Selection criteria of internal parameters of turbofan engine for the type of multi-task aircraft mission

Abstract: The problem described in the paper concerns the selection strategy of the so-called calculation point of the turbofan engine of the multi-task aircraft already at the stage of preliminary design of the aircraft and aircraft engine as a system. A multi-task airplane during each mission needs to perform a series of tasks with different levels of power utilization, while changing the mass, at different altitudes and flight velocities. There was defined the criterion of the LoLoLo mission based on an assessment of mission energy consumption. The calculation results were presented from the point of view of the influence of thermo-gas-dynamic parameters of the comparative cycle circulation (such as compression, turbine inlet temperature, the degree of streams distribution) on the criterion of energy resources usage. The presented criterion allows to make a correct selection of parameters of thermodynamic cycle from the viewpoint of energy possibilities of an aircraft and a mission.

Keywords: multi-task airplane, turbofan engine, engine parameters optimization

1. Introduction

Aircraft engine design is carried out on the assumption that it is an element of a complex system which is the airplane performing the air tasks [1],[2],[3],[5],[6]. In the aircraft design process it is important the choice of the strategy of the selection aircraft engine characteristics, in order to obtain a system that is able to comply with specific tasks with the smallest energy expenditure [3],[5],[6]. At the stage of preliminary design, an analysis and optimization of the parameters of an engine as an aircraft subsystem must be carried out on the basis of the results obtained from the studies on the simplified mathematical models. The required degree of such a simplification on the one hand, results from the need to keep the physical-compliance of the run of the engine characteristics, and on the other one with the possibilities of model analysis for which the characteristics will be “refined” at the further design stages [6]. Considering the energy balance of the aircraft, both the energy required for the flight and the available one and specifying the proportions between these energies for the task, is the most important object determining the choice of engine parameters [7].

The power unit should provide the aircraft with the required performance at all stages of the task, i.e. the air during take-off, climb, overshoot, while performing complex combat maneuvers (turn, hill, full-loop). Energy requirements of the aircraft are for the engine designer rigid constraints. They determine an area of possible changes in the thermogas-dynamic parameters of comparative cycle of the engine, its dimensions, weight, and method of control. To meet the flight requirements of the aircraft, it is necessary, by building a mathematical model of the engine, to include it in a mathematical model of the aircraft.

Obtaining the required accuracy in the design process depends primarily on the mathematical model (models) of the tested aircraft systems. It is assumed that there are known the characteristics of the aircraft for which there will be selected the power unit. This means that it is known an aerodynamic characteristics of the aircraft, the mass parameters of the airplane and the performed tasks, defined by the flight parameters such as air velocity and altitude. Under this assumption, the aim is to find such engine parameters (in terms of thermogas-dynamic parameters of the comparative cycle of the engine), which will identify (for the adopted
criteria and constraints) the optimum best adjusted characteristics of the engine and the aircraft to perform the required air task. The parametric analysis is carried out on the basis of the dimensionless parameters. This allows to find the main directions of the optimization research of the engine and the aircraft as a whole. Taking into account the engine model in the mathematical model of an airplane and the model of the performed air task it is possible to conduct an analysis of the results through the evaluation of the selected criteria which subsequently allow for the dimensioning of physical quantities, which is necessary for further analysis.

In the case of the selection of the power unit for the communication one usually looks for such thermodynamic cycle parameters of the engine, which for the assumed trajectory will minimize fuel consumption. Multi-task airplane, while performing a number of different tasks, often in one mission, needs to use its energy resources to achieve the desired situation of maneuverability. The issue of fuel consumption minimalization, although it is a criterion, is not the most important. Equally important is to get the proper dynamics of the aircraft, which is connected with the engine at long range from those that provide the minimal fuel consumption.

Therefore, while choosing the drive unit for the multi-task aircraft, it should be considered these parameters and evaluation criteria of the engine which provide air tasks at the required energy expenditure. The search for the optimal values of engine parameters must be a compromise between obtaining the required tactical and technical characteristics on the one hand and minimization of the fuel usage as an economical criterion on the other one. Thus, the choice of engine parameters should be based on energy criteria. Some of the selected parameters and the criteria to adapt the characteristics of the engine and the aircraft are not only process design parameters that define the dimensions of the engine and the aircraft, but first of all determine the efficiency of the entire the airplane-air task system.

The energy balance of the aircraft (balancing the power required for the flight and the available power) is the basis for the assessment of:

- aircraft maneuverability,
- mass ratios,
- economic criteria.

The use of dimensionless parameters that further will be treated as "the adjustment parameters" of the engine characteristics to the aircraft, will then allow to build an analytical energetic model of the system: power unit-airplane-air task. As a result, it is possible to further assess the impact of the adopted engine parameters. It is assumed that the characteristics of the aircraft (mass, aerodynamic ones) are known and the change of parameters of the thermodynamic engine circulation do not change them.

The formula for dimensionless thrust loading factor as [3],[7]:

$$v = \frac{a}{g} \frac{dMa}{dt} + \left(1 + \frac{a}{g} M_s^2 \frac{d\Theta}{dt}\right) \sin \Theta + \frac{1}{2} \frac{k c_s M_s^2}{\psi_s}$$

where:

- \(a\) – speed of sound,
- \(g\) – gravitational acceleration,
- \(M_s\) – the Mach’s number,
- \(t\) – time,
- \(\Theta\) – the angle of aircraft climbing,
- \(k\) – adiabatic exponent for the air,
- \(c_s\) – aerodynamic drag coefficient of the aircraft,
- \(\psi_s\) – dimensionless coefficient of aircraft mass.

It is a formula that for the established parameters for air tasks (\(M_s\), \(\psi_s\)) allows to determine the necessary flight coefficient \(v\). It is further assumed that the value of \(v\) determined from the aircraft flight conditions will be later marked with the index "N" - \(v_N\).

On the other hand, the equation (1) shows that the value of \(v\) can be determined for the power unit. This value, as the available one, will be in the case for its calculation for the engine marked with the index "R". According to [7]:

$$v_R = \frac{K_{st} S_{ZN}}{\psi_s}$$

where:

- \(K_{st}\) – dimensionless thrust loading factor [3],[7],
- \(S_{ZN}\) – dimensionless ratio of engine-aircraft geometric matching.

2. The degree of use of aircraft energy resources during the mission

Performing the flight under certain conditions (altitude, speed, aircraft weight) is possible only if engine available thrust balances engine drag forces and the forces of inertia. The essential value of the thrust loading factor \(v_N\) is balanced during the flight by the available value of this parameter for the engine \(v_R\) which is described by formula (2). Of course, the value required for the flight changes during a mission depending on the flight conditions. If one assumes that the engine at every stage of the flight works on the maximum range, then the physical course of the speed-altitude characteristics of the engine is conditioned by the calculation point (\(H, M_s\)) and the thermodynamic cycle variables. This means that the value of the available thrust for the selected flight conditions may differ from the required one. It can therefore be introduced a crite-
rion to assess the level of energy use in the mission, described by the formula:

\[ A = \frac{\sum_{i=1}^{k} V_{R,i} L_{n}}{L} \]  

(3)

The condition of searching such a solution of this criterion is that at each \( n \)-th stage of the mission the value

\[ V_{R,n} \geq 1 \]  

(4)

If this value is less than unity, the task cannot be performed, because the thrust of the engine is lower than necessary. Criterion (4) is a measure of matching the engine characteristics to the airplane and the mission. If this value is less than unity, then the task cannot be performed, because the thrust of the engine is lower than the required one. Criterion (4) is a measure of matching the characteristics of the engine to the aircraft and the mission. In the extreme case, i.e. when the engine is optimized for the aircraft, the value of this criterion is equal to the number of elementary stages of the mission "k". This is due to the fact that at every stage of the mission, the ratio of (4) can achieve a minimum value equal to unity (perfect choice), and the sum of elementary sections is equal to \( L \). Thus, while choosing the engine to the power requirements of the mission, it needs to be chosen so that the value of the index (3) was the lowest while maintaining the condition (4). But this is not the only measure of matching the engine to the aircraft.

3. The energetic analysis of the aircraft mission

The variety of air missions are discussed in detail in [3],[4]. On the basis of the data presented, the further analysis it was chosen the LoLoLo mission (Low ceiling intake, combat maneuvering at Low altitude, back to Low height) This is a typical mission to support the battlefield. The proceed to the combat zone (weapon release) is at low altitude of the flight with subsonic flight speed (ca. \( Ma=0.5...0.8 \)). Air battle itself, as in this and subsequent missions, is modeled for the need to work with a series of turns, by full 360 degrees, with different overload factor and the different velocities of flight.

In the case of the mission as shown in Figure 1 the speed maneuver is \( Ma=0.8, H=0 \). It is assumed that during the struggle the aircraft gets rid of the load – weapon which constitutes 0.2 of the aircraft take off mass. The return to the airport takes place at the same altitude but with the bigger velocity of \( Ma=0.8 \).

The study developed the model simulation, and the following assumptions for the calculation of parameters of the air mission were considered:

1. take off (\( H=0-500m \)), run at the length of 500 m,
2. approaching to the battlefield, the length of the run 300 km, \( Ma=0.8 \),
3. air battle (turn 360°, load coefficient \( n=4 \), \( Ma=0.8 \), manoeuvre time 40s.). During the manœuvre there is the drop of the load of the relative mass of 0.2 of the aircraft take off mass,
4. return to the airport \( Ma=0.8, H=0 \), length of the run 300 km.

It is assumed that the engine in each of the flight stages works at its maximum but without using an afterburner. In the calculations the acceleration stage to \( Ma=0.8 \) was removed, as irrelevant from an energy point of view.

Figure 2 shows the necessities to obtain the value of the \( V_N \) coefficient in subsequent stages of the mission. The highest values of the coefficient of the required thrust are defined for the turn and the subsonic flight, and these states are critical for determining the engine thrust in the mission LoLoLo.
The calculations of the available thrust (in a dimensionless form) for a number of established values of the variables that define the engine cycle were conducted. The calculation results are shown in Figure 3.

![Fig. 3 Influence of the selected parameters of engine thermodynamic cycle \( \pi, T_3, \mu \) and flight velocity on the value change of the thrust load coefficient \( v_R \)](image)

A turn is possible only if the value of \( v_R \) is at least equal to the required value of this coefficient. It is, therefore necessary to choose such parameters of the calculation circuit and the calculation point to obtain, due to the engine characteristics, at the flight velocity in turn, the required value of \( v_R \). In order to illustrate the example there were examined several variants of the engine which differed by the values of engine compression \( \pi \) and the turbine inlet temperature \( T_3 \).

The value of the by-pass ratio was constant, and in this particular calculation example was \( \mu = 0.5 \). The engine, in which the overall compression of the compressor was \( \pi = 25 \), and \( T_3 = 1500[K] \) met the requirements only at the take off. Increasing the temperature \( T_3 \) (up to 1750 K) and an increase in the compression to \( \pi = 25 \) enable to meet the air requirements in the critical for the calculations stage of the flight.

In this example, the computational terms of the engine (dimensioning the engine) were adopted for the take off. Meeting the requirements for the turn maneuver causes significant "overdimensioning" of engine energy possibilities for the take off, which will necessitate with the engine throttle at this stage of the mission. The calculation results of Figure 3 do not take into account the decrease in the weight of the aircraft as the result of fuel consumption at the stages prior to a turn. Mass reduction (specifically in the adopted model, the reduction of the wing load factor), to obtain the required value of the thrust load, will simultaneously reduce the dimensionless thrust. Thus, the actual value of \( v_R \) will be lower than that presented in Figure 3.

The selection of the engine calculation point significantly affects the change course of the available thrust coefficient, as shown in Figure 4. For the same engine thermal cycle parameters, there were conducted the calculations for the two calculation points. Red lines show the run of the dimensionless load coefficient \( v_R \), when the calculation point is determined for takeoff conditions (\( Ma = 0, H = 0 \)). In blue it is marked the course of the values of \( v_R \) when the calculation conditions were \( Ma = 0.8, H = 0 \) (as for the turn and return to the airport). It is clear that a change in the conditions of the calculation (as in a flight) causes the increase in the available value of \( v_R \), considerably above the value required for the conduct of the flight. In addition, on the same figure it was presented the influence of the afterburner switch-on the available characteristics. In case of the conducted analysis, the solution may be the selection of the calculation conditions for the take off and the performance of the turn quarantines’ the use of the afterburner at this stage of the flight. The engine in this case is relatively little "overdimensionised" to comply with the necessary conditions.

![Fig.4 The influence of the selection of engine calculation point on the course of the value of load factor of engine available thrust. The calculations were made for: \( \pi = 25, T_3 = 1500[K], \mu = 0.5 \). Broken lines mean the run of the characteristics with the switched on afterburner. DP-design point, AB-afterburner](image)

Determination of the engine dimensions is the primary task assigned to the first stage of design. Engine dimensions (diameter, length) determine the characteristics of mass, velocity and altitude as they determine the mass air intensity flowing through the engine. Knowing the energy requirements of the aircraft at any stage of the mission and the power possibilities of the engine, at the given variables of the comparative circuit, it was determined the minimum value of \( S_{ZN} \) for the researched mission (Fig. 5). The condition to perform each of the stages is to satisfy the condition (4). The graph in Figure 5 shows that the choice of \( S_{ZNMIN} \) is determined by the conditions of take off. The value of the parameter \( A \) (3) during the take-off, for the assumed engine variables, is exceeded only for the \( S_{ZN} \) close to 0.05. Steady flight at subsonic velocity is not a state dimensioning the engine, already for the small values of the relative engine dimension the obtained values of \( v_R/v_N \) are greater than unity whereas for small values of \( S_{ZN} \) the inclination angles of the curves are rising much faster than for the take off and the turn. The turn maneuver is possible with \( S_{ZN} \) values similar to those as for the take off (for the
assumed value of G-load coefficient in the turn and the turn velocity). For the higher values than G-load coefficient \( n \) and the velocity in turn, it would be the turn as the state dimensioning the engine, but this would increase \( S_{ZN_{MIN}} \) values. Lowering the value of the \( S_{ZN} \), for the selected limiting states of the flight is possible with the use of afterburner. Then the effect of “over-dimensional” of the engine for the subsonic steady flights could be reduced.

\[ \text{Fig.5 Determining of the minimum value of } S_{ZN} \text{ parameter for the condition (4), at each stage of the mission (for the exemplary engine data } \pi=25, T_3=1700[K], \mu=0.5) \]

Assuming further as a prerequisite to perform the task condition (4), it is possible to check how variables of the engine circuit influence on the change of its value (with simultaneous indication of the parameter combinations for which the task cannot be performed).

Figure 6 shows the curves run of \( \nu_b/\nu_N \) as a function of engine compression ratio \( \pi \), for the two limit values of the turbine inlet temperature: \( T_3=1300 \) K (solid lines) and \( T_3=1750 \) K (broken lines).

\[ \text{Fig.6 Influence of the compression } \pi \text{ on the run of the values } \nu_b/\nu_N \text{ at the selected mission stages, for two values of turbine inlet temperatures: } T_3=1300 \text{ K (solid lines) and } T_3=1750 \text{ K (broken lines), for } \mu=0.5 \]

For smaller values of turbine inlet temperature the engine is not able to provide the required excess of the thrust at any stage of the mission. The obtained values of \( \nu_b/\nu_N \) are less than unity at all stages of the mission. For the maximum temperature value \( T_3 \), adopted for the calculations, it is obtained the required value of the ratio of relative thrusts \( \nu_b/\nu_N >1 \) for the take-off, in the accepted for the calculations range of the thrust changes. In Figure 6 for the sake of transparency of the figure, it was not shown the run of the curves \( \nu_b/\nu_N \) for subsonic flight (with \( \text{Ma} = 0.5 \) and \( \text{Ma} = 0.8 \)), as the determined values for this flight conditions of \( \nu_b/\nu_N \) repeatedly exceeded the minimum requirements. With the increase of the compression in the compressor there is a noticeable drop in the value of \( \nu_b/\nu_N \) especially for subsonic flight and the turn, but for the take-off, especially for maximum adopted \( T_3 = 1750 \) K, the value of \( \nu_b/\nu_N \) is practically constant, close to 1. This means that the take off conditions determine not only the selection of geometric dimension of the engine \( (S_{ZN}) \), but most of all the choice of variables of the engine thermodynamic cycle. The increase in the by-pass ratio \( \mu \) will decrease the available thrust compared to the required one (Fig. 7).

\[ \text{Fig.7 Influence of the by-pass ratio } \mu \text{ on the run of } \nu_b/\nu_N \text{ in the selected mission stages for two values of the turbine inlet temperature } T_3=1300 \text{ K (solid lines) and } T_3=1750 \text{ K (broken lines), compression ratio } \pi=25 \]

The range of possible acceptable values of \( \nu_b/\nu_N \) when changing \( \mu \), is relatively small, in spite of the inclusion in the calculations the relatively high values of engine cycle. Limitations, due to the reduction of the thrust ratio are shown in Figure 7 as a vertical line, plotted on the curve characterized the take off conditions. Above the limit value \( \mu = 0.55 \) in the presented example, the execution of the take off on the given distance of the take off run \((500 \text{ m})\) is not possible. To make a take off with higher values of \( \mu \) it is necessary either to use afterburning or lengthen the runway of the aircraft. The performance of the elementary stages of the mission, for the engine variant characterized by lower values of cycle variables (solid lines in Figure 7) is virtually impossible.

5. Conclusions

To sum up this part of the research it can be con-

cluded that for the execution of all stages of the mission of the aircraft, marked with the symbol
LoLoLo, it is necessary to adopt the engine variant with high values of thermodynamic cycle parameters and possible low values of by-pass ratio. The determined, based on the energetic criteria of the aircraft mission, variables of the cycle \( \pi, \mu, T_3 \) restrict the area of the selection of the possible combination of these variables. The values of these parameters determined in the example are consistent with those that characterize the engines of the multitask airplanes in service today. These specific parameters determine the technological level of the engine and the airplane and allow for the comparison of generations of the engines and choosing the right combination of parameters at the stage of the initial design of the aircraft.

**Bibliography/Literatura**


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