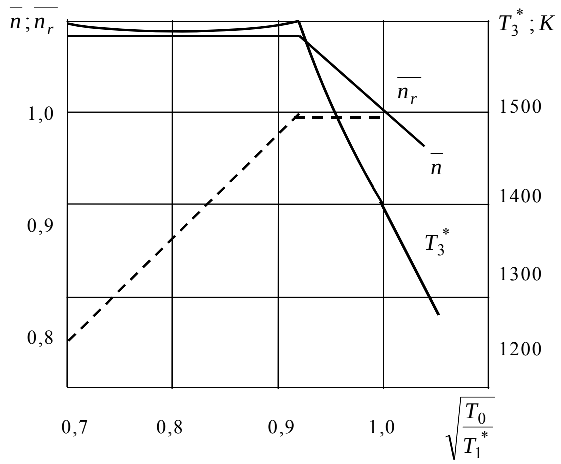

Fig. 1

In Fig. 2 an example of combined control law like $n = \text{const.}$, $T_3^* = \text{const.}$,; $\overline{n_r} = \text{const.}$, $T_3^* = \text{const.}$ is presented.
This control law is possible at the engines with the critical section area af the nozzle $S_{5crt.}$, variable.

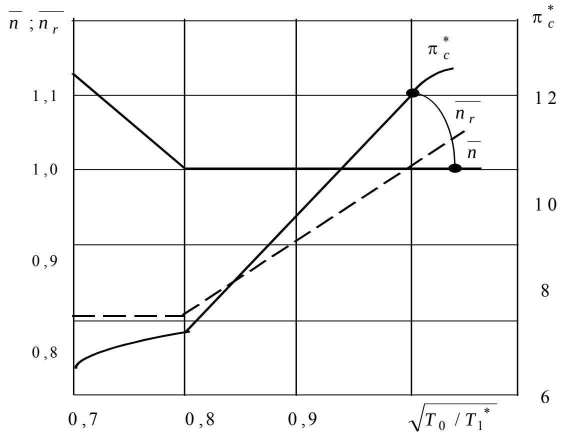

Fig.2

CONCLUDING REMARKS

The equation (6) allows the plot of the line of the functioning regimes of the engine in coordinates $\left(\pi_c^*, q(\lambda_1)\right)$ depending on the law or control program. We see a very big influence on the position of the line of the functioning regimes has a surface of the section of exit from the combustion chamber and turbine entrance. The difficulty of achieving such a variable section is obvious in this area of the engine because of the very high temperature of the combustion gas. Another conclusion drawn from this is that of the possibility of improving the performance of turbofan with a mixture of the two flows after the turbine by modification of this section.

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