Development of an Expendable Turbojet Engine for the Propulsion of an Unmanned Aerial Vehicle

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This paper presents preliminary engineering methodology for the design of expendable turbojets in the range of 400 daN thrust. This class of engines are among biggest used for missiles, typical applications are Harpoon or Storm Shadow missiles and variety of target drones. The term "expendable" dictates different philosophy in design: simple, reliable and cheap are higher demands than performances. The objective was to analyze concepts of design and development employed in existing engines and according to this to present methodology of design followed with numerical example. Moreover, the aim was to show how to transform design in real construction of such engine followed with engineering considerations and logic conclusions.The main contribution of this work is that defines and explains steps required to design expendable turbojet engine, combining scientific and engineering approaches.

Keywords: *turbojet, development, design, expendable, engineering, methodology,uav*

1. INTRODUCTION

Expendable turbojet engines are used to power variety of unmanned air vehicles. Most of them are from size of 40 to 400daN of thrust which implies that their sizes are from 150mm to 400mm in diameter and from 4kg to 70kg of mass. Typically, their life is lasting only one mission and their design is subordinated to that fact. That means that they have to be simple and reliable rather than achieving long life and high performances such as aircraft engines. However, these engines are not so simple in design because lot of solutions which are proven at bigger aircraft engines can not be used in small scale. This paper suggests engineering metho–dology for design of such an engine with engineering considerations and conclusions followed with numerical example. Finally, the numerical design is transferred to real construction showing how to make real engine from numbers.

1.1 EXISTING ENGINE'S DESIGN

The first step is reviewing existing engines and their design. The purpose is to see the differences between engines and what are the pros and cons of different design solutions. Main focus was at engine architecture (axial or radial compressor or turbine, type of combustor, position of bearings), cycle parameters (pressure ratio and turbine inlet temperature), engine assembly, bearing lubrication, cooling and support stiffness, starting and ignition system. It is obvious that we should select typical representatives rather than to mention each of existing engines. Chosen engines for the analysis are:

1. TRI 60-1 MicroTurbo
2. J 402 Teledyne
3. NASA model turbojet engine

1.1.1 TRI 60-1 MICROTURBO

TRI 60 family of engines started in the beginning of the seventies and upon authors info half of the world was trying to make reverse engineering of this engine. One reason that it was showing great reliability while the design looks simple at first sight. This turbojet has three stage axial compressor, annular combustion chamber with twelve fuel injectors and a single stage axial turbine [1]. Rotor is supported at two bearings, front is before compressor and second before turbine, as can be seen from Figure 1. The bearings are lubricated with closed oil system. In the newer generation of TRI 60 family [2], closed oil system is replaced by spill out kerosene system. The engine is capable of delivering 350 daN maximum thrust at sea level static and SFC of 1.2 kg/daNh. Total weight of engine is 47 Kg, maximum diameter 330 mm and length of 780mm. Engine is started by windmilling. The variants of this engine are also in operation having slightly different performance and size parameters.

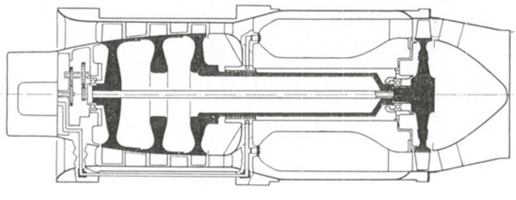

Figure 1. TRI 60-1 layout

1.1.2 J 402 TELEDYNE

The origin of architecture of this engine is from Turbomeca. French company sold license to Teledyne for bigger engine with recognizable design. J 402 is

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smaller derivative of that engine but with "expendable perfection". J 402 turbojet is turbojet designed originally for Harpoon missile [1]. Later it had few modified versions [3]. It's compressor consists of one stage axial plus one stage radial compressor, slinger type combustion chamber, with hollow turbine stator vanes practically part of combustor module and single stage axial turbine as shown at Figure 2. Engine is capable to produce 300 daN of thrust with specific fuel consumption of 1.2kg/daNh. Weight of engine is 44.5 kg with length of 760mm. This engine has some unique features: directly coupled high speed generator, starting system is pyro-starter which powers by impinging radial compressor blades and Turbomeca type slinger combustion chamber. Bearings, depending on variant, were lubricated by fuel or grease. Most of parts were produced by casting in order to minimize the cost.

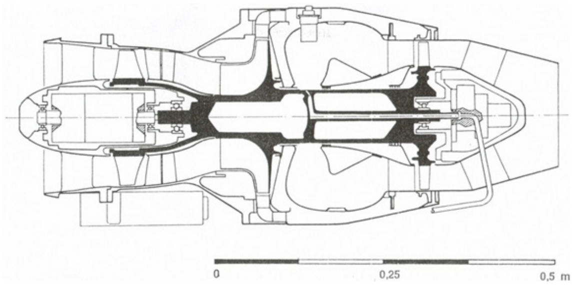


Figure 2. J 402 Teledyne layout

1.1.3 NASA SMALL TURBOJET ENGINE

It is experimental turbojet engine designed and fabricated to demonstrate the feasibility of low-cost concepts that were studied at NASA-Lewis [4]. The engine design was intended mainly for an expendable application. Engine has four stage axial compressor, annular combustor with 12 pressure swirl atomizers and single stage axial turbine. The engine has a maximum diameter of 290 mm and an overall length of 965 millimeters; it weighed about 59 kilograms. It was designed with a compressor pressure ratio of 4.0 and was expected to produce a thrust of 270 daN at sea-level static conditions.

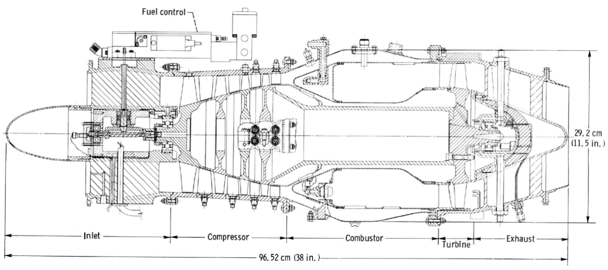


Figure 3. Simple, low cost expendable engine developed by NASA

The main focus was on simplicity of design, minimum number of components and machining processes required to reduce cost as much as possible. Therefore, most of main engine components were produced by casting. The other unique features of engine were all-welded compressor rotor main shaft, a simplified combustor assembly, an oil-mist bearing-lubrication system and a compressor-turbine rotor simply supported through a soft-mounted, two-main bearing system.

Figure 3 shows schematic for main components and their assembly for the engine. The NASA model engine may serve as reference engine for low cost and compact nature which is main focus for expandable engines.
Table 1 provides details of engine performance parameters and sizes of engines discussed above.

Table 1. Engines Specification

	TRI 60-1	J 402	NASA Engine
Application	Target drones and missiles	Target drones and missiles	Experimental engine for low cost concept validation
Thrust (daN)	350	300	270
SFC (kg/daNh)	1.2	1.2	1.1
Mass flow rate (kg/s)	5.84	4.35	4.4
PR	3.6	5.6	4.0
RPM	28500	41200	38000
TET (K)	1173	1283	1255
Compressor	3 stage axial	1xAxial + 1xRadial	4 stage axial
Turbine	Single stage axial, uncooled	Single stage axial, stator cooled	Single stage axial, uncooled
Combustor	Annular, pressure swirl atomizers	Annular slinger	Annular, pressure swirl atomizers
Weight (kg)	47 Kg	44.5 Kg	59 Kg
Dimensions (D x L, mm)	330 x 780	320 x 760	290x960

2. DESIGN METHODOLOGY

Proposed methodology consists of following steps:

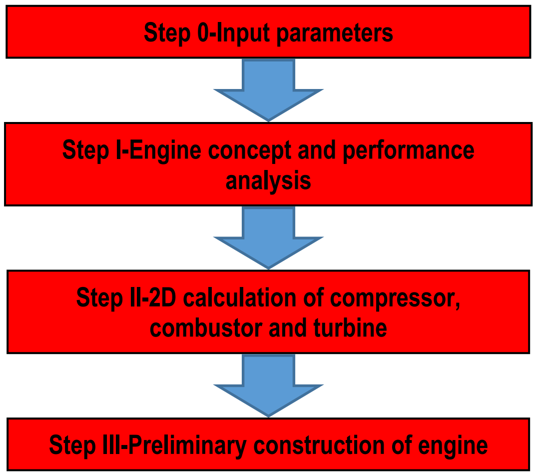
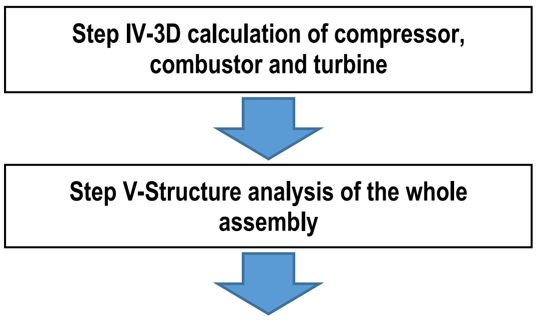


Figure 4. Design methodology-preliminary design block diagram



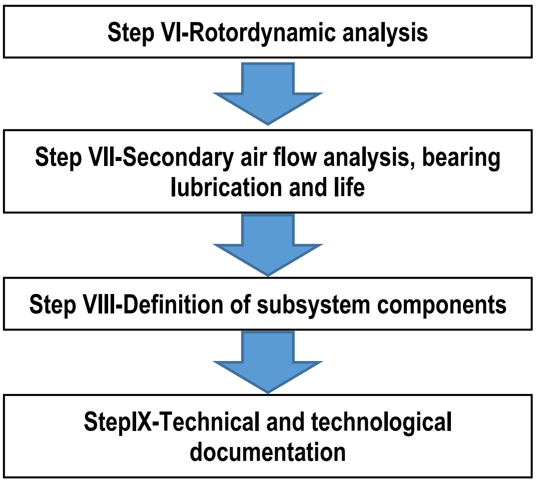


Figure 5. Design methodology-detail design block diagram

As proposed at Figure 4, preliminary design consists of four steps.

Zero step is definition of input parameters. For our numerical example we will define following input parameters i.e. technical requirements:Thrust>380daN, specific fuel consumption 1.15 kg/daNh, maximum diameter 330mm, maximum weight 60kg. Engine concept based on axial flow compressor

First step is definition of engine concept and performance analysis. Engine concept for this numerical example is chosen to be four stage axial compressor, straight flow combustion chamber and single stage axial turbine. Rotor is supported at two bearings, front before compressor, and rear before the turbine. Although detail design will give precise answer, it is expected that both bearings should be supported at elastic supports in order to avoid bending critical rpm. No damping will be used because of expendable nature of engine. Second step, after parameter were chosen from first step, is preliminary 2D calculation of compressor, combustion chamber and turbine in order to get engine preliminary layout.

In third, final step of preliminary analysis, it is possible to make preliminary construction based on geometry resulted from previous step. Next chapters are showing more detailed steps two and three.

3. PERFORMANCE ANALYSIS OF X-400 ENGINE

According to previous overview it is obvious that for expendable turbojet typical cycle parameters are in the range:
-Pressure ratio 3-5
-Turbine inlet temperature < 1200K (uncooled turbine)
According to previous experience with similar axial compressor and turbine efficiencies of 0.78-0.8 and 0.8-0.85 for compressor and turbine are chosen respectively. Combustion chamber pressure drop and efficiency is chosen to be 7% and 94% respectively. Design point calculation, parameter study and off design analysis is performed with licensed GasTurb 11 software [5]. In the following figures input and output data from GasTurb 11 software are presented.
As it is shown in Figure 6:
-Inlet corrected air mass flow rate chosen 7kg/s because after calculation it gives 380+daN of thrust.
-Intake pressure ratio is chosen 0.99. Usual range for subsonic intakes is 0.97-0.99.
-Pressure ratio is chosen 4.5 It is expected for four stage axial transonic compressor pressure ratio with average pressure ratio of 1.5 per stage.
-Burner exit temperature is chosen 1100K as maximum for existing materials and uncooled turbine is around 1200K. The reason is to leave certain amount of safety margin and to go on the side of lower SFC.
-Burner design efficiency is chosen 0.94. Usual range is 0.92-0.98.
-Power off-take is chosen 3kw for the generator.
-Burner pressure ratio is chosen 0.93. Usual range is 0.92-0.98.
-Turbine exit duct pressure depends on certain design and its usual range is 0.97-0.99.
*Labels in the GasTurb results refer to GasTurb 11 version.

```
             W          T          P           WRstd
Station    kg/s         K         kPa          kg/s
 amb                 288.15     101.325
  1        7.000     288.15     101.325
  2        7.000     288.15     100.312        7.071
  3        7.000     479.77     451.403        2.028
 31        6.790     479.77     451.403
  4        6.909    1100.00     419.805        3.258
 41        7.119    1083.06     419.805        3.332
 49        7.119     918.04     183.594
  5        7.119     918.04     183.594        7.013
  6        7.119     918.04     181.758
  8        7.119     918.04     181.758        7.084
Bleed      0.000     479.77     451.403
-------------------------------------------------------
P2/P1 = 0.9900  P4/P3 = 0.9300  P6/P5 0.9900
Efficiencies:   isentr  polytr   RNI      P/P
 Compressor     0.8000  0.8366  0.990    4.500
 Burner         0.9400                   0.930
 Turbine        0.8300  0.8149  0.875    2.287
-------------------------------------------------------
Spool mech Eff  0.9900  Nom Spd     29164 rpm
-------------------------------------------------------

hum [%]     war0       FHV      Fuel
   0.0    0.00000    43.323    JP-4

FN         =      3.81 kN
TSFC       =   31.2095 g/(kN*s)
FN/W2      =    544.33 m/s

Prop Eff   =    0.0000
eta core   =    0.2011

WF         =   0.11892 kg/s
s NOx      =   0.08918
XM8        =    0.9680
A8         =    0.0305 m²
P8/Pamb    =    1.7938
WBld/W2    =   0.00000
Ang8       =     10.00 °
CD8        =    0.9779
W_NGV/W2   =   0.03000
WCL/W2     =   0.00000
Loading    =    100.00 %
e45 th     =   0.83015
far7       =   0.01699
PWX        =      3.00 kW
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Figure 6. GasTurb design point calculation

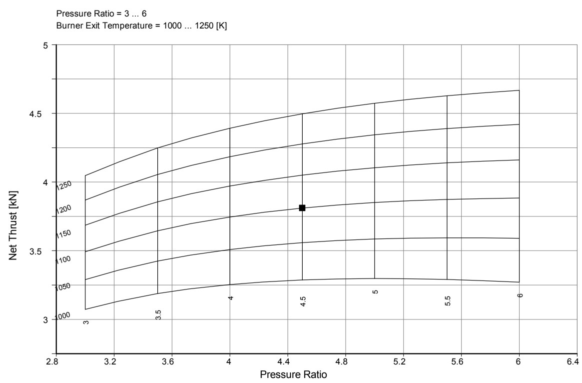

Figure 7. Parameter analysis: Thrust vs. pressure ratio for different temperatures

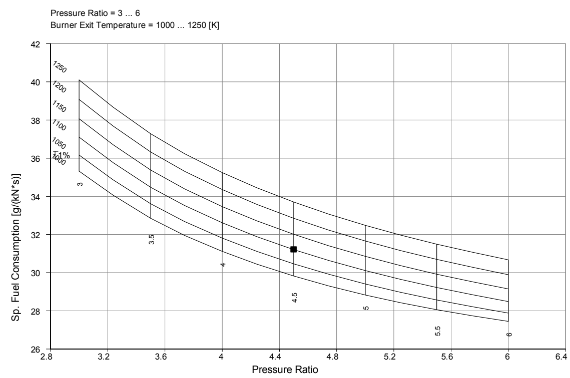

Figure 8. Parameter analysis: SFC vs. pressure ratio for different temperatures

In the parametric analysis pressure ratio and turbine inlet temperature were used as a parameter with other values being constant. Parameter analysis comments are:

-In thrust vs. pressure ratio at Figure 7 is evident that maximum thrust of 470daN can be achieved at pressure ratio 6. At that pressure ratio specific fuel consumption is 1.08. Black square is denoted design point. It means that increasing pressure ratio from 4.5 to 6 and temperature from 1100 to 1250K will increase thrust for about 24% i.e. it would be needed to have five comp–ressor stages to get benefit of 24% in thrust and 3.5% in specific fuel consumption. Five instead of four stages means higher cost, mass, longer compressor shaft and etc. From standpoint view of turbojet for unmanned vehicle five stages are maximum. Moreover, it is interesting to analyze solution with three stages i.e. pressure ratio 3.6 and temperature 1200K. In that case thrust is practically thesame and specific fuel consum–ption higher for 16%, but construction, mass and costs are the benefit.

-Design point of pressure ratio of 4.5 and tem–perature of 1100K is chosen from following conside–rations: from technical specifications compressor maximum number of stages is 4, and specific fuel consumption should be <1.15 kg/daNh. If we assume pressure ratio per stage of 1.5 the overall pressure ratio would be 5.1. Temperature of 1100K is chosen to leave margin in the case of lower efficiency then designed but to have reasonable size (lowering temperature will increase air flow rate and consequently dimensions).

-In specific fuel consumption vs. pressure ratio for different turbine inlet temperatures at Figure 8 is evident that change in temperature for 100K increases thrust for 10% and specific fuel consumption for 5%. However, from limitations because of material of turbine and uncooled turbine blade maximum temperature of 1200K can be exceeded only for short period of time, i.e. for some special short duration application.

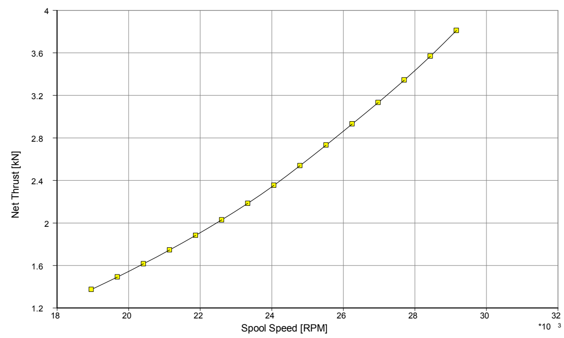

Figure 9. Off design analysis: Thrust vs. spool speed sea level static

After parametric analysis it was performed Off-design analysis. For the off-design calculation are used compressor and turbine maps from previous designed compressors. At this stage the compressor and turbine maps are not known, so it was used most similar case: compressor designed with same program, with same type of airfoils and distribution along blade height, and with almost same turning angles, pressure ratios per stage and efficiencies. Later, in detailed design compressor and turbine map will be calculated with designed compressor and turbine maps. Thrust and specific fuel consumption are shown as a function of rpm at Figures 9 and 10 respectively, while Thrust and fuel flow rate are shown as a function of flight Mach number for different altitudes at Figures 11 and 12.

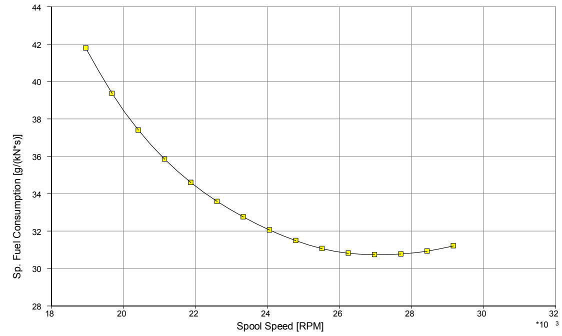

Figure 10. Off design analysis: SFC vs. spool speed sea level static

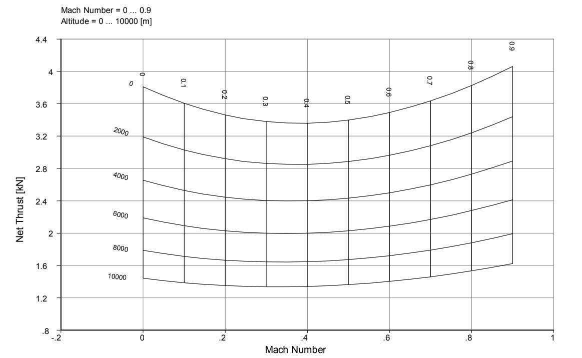

Figure 11. Off design analysis: Thrust vs. Mach number for different altitudes

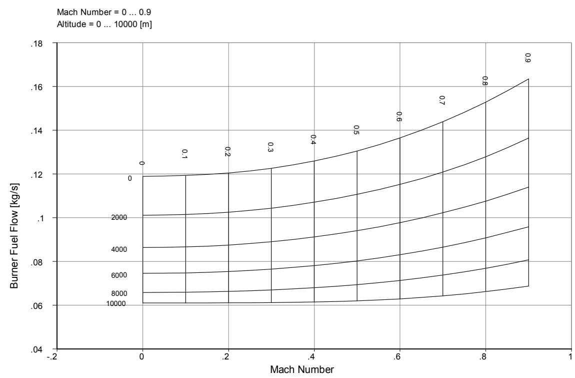

Figure 12. Off-design analysis: Burner fuel flow vs. Mach number for different altitudes

3.1 2D Design of compressor

Here is first presented fast hand calculation to preli–minary estimate the size of compressor. Compressor preliminary sizing is related to some guide parameters related to existing designs further related to existing technology and materials. These parameters are:

-Air mass flux

$$G = \frac{m_a}{K_g \times A} \tag{1}$$

where G is air mass flux (kg/s/m2), m_a is air mass flow rate (kg/s), Kg is blockage factor wich includes effect of

boundary layer on flow cross section and A is flow area (m^2).
-Tip speed or tangential velocity at the tip of rotor blade

$$V_t = R_t \times w \tag{2}$$

where V_t is tip speed (m/s), R_t radius at the rotor tip (m) and
w rotor angular velocity (rad/s).
-First stage rotor root (hub) to tip radius ratio

$$\frac{R_r}{R_t} \tag{3}$$

where R_r is the radius at inlet of first stage rotor root (m).

Typical values which are used according to comp–ressor theory, experience and from reference [4].

G =199 *kg/s/m2*

Vt =383 *m/s*

Rr/Rt =0.55

Kg =0.985

Generally, the greater value of G gives smaller dimension and lower efficiency. Tip speed is related to steel which will be intend to use for fabrication. If titanium is used it can be up to 500m/s. Also, sometimes the problem is not compressor stress but turbine which is rotated with same rpm but at higher temperatures. Root to tip radius ratio is related to desire to have maximum cross section and minimum tip radius. Again, stress of the blade is limiting factor, sometimes blade vibrations. Blockage factor is effect of boundary layer on geometrical cross section. The bigger the absolute size (bigger engine), the bigger is blockage factor or smaller is the effect of boundary layers.

Below is shown and approximate calculation and then calculation with EDePro company program.

$$G = 199\frac{kg}{s}/m^2 = \frac{m_a}{K_g \times A} = \frac{m_a}{K_g \times R_t^2\left(1-\left(\frac{R_r^2}{R_t^2}\right)\right)\pi} \tag{4}$$

$$R_t = \sqrt{\frac{m_a}{K_g r \times G \times \left(1-\left(\frac{R_r^2}{R_t^2}\right)\right)\pi}} = 127.7\text{mm} \tag{5}$$

$$R_r = 0.55 \times R_t = 70.2\text{mm} \tag{6}$$

$$Vt = 383m/s = Rt * w \tag{7}$$

$$rpm = w * \frac{30}{\pi} = \frac{Vt}{Rt} * \frac{30}{\pi} = 28640 rpm \tag{8}$$

where m_a=7kg/s according to performance calculation.

If we assume axial velocity at the compressor inlet and outlet ratio of 1.07 (which is usually in range of 1-1.1) we may approximate compressor exit inner radius (while outer radius is constant):

$$V_i = \frac{G}{\rho_i} = \frac{199}{1.04} = 191m/s \tag{9}$$

$$V_0 = \frac{m_a}{K_g \times \rho_0 \times A_0} = 178.5\frac{m}{s} \to R_0 = 110\text{mm} \tag{10}$$

where Vi is axial velocity at compressor inlet, ρ_i density of the air at compressor inlet while V_o and ρ_i are corresponding values at compressor outlet.

The meridional width of the blades can be predicted by using parameter called blade density which is defined as a ratio of blade chord to blade pitch (distance between blades at certain diameter). If the blade density at average diameter is 1.6 and the number of the blades are 22 then we have:

$$\sigma = 1.6 = \frac{chord}{D_{av} \times \pi/22}, \to chord = 45\text{mm} \tag{11}$$

where σ is blade density and D_{av} average diameter.

From the other side, typical chord stagger angle is about 45 degrees so the meridional projection is:

$$width = chord \times \cos(45) = 32\text{mm} \tag{12}$$

We can now approximate all width with 32mm and axial gap of 5mm between each rotor and stator. Then meridional cross section is an circular arc between inner radius at compressor inlet and inner radius at comp–ressor exit with axial distance of eight times axial width plus axial gap i.e. 296 mm.

It is simplified calculation which shows how compressor meridional cross section can be estimated. More detailed is done by EDePro software for axial compressors [6], written in Fortran language which will be used in compressor design. Input and results from that program used for preliminary construction drawing are shown below. Typical input and output are shown in [7].

Input data can be divided into three groups:

Group I: inlet parameters, which are varying but not design parameters. These data are inlet total pressure, inlet total temperature and inlet flow angle.

Group II: parameters which are chosen from the desired engine performances. These data are air mass flow rate, angular velocity and relative radius at rotor root at inlet (last parameter only for the first stage, others are resulted fromprevious stage). To that group also belongs blockage factor, radial gap between blade and housing and axial gap rotor-stator.

Group III: design parameters. These data are varied in order to reach desired **pressure ratio (1.4-1.5), efficiency (0.78-0.82), diffusion factor (≈0.5**) and tip radius dimension (smaller possible). These values are rotor ($10\text{-}25^0$) and stator turning angles ($30\text{-}50^0$), angles of attack ($0\text{-}2^0$), relative thicknesses (4.5% for the rotor and 8% for the stator), solidities (1.4-1.6), number or the blades (according to resulted chord), ratio of axial velocities at exit to inlet (0.9-1.1) and air mass flux (190-210kg/s/m^2). It has to be mentioned that value of diffusion factor was used as criteria for surge limit. In line with this in the design of compressor have been introduced radial grooves above each of four stages according to [8]and[9].It is inexpensive and efficient solution.

Part of compressor design software output which is used for preliminary construction is shown below:

Stage 1:	
Pressure ratio:	1.50
Efficiency:	0.83
Tip radius (mm):	128
Root radius at rotor inlet (mm):	70.4
Root radius at stator inlet (mm):	80.6
Root radius at stage exit (mm):	87.1
Stage 2:	
Pressure ratio:	1.46
Efficiency:	0.83
Tip radius (mm):	128
Root radius at rotor inlet (mm):	87.1
Root radius at stator inlet (mm):	94.1
Root radius at stage exit (mm):	98.6
Stage 3:	
Pressure ratio:	1.46
Efficiency:	0.82
Tip radius (mm):	128
Root radius at rotor inlet (mm):	98.6
Root radius at stator inlet (mm):	102.4
Root radius at stage exit (mm):	106.2
Stage 4:	
Pressure ratio:	1.42
Efficiency:	0.82
Tip radius (mm):	128
Root radius at rotor inlet (mm):	106.2
Root radius at stator inlet (mm):	108.2
Root radius at stage exit (mm):	110.1

3.2 2D design of turbine

There are some guided parameters for preliminary sizing turbine as in case of compressor. These para–meters are:

Blade height to tip radius ratio defined as:

$$\frac{H}{R_t} \tag{13}$$

where H is blade height (m) and $R_t$radius at turbine blade tip (m).

According to existing data and the same reference as in compressor case usual values considering materials, stresses and performances are chosen:

Vt ≈ 454m/s and H/Rt = 0.24.

Having in mind that rpm is already defined we may approximate turbine size as:

$$R_t = \frac{V_t}{rpm \times \pi/30} = 151.4\text{mm} \tag{14}$$

$$H = 0.24 \times R_t = 36.3\text{mm} \tag{15}$$

$$R_r = R_t - H = 115.1\text{mm} \tag{16}$$

More detailed calculation is performed in order to estimate main turbine dimensions as tip and root diameter and width of blades. Design is done by EDePro company software for axial turbine [10]**.** Prog–ram input and results are listed below (arrow shows chosen angle):

CALCULATION OF SINGLE STAGE AXIAL TURBINE

designed with Dsr=const

INPUT DATA

Total temperature at turbine inlet	T3T(K)=	1100
Mass flow rate of gases	mg(Kg/s)=	7.12
Total pressure at turbine inlet	P3T(bar)=	4.20
Desired turbine power	Nt(KW)=	1430
Velocity ratio	Usr/Cad=	0.60
Axial velocity ratio	C2a/C1a=	1.42
Fuel to air ratio	q=mg/mv=	0.017
Starting stator exit angle	AL1(rad)=	0.471
Turbine rpm	PIK=	4.5
Turbine blade material density	Rm(Kg/m3)=	8000
Turbine life	t(h)=	50
Max relative thick. ofrot.blade	Cmaxr=	0.105
Rotor blade roughness	Ks(m)=	0.0005
Trailing edge rotor thickness	Der(m)=	0.0005
Relt.axial gap stator-rotor	Dea/br=	0.1
Relt.radial gap rotor-casing	Der/Hl1=	0.03
Designed angle of attack	irp(rad)=	0.02
Number of rotor blades	Zr=	54
Flow angle at stator inlet	Alf0(rad)=	1.570
Flow angle at rotor exit	Alf2(rad)=	1.570
Relt. max thickness of stator vane	Cmaxs=	0.125
Trailing edge stator thickness	Des(m)	0.0005
Number of stator vanes	Zs=	38
Stator vane roughness	Kss(m)=	0.0002

CALCULATION RESULTS

Adiabate exponent	Kapag=	1.315
Tangent. velocity at mean diam	Utsr(m/s)=	400.9
Mean turbine diameter	Dsr(m)=	0.2674
Critical velocity at stator throat	Lan1kr=	0.969
Critical velocity at rotor throat	Lanw2kr=	0.935

Characteristic angles

alfa1(st.)	28.00	29.00	30.00	31.00	32.00	33.00
alf1L(st.)	27.86	28.91	29.96	31.00	32.02	33.00
Bet1(st)	69.44	69.99	70.61	71.14	71.60	71.98
Bet1L(st)	68.29	68.85	69.46	69.99	70.45	70.83
Bet2(st.)	43.31	44.58	45.80	47.01	48.23	49.44
Bet2L(st.)	43.22	44.56	45.80	46.96	48.03	48.97

Velocity triangles

C1a(m/s)	266.3	278.4	290.5	303.1	316.3	330.1
C1u(m/s)	500.8	502.3	503.1	504.4	506.1	508.3
C1(m/s)	567.2	574.3	581.0	588.5	596.8	606.1
W1(m/s)	284.4	296.3	308.0	320.3	333.3	347.1
C2(m/s)	378.1	395.3	412.5	430.4	449.1	468.7
C2A(m/s)	378.1	395.3	412.5	430.4	449.1	468.7
C2U(m/s)	0.30	0.31	0.33	0.34	0.36	0.37
W2(m/s)	551.3	563.2	575.4	588.4	602.2	617.0
Lan1	0.944	0.956	0.967	0.979	0.993	1.008
Lanw1	0.50	0.52	0.54	0.56	0.58	0.61
Lan2	0.655	0.685	0.715	0.747	0.779	0.814
Qlan2	0.882	0.904	0.924	0.942	0.959	0.973
Lanw2	0.923	0.943	0.964	0.986	1.009	1.035

Rot	0.376	0.374	0.373	0.371	0.369	0.366

Gasdynamic parameters

Dels	0.970	0.969	0.968	0.967	0.965	0.963
Delr	0.956	0.952	0.949	0.944	0.940	0.934
P1t(bar)	4.07	4.07	4.06	4.06	4.05	4.04
p1(bar)	2.38	2.34	2.30	2.26	2.22	2.17
P1wt(bar)	2.74	2.73	2.72	2.71	2.70	2.69
T1(K)	967	963	960	956	952	948
T2t(K)	933	933	933	932	932	931
T2(K)	874	868	862	855	848	839
P2t(bar)	1.96	1.95	1.94	1.92	1.90	1.88
P2wt(bar)	2.62	2.60	2.58	2.56	2.54	2.51
P2(bar)	1.48	1.45	1.40	1.35	1.30	1.24
Pit	2.84	2.89	2.99	3.10	3.24	3.39
PITT	2.14	2.15	2.17	2.19	2.21	2.23

Geometrical parameters

Hrl2(mm)	38.3	37.7	37.1	36.7	36.3	36.2
Hrl1(mm)	37.4	36.2	35.0	34.0	33.1	32.3
Dv1(mm)	304.8	303.6	302.4	301.4	300.5	299.7
Dk1(mm)	229.9	231.2	232.3	233.4	234.3	235.1
Dv2(mm)	305.7	305.0	304.5	304.0	303.7	303.6
Dk2(mm)	229.1	229.7	230.3	230.7	231.0	231.2
Ts(mm)	22.09	22.09	22.09	22.09	22.09	22.09
Hl1/as	3.6	3.4	3.2	3.0	2.8	2.7
Bs(mm)	30.7	30.9	31.1	31.2	31.3	31.4
Tr(mm)	15.55	15.55	15.55	15.55	15.55	15.55
Hl2/ar	3.6	3.5	3.3	3.2	3.1	3.1
Br(mm)	18.88	18.83	18.76	18.69	18.63	18.56

Performances

Fi	0.975	0.975	0.974	0.974	0.974	0.973
Psi	0.965	0.964	0.963	0.962	0.961	0.960
Etad	0.912	0.906	0.901	0.896	0.890	0.884
Ets	0.964	0.965	0.965	0.965	0.965	0.965
Et	0.862	0.856	0.852	0.847	0.842	0.837
Ni	3.03	3.08	3.13	3.17	3.19	3.21
Nt(KW)	1369	1346	1351	1354	1358	1364
Ntu(kJ/Kg)	192.3	189.1	189.9	190.3	190.9	191.6

3.3 2D design of combustion chamber

Volume of combustion chamber is calculated from Loading parameter [11],[12] which is shown at Figure 13. Volume of combustion chamber should be chosen according to loading factor of 50 kg/s/bar$^{1.8}$/m^3. Accor–ding to previous data that value is related to efficiency greater than 0.95. According to pressure of 4.5 bar, temperature of 480K and mass flow rate of the air of 7 kg/s volume calculated should be around 7 liters.

$$L = 50\frac{\frac{kg}{s}}{bar^{1.8}\times m^3} = \frac{m_a}{P^{1.8}\times V\times 10^{0.00145\times(T-400)}}, \quad (17)$$
$$\rightarrow V \approx 7\,liters$$

where air flow rate is m_a = 7kg/s, pressure and tempera–ture in front of combustion chamber are respectively $P=4.5bar$ and $T=480K$, known from design point calculation.

Material chosen for the combustion chamber flame tube is heat resistant steel in the form of sheet for further laser beam welding, type 310S.

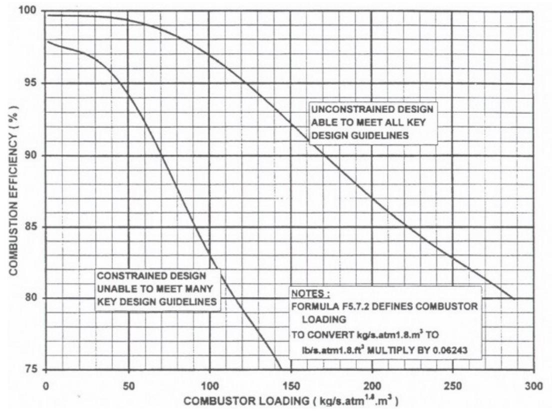

Figure 13. Combustor loading [11]

The initial air distribution is chosen as in [13], [14] and [15].The combustion chamber preliminary design was made according to recommendations from [16], [17]and [18] and initial shape is shown at the Figure 14.Chosen type of atomizer is pressure swirl and having in mind desired droplets less than 50 microns minimum pressure drop should be 7 bars. Commercially available atomizers would have orifice around 1 mm with fuel JET A1 or similar [19]. Preliminary calculations of combustor processes could be performed as in [20].

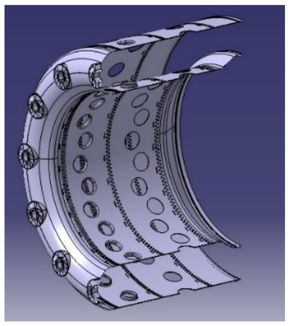

Figure 14. Initial model of combustion chamber

According to previous 2D calculations of comp–ressor, turbine and combustion chamber an simple 2D layout could be made and it is shown at Figure 15.

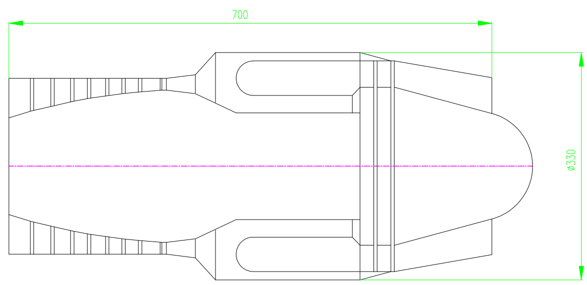

Figure 15. First engine layout after 2D calculations of compressor, combustor and turbine

4. PRELIMINARY CONSTRUCTION OF THE ENGINE

The engine construction is made with following prin–ciples: It has two supports, front, in the intake, and rear, before the turbine, [21]. Both supports are intended to be elastic due to further rotordynamic analysis [22]. The rotordynamic model is shown at the Figure 16.

Rear support is in the diffuser arm and it is connected to turbine stator with elastic connection in order to minimize influences from turbine stator side due to non-uniform temperature field. The compressor housing is strengthened with ribs in order to increase bending modes. Compressor discs and turbine shaft are connected with bolts, no welding is intended for rotating parts. Combustion chamber is annular type and it is intentionally left space between diffuser arm and liner, so either volume of liner or strength of the arm can be increased related to future analysis. Combustor is not needed special materials because engine is intended for short life. Starting will be possible with windmilling, configuration with axial compressor is superb for this option comparing to radial compressor [9], and by air or pyro impingement at turbine rotorblades [23]. That's why an zone is separated above turbine rotor, as shown at figure below. Lubrication of bearings is intended to be with fuel i.e. part of the fuel, usually around 2%, will be directed to the bearings from fuel system. However, air should be used for transporting this fuel and further cooling of bearings. Due to this, secondary air flow, for front bearing, is proposed to enter from atmosphere, through ribs and will be entering to main flow in front of compressor. It is possible because of vacuum gage pressure in front of compressor rotor. Air for rear bearing is extracted after compressor and it is entering the main flow in front of turbine rotor. That air is also cooling turbine disc. Due to higher air temperature after compressor, amount of fuel for bearing will be slightly increased to maintain temperature of around 180 degrees Celsius. Proposed construction is shown at Figures 17 and 18.

Proposed materials are [24]:

-Compressor rotors steel 17-4PH, proposed production with investment casting.

-Compressor stators 1,2 and Aluminum C355, proposed production with investment casting.

-Compressor stator 4. Stage steel 316 or 310S, proposed production with investment casting.

-Intake, Compressor casing, Diffuser, diffuser arm and all other except mentioned AISI 4140 steel, proposed production with investment casting or CNC machining.

-Combustion chamber liner and casing, nozzle steel 310S, proposed production with laser cutting and welding.

-Turbine rotor and stator Inconel 713LC (uncooled), proposed production with investment casting.

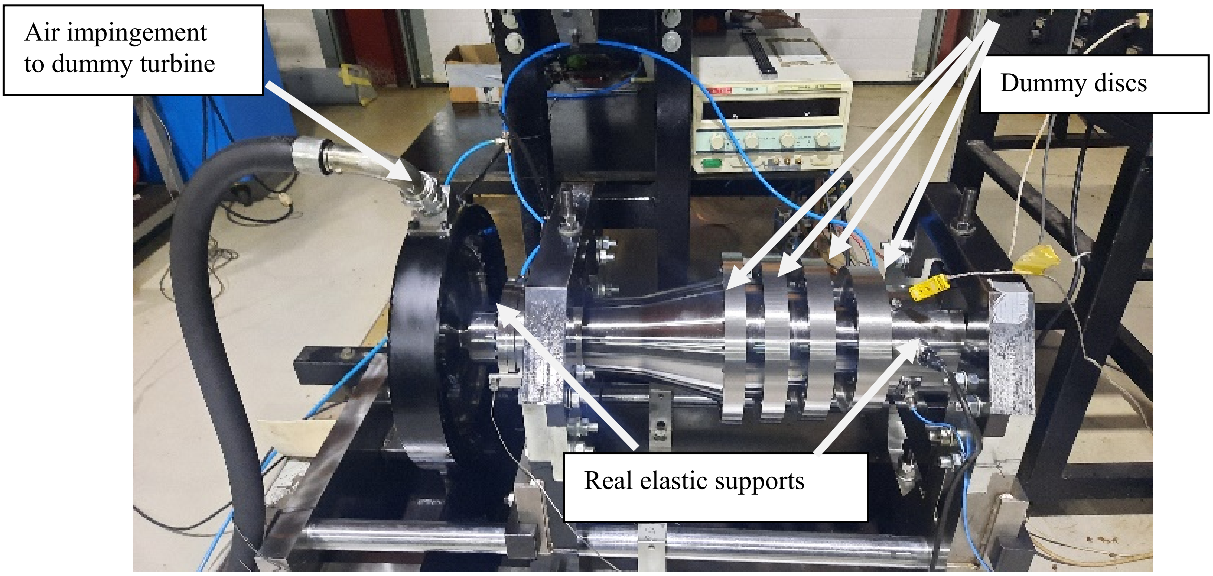

Figure 16. Rotordynamic test of X-400 engine rotor

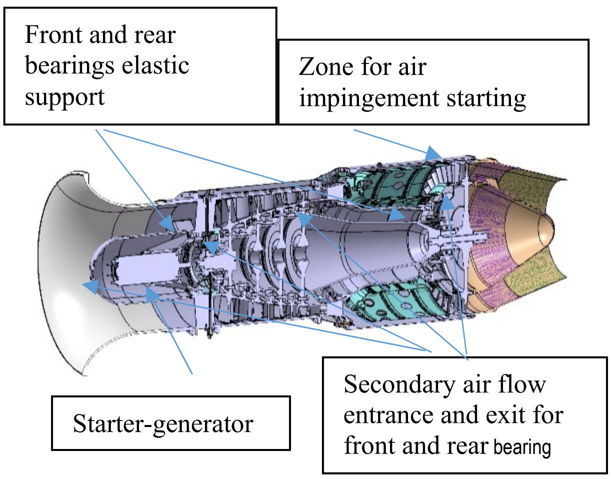

Figure 17. 3D assembly of X-400

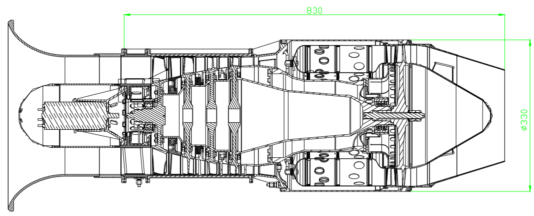

Figure 18. X-400 2D cross section

Figure 19. X-400 produced rotor

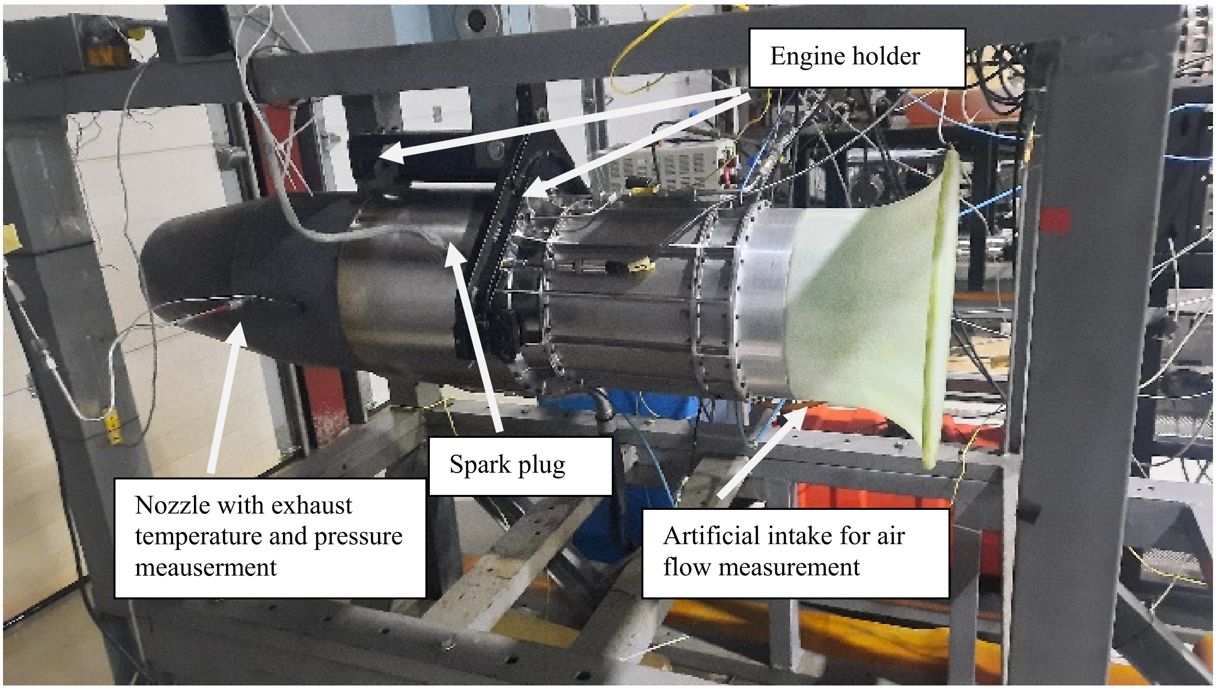

Figure 20. X-400 first prototype at test stand

According to presenting methodology the X-400 engine prototype is produced and the time of writing this paper the first prototype is at test stand. Photos of the X-400 engine produced rotor and engine at test stand areshown at the Figures 19 and 20.

5. CONCLUSION

This study proposes preliminary engineering metho–dology for design of expendable turbojet engine in thrust range of 400 daN. Methodology is followed with numerical example in order to engineeringly prove proposed steps. Introducing term "engineering" implies that procedure should be clear and easy to use but also on the other hand to allow designer its own choice and direction of design. It was shown connection between design of main components such as compressor, com–bustor and turbine and generation of construction of the engine. The process of designing the components is iterative from two main reasons: one is connecting with component performances and limitations while the otherare limitations implied by the engine design: for example volume of the combustor could be achieved with increasing the diameter of the engine and the length; maximum diameter of the engine is limitation for diameter of the combustor while length is not only increasing the length of the engine but could cause rotordynamic problems due to too long distance between bearings.

Finally, it was shown how to perform preliminary design of such engine using proposed methodology. The main contribution of this work is that defines and explains steps required to design expen–dable turbojet engine, combining scientific and engi–neering approaches. It also covers engine system design, taking into account range of achievable component performances. The proposed methodology was used in designing turbojet engines TJE-200 and TJE-400 in EDePro company, Belgrade, Serbia.

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РАЗВОЈ ЈЕДНОКРАТНОГ ТУРБОМЛАЗНОГ МОТОРА ЗА ПОГОН БЕСПИЛОТНИХ ЛЕТЕЛИЦА

Н. Давидовић, М. Милош, Б. Јојић

Рад презентује прелиминарну инжењерску методо–логију пројектовања турбомлазних мотора једно–кратне намене у рангу потиска до 400 даН.Ова класа мотора је међу највећима који се користе у беспи–лотним летелицама, типичне апликације су пројек–тили Харпун и Сторм Шадоу, као и разнолике мете. Термин “једнократни” намеће другачију филозофију у пројектовању: прост, поуздан и јефтин су захтеви вишег реда у односу на перформансе. Циљ је био да се анализирају концепти пројектовања и развоја примењени у постојећим моторима и да се према томе представи методологија пројектовања праћена нумеричким примером. Штавише, жеља је да се прикаже трансформација пројектовања у стварну конструкцију таквих мотора, праћена инжењерским разматрањима и логичним закључцима.Главни допринос овог рада је што дефинише и објашњава кораке потребне за пројектовање турбомлазног мотора једнократне намене комбинујући научни и инжењерски приступ.