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SPECIFIC NET WORK OR THERMAL CYCLE EFFICIENCY, ONE OF THE QUESTIONS, ENGINEERS MUST FACE DESIGNING HELICOPTER TURBOSHAFT ENGINES [2][3]

The real gas turbine engine cycles, between given temperature limits and component efficiencies, have optimums of specific net work and thermal cycle efficiency, which are function of compressor pressure ratio. These two optimums are significantly different from each other. In accordance with it, it is impossible to produce the maximum specific net output work and the maximum thermal efficiency in the same time. Using this programme, described in this paper, the user can determine the optimum pressure ratios for the maximum specific net work, the maximum thermal efficiency and as a compromise seeking, for the maximum of their multiplied value, meanwhile gets the actual specific net work and thermal efficiency. Furthermore this programme provides the possibility to analyze how the above mentioned question at the today existing turboshaft engines was answered by engine specialists.

FAJLAGOS HASZNOS-MUNKA VAGY TERMIKUS HATÁSFOK, EGYIKE A MÉRNÖKÖK ÁLTAL MEGVÁLASZOLANDÓ KÉRDÉSEKNEK HELIKPOTER HAJTÓMŰTERVEZÉS ESETÉN

A valós gázturbinás körfolyamatok, adott hőmérséklet határok között és gépegység veszteségek mellett, mind fajlagos hasznos-munka, mind pedig termikus hatásfok szempontjából rendelkeznek optimumokkal, amelyeket a kompresszor nyomásviszonyával lehet jellemezni. A fajlagos hasznos-munka és a termikus hatásfok maximumaihoz tartozó nyomásviszony értékek jelentősen eltérnek egymástól. Ennek megfelelően egy adott hajtóművel egy időben nem produkálható mindkét optimum. Ebben a cikkben leírt program lehetővé teszi a maximális fajlagos hasznos-munkához, a maximális termikus hatásfokhoz és a kompromisszumos optimumkeresés jegyében a szorzatuk maximumához tartozó nyomásviszonyok, illetve a hozzájuk tartozó fajlagos hasznos-munka és termikus hatásfok értékek meghatározását. Továbbá a program lehetővé teszi, hogy megválaszoljuk, hogy a fenti kérdést a ma létező helikopter gázturbinák esetében hogyan válaszolták meg a hajtómű specialisták.

Nomenclature

$\oint du$: change of internal energy around the cycle loop;

$\oint dw$: work transfer around the cycle loop;

$\oint dq$: heat transfer around the cycle loop;

w_{net}: specific net work (the useful mechanical work per unit of mass);

w_e: specific expansion work (the total work per unit of mass, done by the system);

w_c: specific compression work (the work per unit of mass, transferred to compress the incoming air);

q_{in}: input specific heat (the whole amount of specific heat transferred into the cycle);

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q_{out}: output specific heat (rejected or waste specific heat in the thermodynamic cycle);
η_t: thermal cycle efficiency of real cycle;
π: compressor pressure ratio;
T_1: intake inlet temperature;
T_3: turbine inlet temperature;
η_{ti}: thermal cycle efficiency of ideal cycle.
$w_{net}(\pi)$: specific net work as a function of pressure ratio;
$w_{net}'(\pi)$: first derivative of specific net work as a function of pressure ratio;
$q_{in}(\pi)$: input heat as a function of pressure ratio;
$q_{in}'(\pi)$: first derivative of input heat as a function of pressure ratio;
$\eta_t(\pi)$: thermal efficiency of real cycle as a function of pressure ratio;
$\eta_t'(\pi)$: first derivative of thermal efficiency as a function of pressure ratio;
κ_a, κ_g: adiabatic exponent for compression and expansion process;
c_{pa},c_{pg},c_{pb}: isobaric specific heat for compression and expansion and combustion process;
σ: pressure loss for the whole engine (air intake, combustor, exhaust pipe, others);
η_{polc},η_{pole}: polytrophic efficiency of compression and expansion process;
η_m: mechanical efficiency and the power needed to drive auxiliary units;
η_b: combustion efficiency.

INTRODUCTION

Definition of specific net work and thermal cycle efficiency

Examining the thermal cycles we can conclude that the internal energy changes during the course of the cyclic process, but when the cyclic process finishes, the system's energy is the same as the energy it had when the process began. If the cyclic process moves clockwise around the loop, then it represents a heat engine, and we get net output work. If it moves counter clockwise then it represents a heat pump. The gas turbine engine cycle, which is based on the Brayton (or with other name Joule) cycle, is represent one of the heat engine cycles. Here I deal with two favourable features which are really important while designing a gas turbine engine cycle. They are the specific net output work and thermal cycle efficiency, which significantly define the worthiness of a gas turbine engine.

Mechanical (specific net) work (w_{net}) is the desired output of the thermodynamic cycle. In accordance with the second law of thermodynamics the mechanical work is always less than the input heat. The remaining heat dissipated as wasted heat into the environment. The cyclic integral (1) is zero because the internal energy (u) is a function of state, and any cycle returns the system to its starting state. In this case all energies (work and heat) are written in terms of a unit of mass (J/kg). The advantage of this method that the mechanical work (specific net work) as a desired output, does not depend on the size of the heat engine (gas turbine), given, with the thermal cycle efficiency, a suitable performance indicator for the evaluation of any kind of engine. [1]

$$\oint du = \oint dw + \oint dq = 0 \;\rightarrow\; \oint dw = -\oint dq \;\rightarrow\; w_{net} = w_e - w_c = q_{in} - q_{out} \quad (1)$$

Thermal cycle efficiency (η_t), in general, energy conversion efficiency is the ratio between the useful output of a device and the input, in energy terms (2). For thermal efficiency, the input (q_{in}) to the device is heat, or the heat-content of a fuel that is consumed. [1]

$$\eta_t = \frac{w_{net}}{q_{in}} = \frac{q_{in} - q_{out}}{q_{in}} = 1 - \frac{q_{out}}{q_{in}} \quad (2)$$

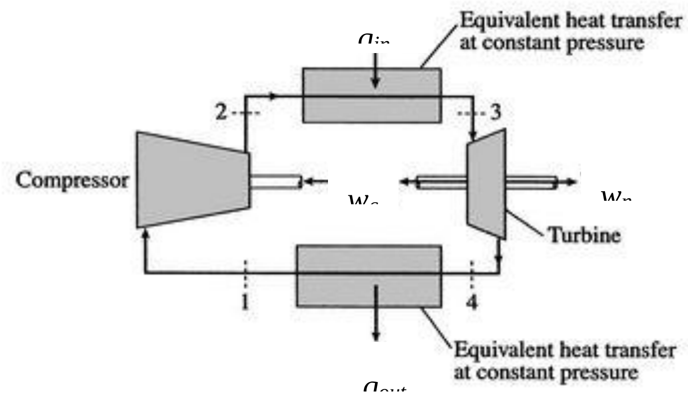

Figure 1. Thermodynamic model of gas turbine engine cycle for power generation [9]

Producing a gas turbine engine both two features are very important design criteria. Higher specific net work for a given shaft power (or thrust) means less mass flow rate decreasing the engine dimensions and weight, while thermal cycle efficiency defines the specific fuel consumption of the engine and through it a significant item of life cost.

To examine the specific net work at first enough to trace the ideal Brayton cycle. As Figure 2 proves us, between the lower and upper limit of temperatures (T_1–T_3) infinite number of cycle can be created.

The upper limit of temperature (T_3), the turbine entry temperature, is fixed by materials technology and cost. (If the temperature is too high, the blades fail). The lower limit of temperature (T_1), the air intake inlet temperature, is determined by the weather conditions and flying altitude, but practically it is always fixed to the temperature of International Standard Atmosphere (ISA) at 0 m altitude, which is 288 K.

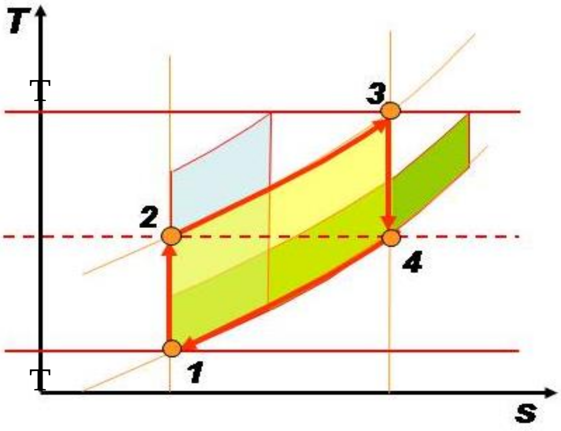

Figure 2. Brayton cycles in T-s diagram representing different specific net output work

It can be seen that for every T_3/T_1 ratio there are two pressure ratios, when the specific net work is zero and there is a pressure ratio when the value of specific net work is maximum. How we can get this pressure ratio? We have to create the $w_{net}(\pi)$ function, where the specific net work (w_{net}) is function of compressor pressure ratio (π). As a next step we have to find the extreme (maximum) of the specific net output work as a function of compressor pressure ratio. Local extreme (maximum) can be found by using the first derivative test of function. Specific net output work is maximum, where the first derivative of specific net work is zero ($w_{net}'(\pi)=0$).

The thermal efficiency is different, because the examination of ideal cycle does not give us any information about the local maximum of thermal efficiency as a function of compressor pressure ratio. Increasing the compressor pressure ratio, the ideal thermal efficiency continuously increases with the rising pressure ratio (π), as it is shown in Figure 3 and Equation 3.

$$\eta_{ti} = 1 - \frac{1}{\pi^{\frac{\kappa-1}{\kappa}}} \tag{3}$$

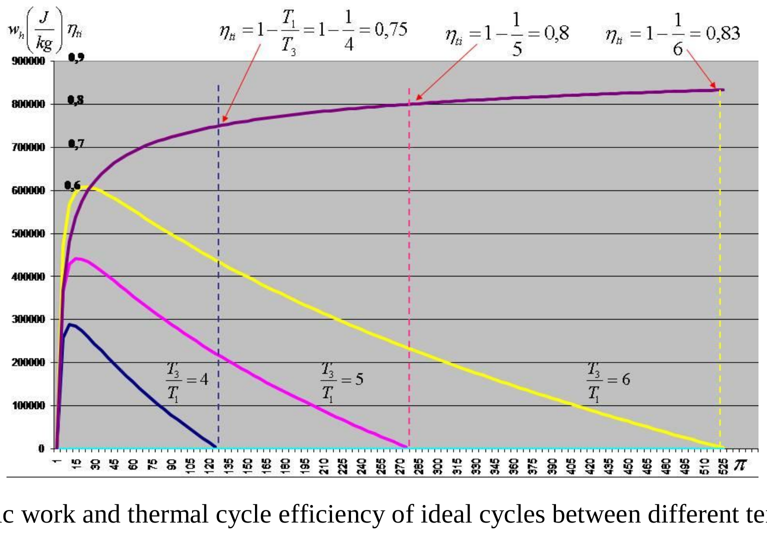

Figure 3. Specific work and thermal cycle efficiency of ideal cycles between different temperature limits

As I mentioned earlier, at ideal Brayton cycle, when the pressure ratio (π) goes to infinity, the ideal thermal efficiency (η_{ti}) goes to one. Despite the ideal, the real Brayton cycle's thermal efficiency is zero when the specific net work is zero in accordance with the Equation 2, quasi it has a pressure ratio at which the thermal cycle efficiency is maximum. Accordingly the real gas turbine engine cycle have optimums of both specific net work and thermal cycle efficiency, which are functions of compressor pressure ratio. These two optimums are significantly different from each other so it is impossible to produce the maximum specific net output work and the maximum thermal efficiency in the same time.

Maximum specific net output work as a function of compressor pressure ratio

For the first step we should define the specific net work as the difference of expansion and compression work (4) taking into consideration the temperature dependence of gas properties and the engine element efficiencies and losses.

$$w_{net}(\pi) = c_{pg} T_3 \left(1 - \frac{1}{(\sigma\pi)^{\frac{\kappa_g - 1}{\kappa_g} \cdot \eta_{pole}}} \right) - \frac{c_{pa} T_1}{\eta_m} \left(\pi^{\frac{\kappa_a - 1}{\kappa_a} \cdot \frac{1}{\eta_{polc}}} - 1 \right) \tag{4}$$

Local extreme (maximum) can be found by using the first derivative test of Function 4. Specific net output work is maximum, where the first derivative of net work is zero, see Function 5.

$$w_{net}(\pi) \quad \text{is maximum, where: } w_{net}'(\pi) = 0 \tag{5}$$

Maximum thermal cycle efficiency as a function of compressor pressure ratio

Thermal efficiency as a function of pressure ratio is expressed in accordance with Equation 2 and it is detailed in Function 6.

$$\eta_t(\pi) = \frac{w_{net}(\pi)}{q_{in}(\pi)} = \frac{c_{pg} T_3 \left(1 - \frac{1}{(\sigma\pi)^{\frac{\kappa_g - 1}{\kappa_g} \cdot \eta_{pole}}} \right) - \frac{c_{pa} T_1}{\eta_m} \left(\pi^{\frac{\kappa_a - 1}{\kappa_a} \cdot \frac{1}{\eta_{polc}}} - 1 \right)}{\frac{c_{pb}}{\eta_b} \left(T_3 - T_1 \cdot \pi^{\frac{\kappa_a - 1}{\kappa_a} \cdot \frac{1}{\eta_{polc}}} \right)} \tag{6}$$

Following the previous method we examine the first derivative of function.

$$\eta_t(\pi) \text{ is maximum, where: } \eta_t'(\pi) = 0 \tag{7}$$

In accordance with the general rules of derivation we get the Function 8. and Equation 9.

$$\eta_t'(\pi) = \frac{w_{net}'(\pi) \cdot q_{in}(\pi) - w_{net}(\pi) \cdot q_{in}'(\pi)}{\left[q_{in}(\pi)\right]^2} \tag{8}$$

$$0 = w_{net}'(\pi) \cdot q_{in}(\pi) - w_{net}(\pi) \cdot q_{in}'(\pi) \tag{9}$$

To process the above mentioned theories Microsoft Excel was used with Visual Basic programming. Microsoft Excel Worksheet provides the communication platform (input and output data) of the created Visual Basic programme. The model produces:

- The above mentioned characteristic curves in turbine entry temperature versus compressor pressure ratio diagram for any kind of combination of engine component efficiencies;
- Calculates the distinguished compressor pressure ratio values;
- Creates specific net work, thermal cycle efficiency web in compressor polytrophic efficiency versus pressure ratio diagram;
- It takes into consideration the change of compressor polytrophic efficiency as a function of blade length giving possibility to evaluate its effect on the distinguished pressure ratios, thermal cycle efficiency and specific net work;
- It provides the analyses and evaluation of existed turboshaft engines.

In the next part of paper I present an example of these engine characteristics and pick a bunch of currently used turboshafts to evaluate their performance data.

Thermal efficiency characteristics

In Figure 5 thermal cycle efficiency can be shown from 20 to 36%, considering the component efficiencies shown in the bottom left corner. Three curves from left to right represent the compressor pressure ratios of best specific work (green), best thermal efficiency (red) and as a compromise between them a maximum value of their multiplication (blue).

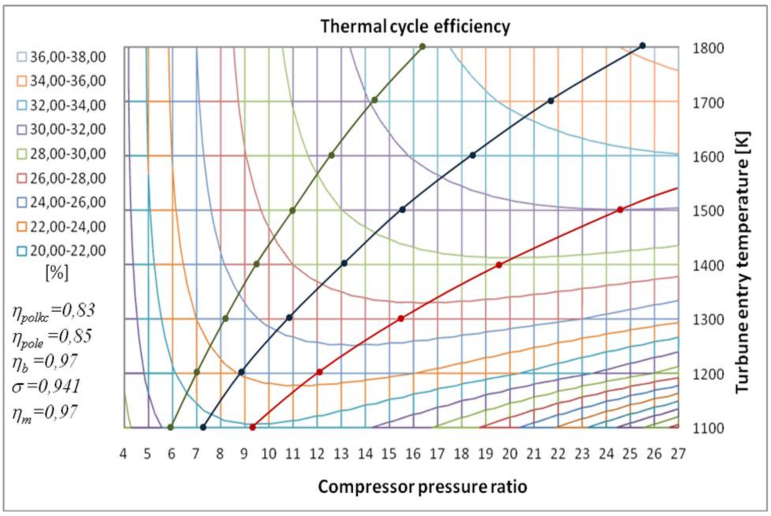

Figure 5. Thermal efficiency curves in turbine entry temperature versus pressure ratio diagram

Thermal efficiency characteristics

In Figure 6 specific net work can be shown from 150 to 450 kJ/kg. Engine component efficiencies and distinguished pressure ratio curves are unchanged. What is clearly visible that at a given turbine entry temperature and at low compressor pressure ratios both examined excellence indicators worsen heavily. The reason can be proven by simple mathematical deduction. Neglecting the deduction we have to recognise that both functions are rational functions with vertical asymptotes, where at decreasing pressure ratio, the value of turbine entry temperature goes to infinity.

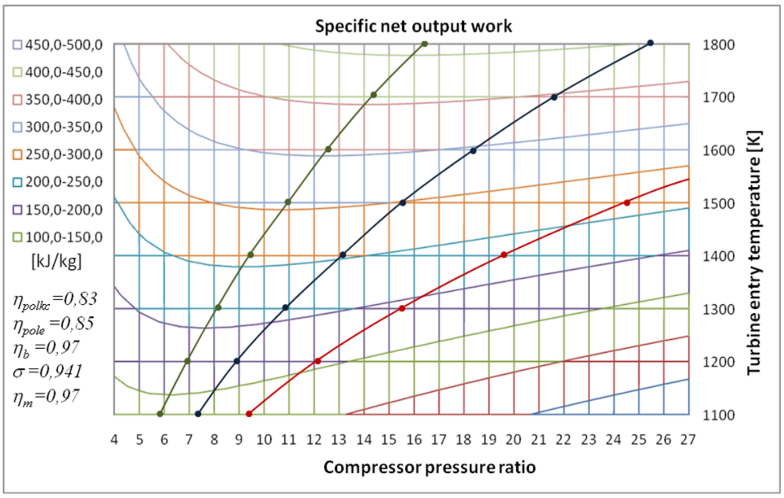

Figure 6. Specific net work curves in turbine entry temperature versus pressure ratio diagram

Ideas about the position of take off working point

Presuming, the designers in most cases can work with given engine components, which mostly determine the component efficiencies and turbine entry tempeature. In this case the only thing they can vary, is the compressor pressure ratio. Of course, the compressor pressure ratio must not be out of the above mentioned tresholds, namely the pressure ratio of maximum specific net work and thermal cycle efficiency. At a low turbine entry temperature and consequently low specicific net work it is reasonable to position the take off working point close to the lower eligible pressure ratio. It is even more justified if the raise of pressure ratio causes significant deterioration of compressor poytrophic efficiency, because in this case the lower compressor polytrophic efficiency can consume the hoped advantage of higher thermal cycle efficiency. At higher turbine entry temperature, consequently higher specific net work there is larger space for the designers to play with the compressor pressure ratio. One reason that the a small drop of the specific net work, from a relatively high value, is not so painful like at originally low specific net output work. In addition at this range the deterioration of specific net work at increasing pressure ratio is not so significant like at lower turbine entry temperature.

Consequently, I presume, that at older turboshaft engines with lower turbine entry temperature the take off working point is posotioned in the first part of possible pressure ratio range, between the green and blue line, by Figure 5 and 6. At newer engines with higher turbine entry temperature it is more likely to tend to the upper limit of pressure ratio even accepting the deterioration of compressor polytrophic efficiency, however it decreases this upper limit of pressure ratio. In the next chapter I would like to prove these ideas analysing some existing turboshaft engines.

ANALYSES OF EXISTING TURBOSHAFT ENGINES

First step for the above mentioned analyses is to collect all possible available performance data of some existing turboshaft engines. Of course, there are numerous turboshaft engines, but the companies are not eager to share too much performance data. Choosing the turboshaft engine I preferred the middle category by their shaft power. There is only one exception. This one is the LM 2500 turboshaft which is used rather as an industrial and marine gas turbine with much larger size and shaft power comparing to the chosen helicopter engines. During the evaluation I concentrated for five important data:

- Shaft power of take off RPM (P_{shaft});
- Turbine entry temperature (T_3) (or any other temperature in hot section);
- Compressor pressure ratio (π);
- Engine mass flow rate $(\dot{m})$;
- Specific fuel consumption (SFC) or thermal efficiency(η_t).

Shaft power, specific fuel consumption and pressure ratio was almost always available. The turbine entry temperature caused the strongest uncertainty as either missing or clearly deformed data. It did not thwart the examination but weaken the verification of results. During the evaluation process the shaft power and the compressor pressure ratio were the two fixed data in the process. The mass flow rate and specific fuel consumption were the two data I used to smooth the model using turbine entry temperature and component efficiencies as variables. I continued changing the variables until both above mentioned data $(\dot{m}; SFC)$ of the model and the existing engine became equal, which was the requirement to accept the final results of model. As I mentioned earlier, the turbine entry temperature in some cases were confusing. Finding exact data was easier in case of two Russian (ex-Soviet) helicopter engines, TV2-117A and TV3-117, because of their available maintenance manuals. Their turbine inlet temperature only slightly differed from the result of the model. From this conformity I concluded that in case of missing turbine entry temperature or when it is clearly out of reasonable range I accept the result of thermal model. To be honest I did not try to achieve totally precise results analysing the existing turboshafts. The reason was the relatively large number of analysed engines. To collect all engine data would have been time consuming and sometimes futile. The second reason was that my main object was rather to present how this thermal model works practically. Simplifications I used during the analyses process:

- I fixed the combustion efficiency (0,97), pressure losses and (0,941) mechanical efficiency (0,97) for all engines (I do not presume significant variance from a good average, in addition their influence is much less than the influence of compressor and turbine polytrophic efficiency) ;
- I did not take into consideration pressure ratio dependence of expansion polytrophic efficiency (it much less depends on the pressure ratio then the compression polytrophic efficiency);
- I tied the compressor **inlet section** polytrophic efficiency to expansion polytrophic efficiency (it does not mean that the two component efficiency is equal, because at rising compressor pressure ratio with the increasing number of stages and shorter compressor

blades decreases the compression polytrophic efficiency);

- I estimated the compressor inlet section medium diameter because of the lack of data and as its consequence I had to fix the estimated axial velocity of the compressor.

Of course, the model would be able to handle the above listed features in case of proper available data, providing more exact results about the examined turboshaft engines. Despite the simplifications I believe the results of next bunch of turboshaft engine analyses bellow are close to their real performance parameters.

TV2-117A turboshaft engine

The TV2-117 turboshaft engine along with the VR-8 main gearbox was developed in 1964 as a power plant for the Mi-8 medium-load helicopter designed by the Mil Design Bureau. It became one of the most popular helicopters in the world. The engine was mass-produced until 1997. About 23,000 TV2-117A engines were produced; their total time in operation exceeds 100 million hours. [3] It has excellent airworthiness and the engine with this helicopter is a real workhorse, but despite its positive features it embodies the technological level of early sixties, as it is proven by its performance data. The shaft power is 1103^{+15} kW, which is not few, although the example of Mi-17 helicopter proves that some extra power would provide better handling. The specific net work (160–164 kJ/kg) and the thermal cycle efficiency (~22.5%) is especially thin today. With its 334 kg mass it is too heavy comparing to the present day engines.

Type: TV2-117A P_{shaft}= 1103 kW	**maximum net work**	**optimum**	**maximum therm. eff.**	TV2-117A	**available data**
comp. pressure ratio [-]	6,24	7,74	9,97	6,60	6,6
turb. entry temp. [K]	1168	1168	1168	1168	1148?
$\eta_{polc(0)}$ [-]	0,848	0,848	0,848	0,848	
η_{pole} [-]	0,848	0,848	0,848	0,848	
compressor exit temp. [K]	539,67	580,67	631,98	550,01	
exhaust pipe exit temp. [K]	808,16	771,13	729,34	798,46	
specific net work[kJ/kg]	162,532	159,794	150,016	162,331	
therm. cycle efficiency [%]	22,26	23,35	23,94	22,58	
η_{polc} [-]	0,8244	0,8226	0,8213	0,8239	
mass flow rate [kg/s]	6,786	6,903	7,353	6,795	6,8
medium diameter [m]	0,2269	0,2269	0,2269	0,2269	
first blade length [m]	0,0661	0,0673	0,0716	0,0662	
last blade length [m]	0,0198	0,0174	0,0157	0,0190	
cooling air [%]	1,20	1,20	1,20	1,20	
spec. fuel cons. [kg/kW*h]	0,374	0,357	0,348	0,369	0,369

Table 1. Placement of TV2-117A engine in the available range of compressor pressure ratio

Using my model, the available compressor pressure ratio ranges from 6,24, to 9,97. The actual take off pressure ratio of this engine is 6,6 which value is close to the lowest compressor pressure ratio (where the specific net work is the highest). It is not surprising, because the medium load helicopter claimed high power meanwhile the specific work was relatively small. So it was reasonable to earn as much specific net work as possible from the engine even at an expense of thermal efficiency (or specific fuel consumption). Surprisingly the compression and expansion polytrophic efficiency is quite reasonable suggesting that higher turbine entry temperature would result much better engine performance data. Presumable, the then Soviet industry technologically was not mature enough to produce suitable cooled and heat resistant turbine blades.

TV3-117VM turboshaft engine

The TV3-117 turboshaft engine was developed in 1974 as a component of power plants for Mi-24 and Mi-14 helicopters. Later it was installed on 95% of all helicopters designed by Mil and Kamov Engineering Centre. The power plants of these helicopters (except Mi-28) incorporate main gearboxes that were also developed by the Klimov Company (VR-14, VR-24, VR-252 and VR-80). Over 25,000 TV3-117 engines have been manufactured since the start of production. Their total time in operation exceeds 16 million hours. [3]

Type: TV3-117VM P_{shaft}= 1699 kW	**maximum net work**	**optimum**	**maximum therm. eff.**	TV3-117VM	**available data**
comp. pressure ratio [-]	7,18	9,12	12,20	9,45	9,45
turb. entry temp. [K]	1250	1250	1250	1250	1243
$\eta_{polc(0)}$ [-]	0,846	0,846	0,846	0,846	
η_{pole} [-]	0,846	0,846	0,846	0,846	
compressor exit temp. [K]	564,66	612,15	673,97	619,43	
exhaust pipe exit temp. [K]	845,37	802,86	753,63	796,76	
specific net work[kJ/kg]	195,464	191,957	178,719	190,852	
therm. cycle efficiency [%]	24,26	25,53	26,23	25,66	
η_{polc} [-]	0,8260	0,8242	0,8226	0,8240	
mass flow rate [kg/s]	8,539	8,695	9,339	8,745	8,75
medium diameter [m]	0,2231	0,2231	0,2231	0,2231	
first blade length [m]	0,0846	0,0862	0,0925	0,0867	
last blade length [m]	0,0230	0,0199	0,0176	0,0197	
cooling air [%]	1,45	1,45	1,45	1,45	
spec. fuel cons. [kg/kW*h]	0,343	0,326	0,318	0,325	0,325

Table 2. Placement of TV3-117VM engine in the available range of compressor pressure ratio

This engine has similar element efficiencies and losses like its older brother and as I mentioned earlier these efficiency values are quite reasonable in the turboshaft engine category. Not surprisingly, the higher turbine inlet temperature immediately increased and opened the possible compressor pressure ratio range increasing the thermal cycle efficiency with 3% and the specific net work with about 30 kJ/kg. The engine take off pressure ratio is much closer to the upper limit providing relatively higher thermal efficiency reducing the specific net output work.

T58-GE-100 turboshaft engine

The T58 turboshaft engine was born in 1953 with the award of a $3 million contract from the U.S. Navy. Under the terms of the contract, GE was to develop the XT-58 "baby gas turbine" that the company had proposed as a powerplant for helicopters. The engine was to weigh 400 pounds and was to produce 800 shaft horsepower (596 kW). The T58 was developed for helicopter use and was the first turbine engine to gain FAA certification for civil helicopter use (CT58 is the civilian version). This is the engine that would power the Sikorsky Sea King helicopter that recovered the Apollo astronauts and still powers Marine One - the helicopter of the U.S. president since the Kennedy administration. Considering the history of the T58 turboshaft engine family, it is one of the most reliable helicopter engines in the world. [5] The curiosity of the current version I examined, that the shaft power and the specific fuel consumption practically the same, like its contemporary TV2-117A engine's data.

Type: T58-GE-100 P_{shaft}= 1118 kW	**maximum net work**	**optimum**	**maximum therm. eff.**	T58-GE-100	**available data**
comp. pressure ratio [-]	6,52	8,03	10,22	8,40	8,4
turb. entry temp. [K]	1269	1269	1269	1269	1018???
$\eta_{polc(0)}$ [-]	0,824	0,824	0,824	0,824	
η_{pole} [-]	0,824	0,824	0,824	0,824	
compressor exit temp. [K]	558,04	600,00	651,98	609,47	
exhaust pipe exit temp. [K]	885,65	848,27	806,31	840,30	
specific net work[kJ/kg]	179,977	177,262	167,667	175,996	
therm. cycle efficiency [%]	21,49	22,44	22,95	22,59	
η_{polc} [-]	0,8004	0,7986	0,7972	0,7983	
mass flow rate [kg/s]	6,212	6,307	6,668	6,352	6,35
medium diameter [m]	0,2126	0,2126	0,2126	0,2126	
first blade length [m]	0,0646	0,0656	0,0693	0,0661	
last blade length [m]	0,0191	0,0169	0,0153	0,0166	
cooling air [%]	1,51	1,51	1,51	1,51	
spec. fuel cons. [kg/kW*h]	0,388	0,371	0,363	0,369	0,369

Table 3. Placement of T58-GE-100 engine in the available range of compressor pressure ratio

It is obvious, that the worse compressor and turbine polytrophic efficiency of this engine was compensated by higher turbine entry temperature. The available turbine entry temperature (1018 K), by the reference [6], is surely out of the acceptable range.

MTR390-E turboshaft engine

This engine family was developed in cooperation of Turbomeca, Rolls-Royce and MTU and used on the French-German escort and anti-tank Tiger helicopter. Production engines of the MTR390-2C basic version were delivered from 2002 to mid-2010 (of which 242 for the Franco-German tiger program and 51 for the Australian Armed Reconnaissance Helicopter program).

Type: MTR390-E P_{shaft}= 1094 kW	**maximum net work**	**optimum**	**maximum therm. eff.**	MTR390-E	**available data**
comp. pressure ratio [-]	9,45	12,24	16,73	14,00	14
turb. entry temp. [K]	1627	1627	1627	1627	
$\eta_{polc(0)}$ [-]	0,815	0,815	0,815	0,815	
η_{pole} [-]	0,815	0,815	0,815	0,815	
compressor exit temp. [K]	645,95	708,40	789,99	742,70	
exhaust pipe exit temp. [K]	1 083,57	1029,77	967,49	1002,69	
specific net work[kJ/kg]	315,233	310,116	290,783	303,466	
therm. cycle efficiency [%]	26,13	27,40	28,15	27,83	
η_{polc} [-]	0,7795	0,7759	0,7721	0,7740	
mass flow rate [kg/s]	3,470	3,528	3,762	3,605	3,6
medium diameter [m]	0,1610	0,1610	0,1610	0,1610	
first blade length [m]	0,0476	0,0484	0,0517	0,0495	
last blade length [m]	0,0113	0,0097	0,0084	0,0091	

cooling air [%]	2,58	2,58	2,58	2,58	
spec. fuel cons. [kg/kW*h]	0,319	0,304	0,296	0,299	0,299

Table 4. Placement of MTR390-E engine in the available range of compressor pressure ratio

In total, the order book stands at 445 engines, including exports. In cooperation with Spain's ITP, MTU developed an enhanced engine version (MTR390-E), which delivers 14 percent more power. [7] I chose to analyse this last version of MTR390. This engine is much newer than the above examined turboshaft engines. The designers follow totally different way to achieve good performance indicators. What is strange firstly, the poor compression and turbine polytrophic efficiency. Unfortunately I could not find any information about the turbine entry temperature. Of course, this fact can cause some uncertainties during the analyses, but considering the very low mass flow rate, this very high specific net work is achievable by extreme high (in turboshaft category) turbine entry temperature. The law mass flow rate resulted small weight and size, but the small size had negative effect on polytrophic efficiencies.

T800-LHT-801 turboshaft engine

The next-generation T800 turboshaft/turboprop family has been developed by the Light Helicopter Turbine Engine Company (LHTEC), a 50:50 partnership between Rolls-Royce and Honeywell. Setting a new benchmark in engine performance and reliability, the T800 was originally developed to power the US Army's Boeing Sikorsky RAH-66 Comanche stealth helicopter. Military qualification and civil certification of the baseline T800-LHT-800 was completed in 1993. A 17 percent growth version, designated T800-LHT-801, has since been developed, first flown in the Comanche in June 2001. A further 7 percent 'throttle push' upgrade has resulted in the T800-LHT-802 engine, delivering 1199 kW. [8]

Type: T800-LHT-801 P_{shaft}= 1116 kW	**maximum net work**	**optimum**	**maximum therm. eff.**	T800-LHT-801	**available data**
comp. pressure ratio [-]	8,92	11,67	16,41	15,00	15
turb. entry temp. [K]	1444	1444	1444	1444	
$\eta_{polc(0)}$ [-]	0,849	0,849	0,849	0,849	
η_{pole} [-]	0,849	0,849	0,849	0,849	
compressor exit temp. [K]	613,13	672,39	754,00	731,90	
exhaust pipe exit temp. [K]	944,82	892,84	830,02	846,23	
specific net work[kJ/kg]	275,781	270,760	250,649	257,353	
therm. cycle efficiency [%]	27,54	29,05	29,99	29,85	
η_{polc} [-]	0,8141	0,8105	0,8068	0,8076	
mass flow rate [kg/s]	4,228	4,306	4,652	4,531	4,53
medium diameter [m]	0,1862	0,1862	0,1862	0,1862	
first blade length [m]	0,0502	0,0511	0,0552	0,0538	
last blade length [m]	0,0120	0,0102	0,0088	0,0091	
cooling air [%]	2,03	2,03	2,03	2,03	
spec. fuel cons. [kg/kW*h]	0,302	0,287	0,278	0,279	0,279

Table 5. Placement of T800-LHT-801 engine in the available range of compressor pressure ratio

I chose from this turboshaft/turboprop family the T800-LHT-801engine. Analysing this engine I get one of the most well balanced performance indicators. Specific net output work and thermal cycle efficiency is reasonable while it is achieved by relatively low turbine inlet temperature. It is the result of good polytrophic compression and expansion efficiency. Although in this case I also did not have a reference turbine inlet temperature to compare it with the calculated value, but the available data allowed good approximation with an acceptable turbine entry temperature.

RTM322-09/1 turboshaft engine

The RTM322 is a collaborative turboshaft engine between Rolls-Royce and Turbomeca. The initial production order for the RTM322 to power the Royal Navy Merlin HM Mk1 helicopters was received in 1992, and the type entered service in 1998. The RTM322 also powers the EH101 HC Mk3 utility helicopters operated by the Royal Air Force, and the AgustaWestland WAH-64 Apache attack helicopters now in service with the UK Army. Denmark, Portugal and the Japan Maritime Self Defense Force have also selected the RTM322 to power their EH101's. The RTM322 also powers the European NATO Helicopter Industries twin-engined NH90 maritime and tactical transport helicopter. Over 1,500 RTM322s have been ordered or are on option. [10]

Type: RTM322-09/1 P_{shaft}= 1799 kW	**maximum net work**	**optimum**	**maximum therm. eff.**	RTM322-09/1	**available data**
comp. pressure ratio [-]	10,36	14,01	20,81	14,70	14,7
turb. entry temp. [K]	1507	1507	1507	1507	
$\eta_{polc(0)}$ [-]	0,857	0,857	0,857	0,857	
η_{pole} [-]	0,857	0,857	0,857	0,857	
compressor exit temp. [K]	636,58	704,54	801,98	715,79	
exhaust pipe exit temp. [K]	956,25	896,85	823,48	887,74	888
specific net work[kJ/kg]	319,362	312,923	285,706	310,768	
therm. cycle efficiency [%]	30,21	32,04	33,20	32,26	
η_{polc} [-]	0,8268	0,8230	0,8187	0,8224	
mass flow rate [kg/s]	5,633	5,749	6,297	5,789	
medium diameter [m]	0,1929	0,1929	0,1929	0,1929	
first blade length [m]	0,0646	0,0659	0,0722	0,0663	
last blade length [m]	0,0137	0,0115	0,0096	0,0112	
cooling air [%]	2,22	2,22	2,22	2,22	
spec. fuel cons. [kg/kW*h]	0,276	0,260	0,251	0,258	0,258

Table 6. Placement of RTM 322-09/1 engine in the available range of compressor pressure ratio

From this engine family I have chosen the RTM322-09/1 version to analyse its performance data. Its shaft power is 1799 kW, which represent the upper middle category of helicopter turboshafts. Its compression and expansion polytrophic efficiency is the best I have found. Unfortunately, I did not have information about the mass flow rate and turbine entry temperature but I had the temperature of exhaust gas. Using the three available parameters I got reasonable results. Taking into consideration of all performance indicators this engine shows the best values of all examined turboshafts.

LM 2500 turboshaft engine

The LM2500 marine gas turbine is GE's most widely applied gas turbine. It powers more than 400 ships in 30 world navies, fast ferries, coast guard cutters, supply ships and cruise ships. [4]

The LM2500 marine gas turbine is a simple-cycle, high-performance engine. The LM2500 gas generator consists of a 16-stage, 18:1 pressure ratio compressor with seven stages of variable stators and inlet guide vanes; a fully annular combustor with externally mounted fuel nozzles, and a two-stage, air-cooled high- pressure turbine which drives the compressor and the accessory gearbox. A six-stage, aerodynamically coupled low-pressure power turbine which is driven by the gas generator's high-energy exhaust gas flow drives the output shaft of the marine gas turbine. [11]

Type: LM2500 P_{shaft}= 24000 kW	**maximum net work**	**optimum**	**maximum therm. eff.**	LM2500	**available data**
comp. pressure ratio [-]	12,38	17,98	29,64	18,00	18
turb. entry temp. [K]	1504	1504	1504	1504	
$\eta_{polc(0)}$ [-]	0,872	0,872	0,872	0,872	
η_{pole} [-]	0,872	0,872	0,872	0,872	
compressor exit temp. [K]	652,98	734,27	855,91	734,51	
exhaust pipe exit temp. [K]	911,22	840,06	751,60	839,87	839
specific net work[kJ/kg]	350,686	341,511	300,536	341,451	
therm. cycle efficiency [%]	33,92	36,43	37,96	36,43	36
η_{polc} [-]	0,8610	0,8585	0,8547	0,8585	
mass flow rate [kg/s]	68,437	70,276	79,857	70,288	70,3
medium diameter [m]	0,6434	0,6434	0,6434	0,6434	
first blade length [m]	0,2351	0,2414	0,2744	0,2415	
last blade length [m]	0,0429	0,0340	0,0274	0,0341	
cooling air [%]	2,21	2,21	2,21	2,21	
spec. fuel cons. [kg/kW*h]	0,246	0,229	0,219	0,229	

Table 7. Placement of LM2500 engine in the available range of compressor pressure ratio

The last turboshaft engine is totally different in its size from any other earlier mentioned engine, although its structural arrangement is similar. The output shaft power is more than ten times higher than an average helicopter turboshaft engine. From the comparison clearly visible that the larger size is advantage and provides much higher compression and expansion polytrophic efficiency. In accordingly it shows the best performance indicators. The main reason is that the designers have to make much less compromise designing a large gas turbine engine than a small one.

Conclusion

In table 8 I collected the most important data of analysed turboshaft engines, which provides their easier comparison.

Having gone through the analysis process it is well demonstrated that the helicopter engines, like the other fields of aviation have gone through huge evolution. The increased compressor overall pressure ratio, turbine entry temperature, and the FADEC system (used by all new engines) improved their performance, although much less than we experience in other gas turbine engine categories. The main reason is that the average turboshafts provide about 250–2500 kW shaft power with 2–12 kg/s air mass flow rate. Accordingly their compressors are relatively small, which causes short rotor blade length, especially in rear stages (or last centrifugal stage). This effect has been heightened by the development trend to increase the specific net work decreasing the engine dimensions and weight. It is the reason that in some cases compressor and turbine polytrophic efficiency not significantly higher, what is more sometimes lower, although the engine is much younger. This fact considerably penalizes mainly the compressor polytrophic efficiency. [2] This fact means the compressor pressure ratio is usually not higher than ~15, and the resulted maximum thermal efficiency is less than 35%, while at bigger (new) gas turbine engines (where air mass flow is over 30 kg/s) the thermal efficiency is usually over 40%. Good example is the LM 2500, which does not achieve 40% thermal efficiency, but its thermal efficiency is considerably higher than the efficiency of much smaller turboshafts.

	TV2-117A	**TV3-117VM**	**T58-GE-100**	**MTR 390E**	**T800-LHT-801**	**RTM 322-01/9**	**LM 2500**
P_{sh} [kW]	1103	1699	1118	1043	1166	1799	24000
T_3 [K]	1168	1250	1269	1627	1444	1507	1504
$\dot{m}$ [kg/s]	6,8	8,75	6,35	3,6	4,53	5,79	70,3
π_{wh}-$\pi_{\eta t}$ π [-]	6,24-9,97 6,6	7,18-12,2 9,45	6,52-10,22 8,4	9,45-16,73 14	8,92-16,41 15	10,36-20,81 14,7	12,38-29,64 18
Poz. [%]	9,58	45,28	50,73	62,48	81,19	41,56	32,58
η_{pole} [-]	0,848	0,846	0,824	0,815	0,849	0,857	0,872
η_{polc} [-]	0,824	0,824	0,798	0,774	0,808	0,822	0,858
η_t [%]	22,58	25,66	22,59	27,83	29,85	32,26	36
SFC [kg/kWh]	0,369	0,325	0,369	0,299	0,279	0,258	0,229
w_{net} [kJ/kg]	162,3	190,8	176	303,47	257,3	310,8	341,451

Table 8. Most important data of examined turboshaft engines

The better component efficiencies and the high pressure ratio and combustor exit temperature of the new generation RTM-322-01/9 presents the best overall features. This is clearly shows us that good performance indicators cannot be achieved only by increasing the compressor pressure ratio and turbine entry temperature. To keep the component efficiencies, especially compressor polytrophic efficiency as high as possible, has a same importance.

SUMMARY

In this paper I presented the theoretical background how the compressor pressure ratios of maximum specific net output work and thermal cycle efficiency can be determined. I demonstrated a thermal mathematical model, which is able to:

- define the above mentioned pressure ratios of gas turbine engines at any kind of *turbine inlet temperature and engine component efficiencies*;
- calculate the most important engine parameters of these pressure ratios and plus any other optional pressure ratio;
- draw specific net work and thermal cycle efficiency characteristic curves in temperature versus pressure ratio diagram;
- analyse primarily any kind of turboshaft engine, secondarily any kind of other gas turbine engine.

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