ISSN: 1231-4005
e-ISSN: 2354-0133
DOI: 10.5604/12314005.1138277

INFLUENCE OF THE GAS TURBINE ENGINE DESIGN PARAMETERS ON THE ENERGY CONSUMPTION OF THE MULTIROLE AIRCRAFT MISSIONS

Piotr Wygonik

Rzeszów University of Technology
Department of Airplane and Aircraft Engines
Powstańców Warszawy Avenue 12, 35-959 Rzeszów, Poland
tel.: +48 17 8651241, fax: +48 17 8651444
e-mail: piowyg@prz.edu.pl

Abstract

In the article the analysis of the influence of design parameters of the engine on the performed by the multirole aircraft mission was performed. The problem is complex as a result of performing a series of manoeuvres in various conditions of flight. A special feature of multi-role aircraft mission is a sudden (even pulse) weight change and exactly its reduction as a result of the discharge of cargo bomb, rocket or due to the consumption of ammunition during air combat manoeuvring. Reducing the airplane mass by the weight of the fuel consumed (continuously) and the used weapons radically differ the demands on the energy required to overcome gravity and drag forces. The article shows how the reduction of the aircraft mass influences on the change of the thrust load factor. It was built the mathematical model of the system engine-aircraft-air job (taking into account the flight conditions, elements of the mission – subsonic and supersonic flight, flight time, the heat-and-gasdynamic and mass model of the engine). The model enables for the simulation research of the complex flight missions and their evaluation on the basis of the constructed criteria. The model includes a parametric description of physical processes in the turbofan engine, thus provides a direct assessment of the impact of selection of engine parameters on the effectiveness of the mission. The results of calculations according to classical criteria (e.g. kilometre fuel consumption, specific fuel consumption) were presented. The paper presents new criteria, which enable to analyze the energy consumption of the complex mission of the aircraft (e.g. energy consumption: the unit range, the degree of utilization of energy resources and the carried out the mission engine). Criteria were built in by combining the parameters necessary for the flight with disposable ones. On the basis of these parameters there was done an assessment of the "quantitative" adjustment of a power unit to various missions such as subsonic, supersonic and mixed (for different their proportion), for different levels and the plane range. The results were presented in a pictorial way on numerous charts.

Keywords: *gas turbine engine design, simulation, airplane-engine integration, aircraft mission optimization*

1. Introduction – definition of the mission energy consumption

An aircraft motion is a consequence of specific energy transformations that arise from the method of supplying the energy required for the flight [1-3, 6]. The energy can be fed to the aircraft directly from the power unit (thrust work) or by changing the potential energy to overcome air resistance while reducing the height.

Depending on the way how the aircraft motion is forced by the thrust force one can distinguish the following cases:

- forced motion only by the thrust force, resulting in a flight at a constant speed or accelerated motion,
- reducing the altitude – forced by the working of the component of the terrestrial gravity force,
- delayed motion caused by the shortage of thrust force against the force of aerodynamic drag,
- braking (during landing) by using a parachute, air brakes, thrust reversers causing dispersion of kinetic energy of the aircraft.

Aircraft mission is carried out at certain distances and consists of a number of individual flight stages which include: take off run and the aircraft take off, climb to altitude, flight at a constant speed, horizontal acceleration, accelerating with the climb, manoeuvres (loop, pull up, turn-bend, etc.). Aircraft flight forced by the thrust force is an essential condition of its motion. The necessary amount of energy needed to sustain such a motion is determined by the product of the fuel mass $m_{\Sigma pal}$ supplied to the engine combustion chamber (and possibly to the afterburner) and its calorific value *Wu,* hereinafter referred to as the total energy consumption [5]. The mission of the aircraft consists of successive stages which differ in speed and altitude. This causes the variation of the energy balance. The energy consumption of the aircraft mission is the total energy expenditure at all driven stages of the mission [5]:

$$E_R = \sum_{n=1}^{k} E_{R,n} + \sum_{m=1}^{p} E_{k,m} , \tag{1}$$

where:

$\sum_{m=1}^{p} E_{k,m}$ – sum of the acquired kinetic energy at the *m*-stages of the aircraft acceleration.

In the energy consumption, balance there will not be considered the stages of take-off run and engine work on the range of small acceleration.

2. Energy consumption of the aircraft range

Energy consumption of the range is defined as the ratio of total energy supplied to the aircraft at the driven stages to the distance during the mission:

$$E_z = \frac{\sum_{n=1}^{k} E_{R,n}}{L} , \tag{2}$$

where:

$L = \sum_{n=1}^{k} L_n$ – mission length, the sum of length of the elementary flight stages.

At the certain flight conditions, the work of resisting forces is balanced by the work of thrust force at the given motions. It can therefore be written that:

$$E_{R,n} = K_{sil,n} \, L_n . \tag{3}$$

Therefore, the formulae for the range energy consumption of the mission can be written as:

$$E_z = \frac{\sum_{n=1}^{k} K_{sil,n} \, L_n}{\sum_{n=1}^{k} L_n} . \tag{4}$$

Energy consumption of the range [J/m] means, in a physical sense, the work the thrust force needs to perform in order to move an aircraft at a single distance.

3. Unitary energy consumption

Unitary energy consumption of the n-th stage of the $E_{j,n}$ mission is defined as the ratio of the energy consumption of the motion $E_{R,n}$ to the product of the mass of the aircraft and the distance travelled during the elementary section (stage) of the flight [5, 7-9]:

$$E_{j,n} = \frac{E_{R,n}}{m_n \, L_n} , \tag{5}$$

where:
m_n – aircraft mass at the beginning of the n-th flight stage.

This criterion is defined by the work the thrust force needs to perform to move the mass of the aircraft at the assumed distance. As the energy consumption of the motion is equal to the labour of the thrust force $K_{sil,n}$ on the way L_n, then at the elementary stage of the flight the unitary energy consumption is equal to:

$$E_{j,n} = \frac{K_{sil,n}}{m_n}. \tag{6}$$

Knowing the course of the mission, one can determine the value of the unitary energy consumption for each elementary step of the mission and determine these stages that require to be delivered the greatest energy. These stages will determine the engine and the choice of engine and cycle parameters from the viewpoint of the mission feasibility. Knowing the value of unitary energy consumption at every stage of the mission, one can determine the total (sum) unitary energy consumption of the mission from the formula:

$$E_{j,\Sigma} = \sum_{n=1}^{k} E_{j,n} . \tag{7}$$

Flight operations under certain conditions (altitude, speed, weight of the aircraft) is possible only if the available engine thrust balances the work of the forces of aerodynamic drag and inertial forces. The necessary value of the thrust load factor ν_N is balanced, during the flight, by the available value of this parameter for the ν_R engine, which is described in [8, 9]. The value required for the flight changes during a mission in the manner specified by flight conditions [2, 4, 6]. If we assume that the engine at each flight stage works at the maximum range [2, 4], then the physical course of the engine speed-altitude characteristics is determined by the design point (*H, Ma*) and by changing the thermodynamic cycle. This means that the available thrust value for the given flight conditions may differ from the required thrust. A condition to find a solution for this criterion is that at each (n-th) stage of the mission, the value [2, 8, 9]:

$$\frac{\nu_{R,n}}{\nu_{N,n}} \geq 1 . \tag{8}$$

If this value is smaller than a unity then the task cannot be performed as the engine thrust is smaller than the required. Criterion (8) is a measure of the rate of matching of the engine characteristics to the aircraft and the missions.

4. Calculations results, conclusions

For the analysis it was chosen an exemplary LoLoLo mission (Low approaching ceiling, fighting manoeuvring at Low altitude, return at Low altitude) is a typical mission to support the battle field [1, 2], Fig. 1.

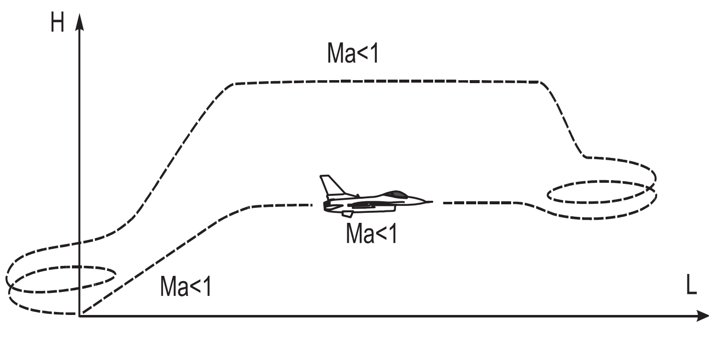

Fig. 1 LoLoLo mission, H = 0–500 m, Ma<1

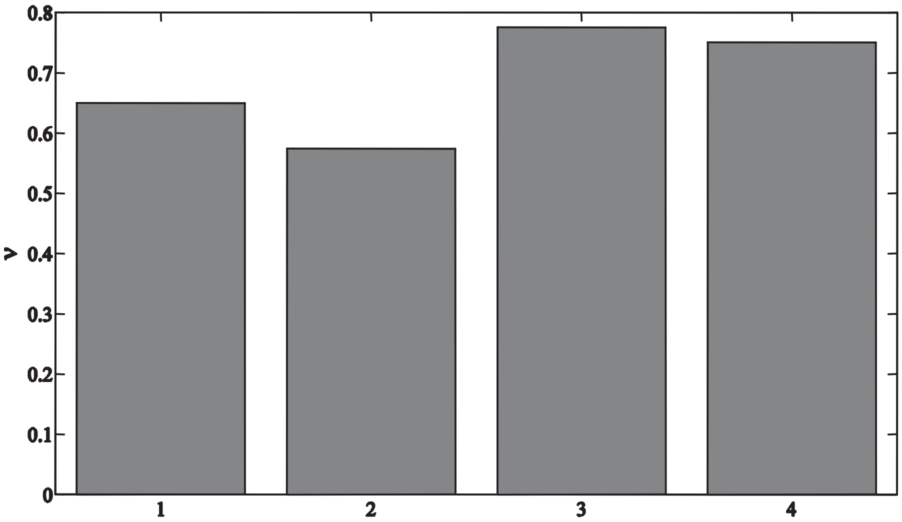

Fig.2 The necessary value of the dimensionless thrust load factor ν at the subsequent stages of the NNN mission

It is the mission carried out according to the diagram in Fig. 1. It consists of the following air stages (tasks):

1. Take off (H = 0–500 m), take off run at the length of 500 m,
2. Approaching to the fight zone, distance length of 300 km, Ma = 0.8,
3. Air fight (turn 360°, G-load coefficient n = 4, Ma = 0.8, the manoeuvre time 40 sec). During the manoeuvre the cargo is discharged of the relative weight $\overline{m}_{uzbr} = 0.2$ (referred to the aircraft take-off weight),
4. Back to the airport with Ma = 0.8, H = 0, the length of the distance 300 km.

It is assumed that the engine at each of these flight stages works at the maximum range, without using an afterburner. In the calculation it was omitted the stage of aircraft acceleration to the speed Ma = 0.5, as irrelevant from the energetic point of view.

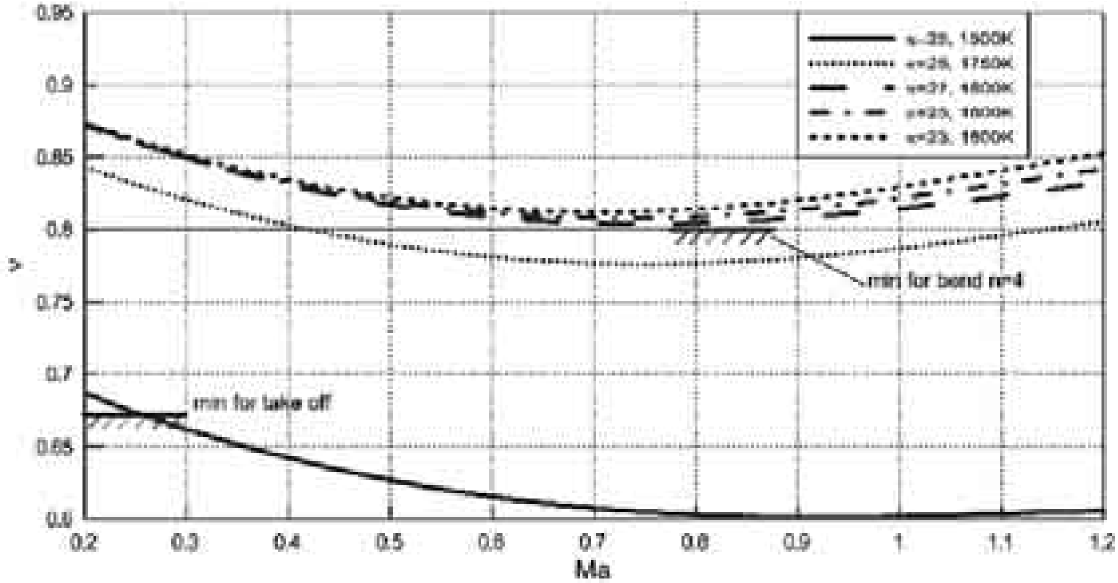

Fig. 3. Influence of the selected parameters of the engine thermodynamical cycle π_S^*, T_3^*, μ and the flight speed on the change of the value of thrust load coefficient ν_R

Figure 3 shows the necessary to obtain values of thrust coefficient ν_N at the subsequent stages of the mission. The highest values of the necessary thrust coefficient are determined for the turn and subsonic flight, and these stages are critical for determining the thrust of the engine in the LoLoLo mission. There were conducted calculations of the available thrust coefficient for a number of established values of the variables that define the engine cycle. The calculation results are shown in Fig.3. The execution of a turn is possible only if the value of ν_R will be at least equal to the necessary value of this coefficient. It is therefore necessary to choose such parameters of thermal cycle of the engine and the design point to, because of course of the engine characteristics, obtain at a speed of flight in the turn, the required value ν_R. Several options of the engine have been considered which differ in values of π_s^*, T_3^*. The value of the by-pass ratio was constant and in this particular example was calculated and was $\mu = 0.5$. The engine, in which the total compression of the compressor was $\pi_s^* = 25$, and $T_3^* = 1500\,\mathrm{K}$, fulfilled just the demands on the take off. The increase of the temperature T_3^* (up to 1750 K) and an increase in compression ratio to $\pi_s^* = 27$ allowed to meet the volatile requirements of the aircraft at the critical for the calculations stage of the flight. In this example, the calculation engine conditions (measuring the engine) were adopted for the take off. Meeting the requirements for the turn manoeuvre causes significant "over-sizing" of the energetic possibility of engine for the take-off, which will be connected with the necessity of the engine throttle at this stage of the mission. The selection of the engine design point influences significantly on the changes of available thrust coefficient as shown in Fig. 4. For the same parameters of the engine, thermal cycle there were conducted the calculations for two design points. The first two lines (in legend) show the course of the dimensionless thrust load factor ν_R when the design point is determined for the take off conditions ($Ma = 0$, $H = 0$). The rest lines (in legends) shows the course of the values of the ν_R coefficient when the calculation conditions were $Ma = 0.8$, $H = 0$ (as for the turn and return to the airport). It may be noted that the change in calculation conditions (as for the overshoot) causes an increase in the available value ν_R, well above the values required for execution of the flight. In addition, the same figure shows the influence of the afterburner use on the available characteristics of the engine. For the conducted analysis, the solution may be the right choice of the calculation conditions for the take off and the execution of the manoeuvre guarantees the use of afterburner at this stage of the flight. The engine in this case is relatively little "oversized" to meet the requirements of the necessary conditions.

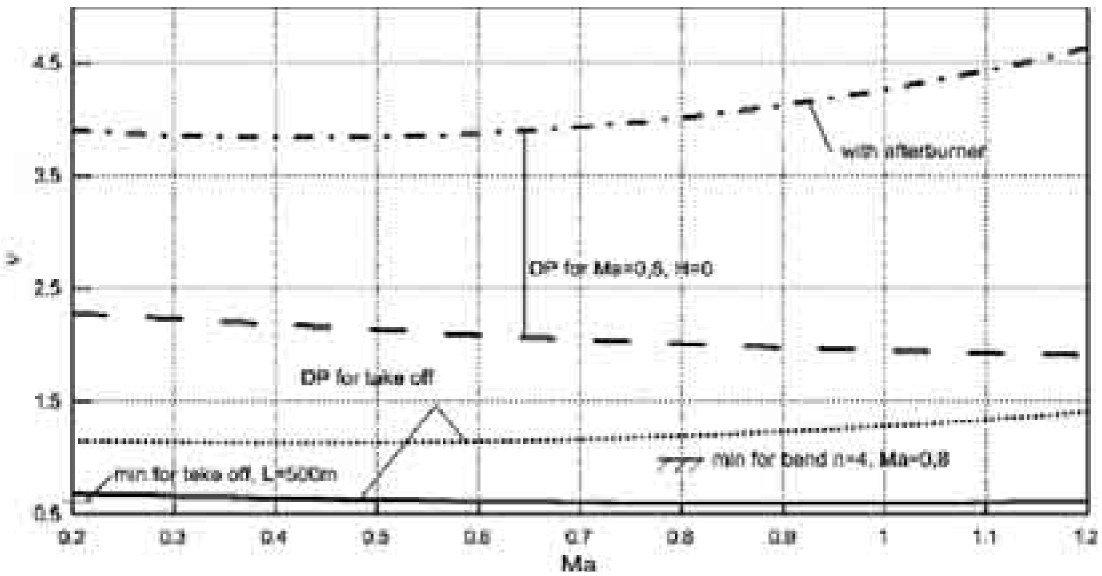

Fig. 4. Influence of the engine design point on the course of the value of the available thrust load factor of the engine. The calculations have been done for the following data: $\pi_s^* = 25$, $T_3^* = 1500$, $\mu = 0.5$. Dashed lines mean the course of the characteristics with the turned on afterburner, DP – design point

Assuming further as a prerequisite for the task execution that the ratio $\nu_R / \nu_N \geq 1$, it is possible to check how the variables of the engine cycle influence on the change of its value (with simultaneous indication for which combination of thermodynamic parameters, the task cannot be performed).

Figure 5 shows the course of the curve $\nu_R / \nu_N \geq 1$ as a function of the compressor's compression $\pi^*_{\Sigma s}$ for two boundary values before the turbine T_3^* – 1300 K (solid lines) and 1750 K (dashed lines). For smaller values of temperature before the turbine, the engine is not able to provide the required excess of the thrust to perform any stage of the mission. The obtained values ν_R / ν_N are smaller than unity at all stages of the mission. For the maximum temperature values T_3^*, which were adopted to the calculations, it is obtained the required value of the relative thrusts ratios for the take-off, in the accepted for the calculations range of compression changes. In Fig. 5, to keep the clarity of the figure, the course of the curves ν_R / ν_N has not been shown for subsonic flight (with $Ma = 0.5$ and $Ma = 0.8$), as the determined values of ν_R / ν_N for such flight conditions exceeded the minimum requirements several times. With the increase of the compression in the compressor, it is a noticeable drop in the value ν_R / ν_N, in particular for the subsonic flight and the turn, whereas for the take off, especially for the maximum adopted $T_3^* = 1750\ \mathrm{K}$, the value ν_R / ν_N is practically constant, close to 1.

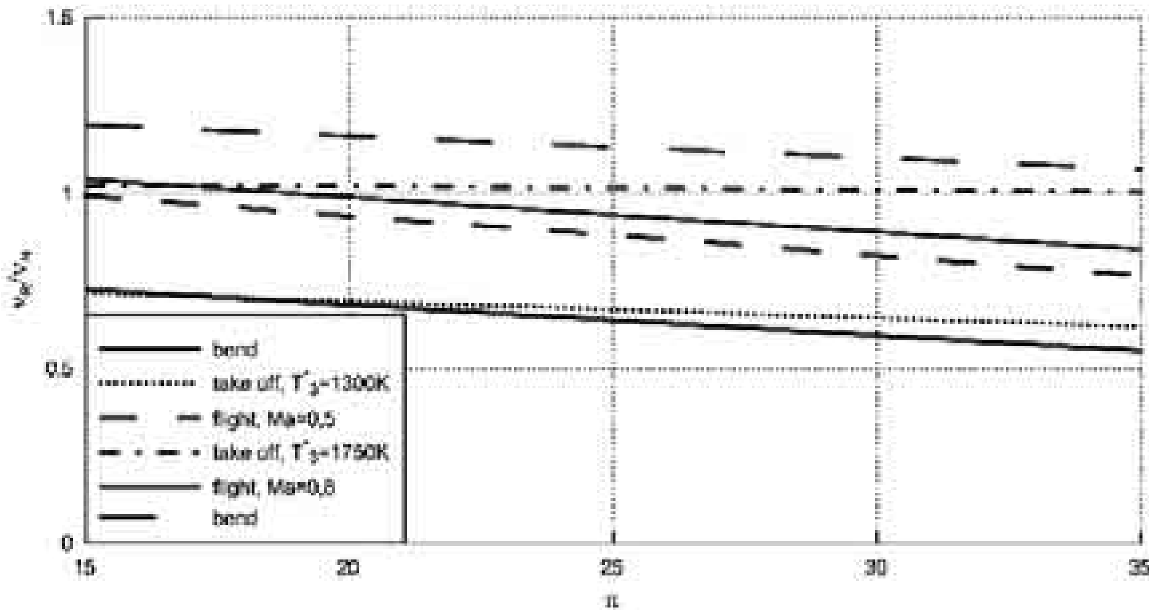

Fig. 5. Influence of the compressor's compression $\pi^*_{\Sigma s}$ on the course of the value ν_R / ν_N at the selected mission stages for two temperature values before the turbine $T_3^* = 1300\ K$ (solid lines) and $T_3^* = 1750\ K$ (dashed lines), $\mu = 0.5$

The increase of the by-pass ratio μ lowers the available thrust in relation to the necessary thrust (Fig. 6). The range of acceptable values ν_R / ν_N while changing μ is relatively small, in spite of the inclusion in the calculations of the relatively high values of the variables of the engine cycle. The reduction, due to a decrease of the thrusts ratios has been shown in Fig. 6 as vertical lines. Above the boundary value μ of 0.55 in this example, the take off at the distance of the take off run (500 m) is not possible. To perform the take off at higher values of μ, it is necessary either to use afterburner or lengthen the road of the take off run. The implementation of the elementary stages of the mission, for the engine variant which is characterized by lower values of cycle variables (solid lines in Fig. 6) is virtually impossible.

To conclude this piece of research it can be stated that for the execution of all stages of the aircraft LoLoLo mission, it is necessary to adopt a variant of the engine with high values of the thermodynamic cycle parameters and possible low values of the by-pass ratio.

The range energy consumption is a criterion of the degree of "engagement" of the thrust work in the flight for the given distance. The greater the work per unit of the travelled distance, the higher the energy intensity of the mission. In Fig. 7, it was shown the influence of the engine cycle parameters on range energy consumption of the aircraft.

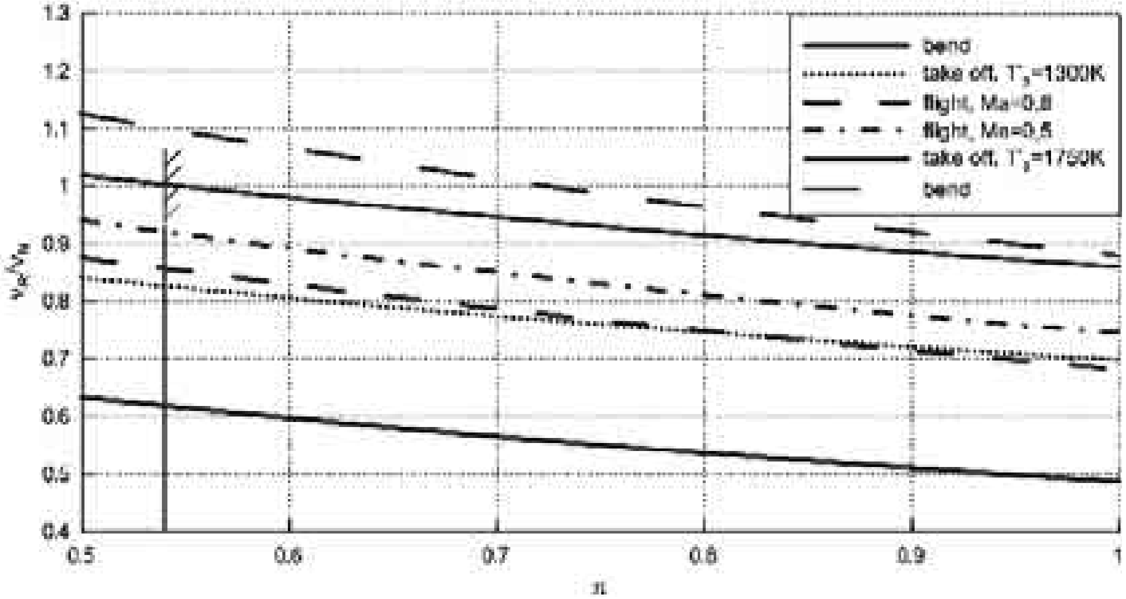

Fig. 6. Influence of the rate of by-pass μ on the course of the value v_R / v_N at the chosen mission stages for two temperature values before the turbine $T_3^* = 1300$ K (solid lines) and $T_3^* = 1750$ K (dashed lines) and the value of the compressor's compression of $\pi^*_{\Sigma s} = 25$

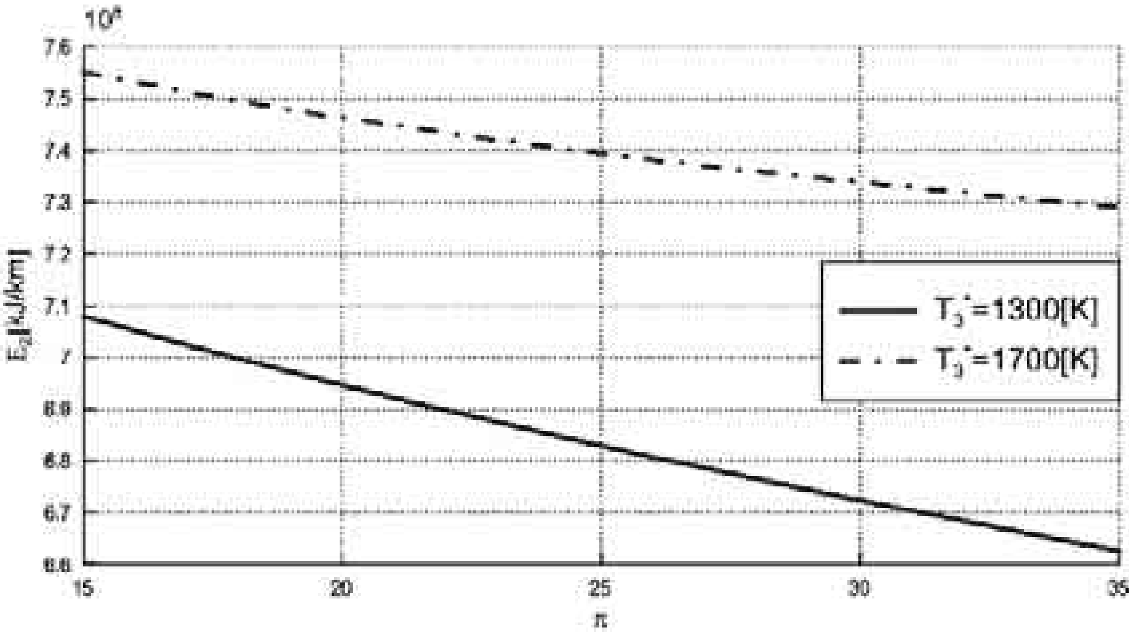

Fig. 7. Influence of the change of the compressor's compression $\pi^*_{\Sigma s}$ and temperature T_3^* on the range energy consumption E_z

The application of high temperatures before the turbine causes that the energy consumption of the mission increases. This increase is however necessary for the execution of all stages of the mission. The increase of the compression in the compressor and the by-pass ratio causes the decrease in the range energy consumption. This means that the amount of work that needs to be done to overcome the individual flight distance decreases. Fig. 8 shows the influence of $\pi^*_{\Sigma s}$ changes on unitary energy consumption i.e. relative to the instantaneous mass of the aircraft. Unitary energy consumption allows to specify the size of the work done by the thrust force to transport the unit of the aircraft mass per unit distance. Therefore, it constitutes a valuable criterion which allows to compare the management of energy efficiency in various missions. In the calculations, the aircraft mass change was taken into account caused by fuel usage and due to the discharge and the whole mass of weapons during a turn manoeuvre of the aircraft. The influence of cycle parameters on the nature of the changes of unitary energy consumption curves is similar to range energy consumption. With the increase in the value of cycle variables, the amount of energy expended to move the mass of the aircraft at a designated distance decreases.

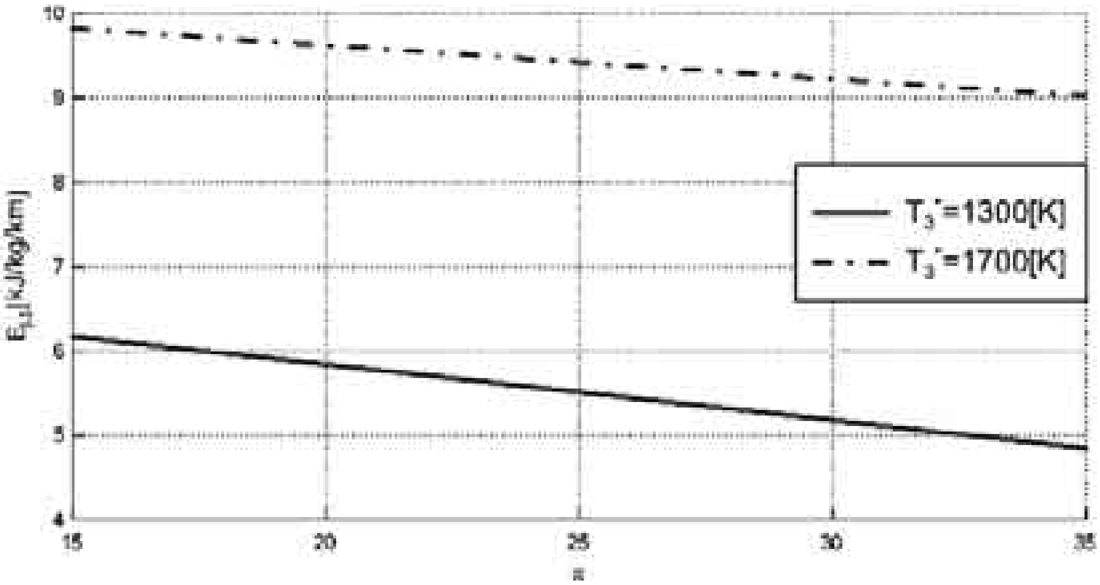

Fig. 8. Influence of the change of the compressor's compression $\pi^*_{\Sigma s}$ and temperature T^*_3 on the unitary energy consumption $Ej,_{\Sigma}$

Energy consumption is therefore a secondary criterion in the selection of the engine parameters for a multi-role aircraft, and the course of the curves without any clear extreme does not determine the uniqueness of the selection. The most important for the engine selection is the criterion (8). Thanks to this criterion, it is possible to check the influence of engine parameters and the choice of the calculation point on the possible sub-tasks and the entire mission.

References

[1] Herteman, J. P., Goutines, M., *Design Principles and Methods for Military Turbojet Engines*, RTO-MP, AC/323 (AVT) TP/9 Design Principles and Methods for Aircraft Gas Turbine Engines, February 1999.

[2] Orkisz, M., et al., *Podstawy doboru turbinowych silników odrzutowych do płatowca*, Biblioteka Naukowa Instytutu Lotnictwa, Warszawa 2002.

[3] *Performance Prediction and Simulation of gas Turbine Engine Operations*, RTO-TR-044 AC/323 (AVT-018) TP/29, RTO Technical Report 44, RTO/NATO April 2002.

[4] Rumiancev, S. W., Sgilevskij, W. A., *Sistemnoe projektirovanie awiacionnogo dwigatela*, Izidatelstwo MAI, Moskva 1991.

[5] Siłka, W., *Energochłonnośc ruchu samochodu*, WNT, Warszawa 1997.

[6] Stricker, J. M., *The Gas Turbine Engine Conceptual Design Process – An Integrated Approach*, RTO-MP, AC/323 (AVT) TP/9 Design Principles and Methods for Aircraft Gas Turbine Engines, February 1999.

[7] Ubysz, A., *Energochłonność samochodu a zużycie paliwa w złożonych warunkach ruchu*, Wydawnictwo Politechniki Śląskiej, Gliwice 2003.

[8] Wygonik, P., *Kryteria doboru parametrów silnika turbinowego do samolotu wielozadaniowego*, Silniki Spalinowe, Nr 4, 2006.

[9] Wygonik, P., *Influence of Basic Energetic Parameters of Turbofan Engine on Multipurpose Aircraft Manoeuvres Indexes*, PTNSS, 2011.