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Design and Modeling of a Micro Turbojet Engine for UAV propulsion

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***Abstract:** This paper presents the design of major components of a turbojet engine that can produce thrust in the range of 4 kN suitable for medium altitude UAV. The paper presents the design and sizing of the turbojet engine components like engine inlet, compressor impeller, compressor diffuser, turbine blades, combustion chamber and its zones as well as the exit nozzle. It is concluded that to develop a thrust of 4 kN, the engine inlet required is 420 mm, while the compressor needs to have a diameter of 250 mm with a turbine of 19 vanes. The combustion chamber length is found to be 259 mm and radius 52 mm while the nozzle exit radius is 125 mm. The paper also presents a detailed estimation of mass of individual components and the total mass of the engine is estimated to be around 46.61 kg. The maximum thrust developed by the engine is found to be 4.4 kN at a rotational speed of 16,540 rpm, which is well above the expected output. Hence, this design can be used for fabrication of Turbojet engine that can develop a maximum of 4 kN thrust required to propel UAVS in the medium altitude range.*

I. INTRODUCTION

There has been a growing need in the past few decades to develop portable power plants for small UAVs and drones in the medium and high altitude range. This lack of appropriate propulsion system has motivated researchers across the globe to research on micro gas turbine engines as a feasible alternative to the currently used piston engines. The performances of piston engines are affected by altitude as there is a significant drop in the power output and increase in the specific fuel consumption with altitude. The use of single spool micro turbojet engines to power UAVs has attracted enormous attention due to its simplicity in construction, lower production, operation and maintenance costs. The efficiency of existing turbine engines needs to be increased to acceptable levels as the performance of these engines are hampered by small size, low pressure ratios, low Reynolds number and high rotational speeds. A detailed study on the design and analysis of individual components of turbine engines will pave way to better understanding and development of future turbine engines that can propel UAVs and drones. This paper provides detailed design of the major components of a single spool turbojet engine and the mass modelling of the designed engine to estimate its weight. The outcomes of this paper will enable designers to design and manufacture components at a faster rate leading to the elimination of the present disadvantages and effective utilization of this engine to propel drones and UAVs in the near future

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II. LITERATURE REVIEW

The availability of detailed design procedure for major components of turbojet engine from the reported literatures is very minimal. There is limited amount of literature available on the design and modelling of individual components of turbojet engine in the published literature. A few of the related findings is summarized as below.

Tarek Nada (2014) investigated different configurations of gas turbine engines and developed full numerical model for the engine. A comparison between single-spool and two-spool engine is made with free power turbine. The result shows that effect of turbine cooling flow is highly nonlinear and causes significant change in net work. If cooling is ignored and considering Turbine Inlet Temperature (TIT) of 1600K and pressure ratio 60, the overall work and efficiency are estimated to be 20% and 5% respectively. It is also seen that, to get maximum efficiency, the reheat combustor has to be placed at nearly 10%-20% of the expansion section[1]. Satish Peruri. et.al (2015) designed a micro gas turbine and performed the analysis on its turbine and compressor to find out the maximum stress and to estimate the total deformation. Using Titanium Alloy for turbine and compressor, the CFD analysis shows that the compressive yield strength and tensile yield strength is 930 MPa and 1070 MPa[2]. Jassim et. al (2015) performed studies to study the impact of equivalence ratio on the flame length and the speed of the jet engine modified from a automobile turbocharger. It is seen that there has been an increase in the speed of the engine when the mixing of primary and secondary air is proper, raising the temperature of the mixture[3]. V. Raviteja. et.al (2015) designed and fabricated a micro jet engine that can be used for future helicopter and hover boards. The results show that engine produces 60-70N thrust at 50,000 to 70,000 rpm. To minimize the weight of the engine single fuel pump lubrication is used. It is seen that, when the compressor and turbine diameter is increased, the thrust produced by the engine is also increased. The fabricated engine is found to cost less when compared to similar other engines[4]. D. Klein et.al (2015) modelled a Turbojet Gas turbine engine theoretically and computationally to study the performance of the engine at different operating conditions. From the results it is observed that, the error between the simulation and calculated performance is within 2% and confirms the efficacy of the models[5].

Mohamed Khalil et.al (2016) carried out analysis on a micro engine to analyse the performance of the engine and to derive the compressor map. The results of the analytical analysis shows that the thermodynamic cycle analysis using empirical formula is suitable for preliminary analysis of the performance of the MGT engine compressor and combustion chamber. Commercial software “GasTurb” was used for comparison of theoretical results[6].

Bela et. al (2016) in his modelling analysis concluded that the high pressure ratio in the compressor and high temperature at the inlet of turbine increases the overall efficiency of the engine. The study is useful in also arriving at the thermal efficiencies and shaft work net output of turbo shaft engines at various compressor pressure ratios and turbine inlet temperatures[7].Priyant Mark et.al (2016) designed an Annular Combustion Chamber for low bypass turbofan engine using Siemens NX 8.0, and the aerodynamics flow characteristics is simulated numerically by means of Ansys 14.5 software unit. The various dimensions of the combustor are calculated based on different empirical formulas. The result shows that SFC was reduced by regulating the temperature and the efficiency and pressure loss was achieved by a thin margin. It also shows that combustor chamber is shorter than other combustors[8]. Bhagawat Yedla et.al (2016) designed a micro gas turbine blade in solidworks based on CH10 airfoil and analysed using modified angles. CFD analysis carried out using ANSYS ICEM CFX concludes an error of 0.737% due to turbulence and shock losses suffered by the blade[9]. Prabjot Singh Virdi et.al (2017) designed the major components of a Turbojet Engine using CATIA and ANSYS software and fabricated the same. The impeller of the compressor was designed for producing 2.9 bar pressure at 8000rpm, and combustion chamber outer liner diameter and inner diameter was set to 320 mm and 240 mm while the overall length of combustion chamber was 284 mm[10]. Michal Czarnecki (2013) carried out the mass modelling of a micro gas turbine single spool turbojet in well-known Scheckling design layout (IR/IA). Weight of structures calculated on the basis of numerical model was 1.371 kg which represents 98% of the real structure. The largest percentage share in micro gas turbine design elements is sheet metal structures (outer casing and combustion chamber) [11].

III. RESULTS & DISCUSSIONS

In a turbojet engine, air is taken in through an opening in the front of the engine which is compressed to 3 to 12 times its original pressure in the compressor. Fuel is added to the air and burnt in the combustion chamber to raise the temperature of the fluid mixture to about 1,100 F to 1,300 F. The resulting hot air is passed through a turbine, which drives the compressor and the exhaust gasses passes through a nozzle outlet where thrust is produced because of the reactive forces of the exit stream. The main components of a turbojet engine are shown in Fig 1.

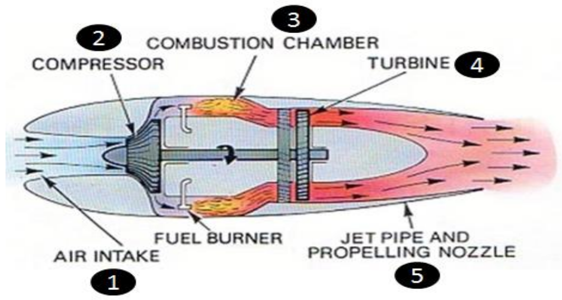

Fig. 1. Main components of a Turbojet Engine

IV. DESIGN CYCLE ESTIMATION

The cycle calculations are the basis for designing the different stages of the turbojet engine. Here the input condition is based on desired thrust and ambient temperature of similar systems. The initial values of input parameters considered in the analysis is given in Table 1.

TABLE I
INITIAL INPUT TO THE THEORETICAL DESIGN

Parameter	Initial Condition
Thrust required	4 kN
Pressure Ratio	2
Inlet Temperature	288 K
Initial Pressure	1.01 Bar
Max Turbine Inlet Temperature	1100 K
Compressor Efficiency	0.85
Turbine Efficiency	0.94
Mechanical Efficiency	0.99
Nozzle Efficiency	0.95
Combustion Efficiency	0.98

The work done by the compressor and turbine can be calculated, with the inlet and outlet temperature for the compressor as below.

$$\frac{W_{comp}}{m} = C_{pa}(T_{02} - T_{01}) = 1.005*72.5 = 72.86 \text{ kW}$$

$$\frac{w_t}{m} = \frac{w_{comp}/m}{\eta_m} = 73.59\text{KW}$$

Based on this, the turbine inlet and outlet parameters are estimated to be and P_{03}=1.939bar (Turbine Inlet), T_{04} = 1035.898K (Turbine outlet temp) and P_{04} = 1.53 bar

4.1 Design of Engine inlet

A bell mouth inlet is selected as it offers practically no air resistance. The duct loss is so small that is considered zero.

$$\text{Inlet diameter} = 2*D = 42 \text{ cm}$$

$$\text{Inlet width} = \frac{3}{4} * D = 15.75\text{cm}$$

4.2 Design of Compressor Impeller Wheel

The key component that makes a compressor centrifugal is the centrifugal impeller. Centrifugal impeller is used primarily for their suitability in handling small volume flows, but other advantages include a shorter length than an equivalent axial compressor and better resistance to Foreign Object Damage (FOD). The input parameters are assumed as: Power input factor (φ) = 1.04, Slip factor (σ) = 0.9, Rotational speed (N) = 16540 rpm = 275.66 rps, Overall diameter (D) = 0.25m, Eye tip diameter = 0.22m, Eye root diameter = 0.02m, Air mass flow (m) = 7.6 kg/s.

The impeller tip speed can be found out by:

$U = \pi * D * N = 216.508$ m/s

Power = $m*C_p*(T_{03}-T_{01})$ = 6125.676 KW

Peripheral speed at the impeller eye tip radius

At tip = $\pi*0.22*275.66 = 190.52$m/s

At root = $\pi*0.020*275.66 = 17.32$m/s

The angle is estimated as

$$\text{At root} = tan^{-1}\frac{165.826}{17.32} = \mathbf{84.03^o}$$

$$\text{At tip} = tan^{-1}\frac{165.826}{190.52} = \mathbf{41.03^o}$$

Now,

$\frac{P_2}{P_{02}} = (\frac{T_2}{T_{02}})^{\frac{\gamma}{(\gamma+1)}}$, and $T_{02} = 43.65+298 = 341.65$K

$T_2 = = T_{02} - \frac{C_2^2}{2*C_p}$ = (341.65-24.76) K = 316.89 K and

$\rho_2 = \frac{P_2}{R*T_2} = 1.286$ kg/m^3

The required area of cross section of flow in the radial direction at the impeller tip is:

$$A = \frac{m}{\rho_2 * C_{r2}} = 0.0356 \text{ m}^2$$

Hence the depth of impeller channel is found to be

$$\text{Depth of impeller} = \frac{A}{\pi*D} = \frac{0.0356}{\pi*0.25} = 0.0453 \text{ m}$$

4.3 Design of Compressor Diffuser

The diffuser manifold is responsible for the conversion of kinetic energy of motion into static pressure energy. The total compression is shared between the rotor and the diffuser, but the diffuser does not perform work on the air. The initial assumptions are: Radial width of vaneless space = 0.04m, Depth of diffuser passage = 0.045m, Number of vanes = 19 and Approx. mean radius of diffuser throat = 0.15 m.

Radius of diffuser vane leading edge (r_2) = over all radius + radius width = 0.165m

The surface area = $2*\pi*r_2$*depth of diffuser = 0.04665 m

Area of cross action of flow is radial direction = $2*\pi*0.165*0.0453 = 0.0469$m^2

$$\theta = tan^{-1}\left(\frac{122.20}{164.021}\right) = \mathbf{36.68^o}$$

As first approximation, we may neglect the thickness of diffuser vanes, so that the area of flow in the radial direction

Area = $2*\pi*0.15*0.045 = 0.0424$m^2

Direction of flow = $\tan^{-1}(136.30/180) = 37.13^o$

Total area of throat passage is = 0.0424*sin (37.13) = 0.0255 m^2. With 19 diffuser vanes the width of the throat in each passage of depth is therefore = 0.02m.

4.4 Turbine Design Estimations

The combustion gas flows first into turbine nozzle guide vane system, accelerating the gases in the direction of rotation of rotor followed by gas expansion. This increases the speed rapidly and at the same time flow strikes the turbine blades. Axial turbine which is chosen for the design in this paper is found to be more efficient.

Assuming, Mass flow (m) = 7.6 kg/s, Isentropic efficiency η_t = 0.9, Inlet temperature T_{01}= 1100K, Temperature drop $T_{01}-T_{03}$ = 43.66K, Pressure ratio P_{01}/P_{03} = 2.4455, Inlet pressure P_{01} = 2bar, Rotational speed (N) = 275.66 rev/s and Mean blade speed U= 216.508 m/s, the dimensions of the turbine are estimated to be as below:

$$\text{Power input } (\psi) = \frac{2C_p\Delta T_{os}}{U^2} = \frac{2*1.148*43.65*10^{\wedge}3}{(216.508)^2} = \mathbf{2.138}$$

When three dimensional effects are included, the reaction will increase from root to tip of the blades and a degree of reaction of 0.84 at the mean diameter might mean too low a value at the root. Throat area of nozzle required is

$$A_{ZN} = \frac{m}{\rho_2 * C_2} \text{ or } A_2 cos\alpha_2 = 0.0709*0.5952 = 0.0421\text{m}^2$$

$$A_3 = \frac{m}{\rho_3 * C_{a3}} = 0.1424\text{m}^2$$

The annulus area is given by

$A = 2*\pi* r_m * h = \frac{U_m * h}{N}$

The height and radius ratio of the annulus can be found from

$$h = \frac{AN}{Um} = \left(\frac{275.66}{216.508}\right) A$$

4.5 Combustion Chamber Design

In this study, an annular combustor is chosen as it uses the maximum space available within a specified diameter and associates minimum pressure loss. Combustion must be maintained in a stream of air moving with a high velocity in the region of 30-60 m/s.

Casing area is found to be

$$A_{ref} = [\frac{R}{2}(\frac{m_3 T_3^{0.5}}{P_3})^2 \frac{\Delta P_{3-4}}{q_{ref}} (\frac{\Delta P_{3-4}}{P_3})^{-1}]^{0.5}$$

$$A_{ref} = 8.3166 * 10^{-3} m^2$$

Liner area

$A_L = 0.66*A_{ref} = 5.489*10^{-3}m^2$

Annulus area

$(A_{an}) = A_{ref} - A_L$ = 0.0083166 – 0.0054890 = 0.0028276 m^2

Casing and liner diameter

$D_{ref} = A_{ref}$ = 8.3166*10^-3 m^2 = 0.1029 m

$D_L = A_L$ = 5.489*10^-3 m^2 = 0.08359 m

Liner length

$$L_L = \mathbf{0.2526m}$$

Primary zone length $L_{PZ} = \frac{3}{4} D_L$ and $L_{PZ} = \mathbf{0.062692\ m}$

Secondary zone length $L_{SZ} = \frac{1}{2} D_L$ and $L_{SZ} = \mathbf{0.041795m}$

Dilution zone length

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$L_{DZ} = D_L(3.83 - 11.83PF + 13.4PF^2)$ $L_{DZ} = \mathbf{0.1429389\ m}$

Swiler diameter $D_{sw} = \mathbf{0.02290m}$

Recirculation zone length

$L_{RZ} = 2 * D_{sw}$ and $L_{RZ} = \mathbf{0.04579m}$

Recirculation zone angle

$$\theta_{RZ} = cos^{-1}\left[\frac{D_L(D_L - 2D_{sw}) - (D_L - 4L_{RZ})\sqrt{D_L^2 - 4D_LD_{SW} + 4D_{SW}^2 - 8D_LL_{RZ} + 16L_{RZ}^2}}{2D_L^2 - 4D_LD_{SW} + 4D_{SW}^2 - 8D_LL_{RZ} + 16L_{RZ}^2}\right]$$

$\theta_{RZ} = \mathbf{14.4^0}$

Dome length

$L_{Dome} = \frac{D_L - D_{SW}}{2tan\theta_{RZ}}$ and

$L_{Dome} = \mathbf{0.03448m}$

The design of number of holes in the different zones of the combustion chamber is shown in Table 2.

TABLE 2
DESIGN OF HOLES IN THE COMBUSTION CHAMBER ZONES

Zone	Main Hole: No. of holes	Main Hole: Hole dia (m)	Cooling Hole: No. of holes	Cooling Hole: Hole dia (m)
Primary Zone	40	0.01	600	0.00395
Secondary Zo ne	20	0.008	480	0.0024
Dilution Zone	20	0.01	600	0.00329

4.6 Thrust Estimation and Sizing of Nozzle Exit

As the flow is assumed to be in subsonic region, a convergent nozzle is chosen in this design where a three dimensional squirting action causes the velocity to increase. However, in this nozzle, velocity cannot exceed the speed of sound.

$$A_t = \frac{m}{P_{04}}\sqrt{\frac{RT_{04}}{\gamma}\left(\frac{\gamma+1}{2}\right)^{\frac{\gamma+1}{\gamma-1}}}$$

$$A_t = 0.039722m^2$$

Subsequent, the diameter (D_t) = 0.22489m, the length of the nozzle (L_N) = 0.026 m and the angle = 30° are estimated. The velocity at exit is found to be V_{ext} = 582.586 m/s and the thrust produced by the engine to be F_t = 4.4 KN, which is more than the expected value.

4.7 Design of Shaft

The shaft transmits power from turbine to compressor. Usually circular in cross section, the shaft must have the ability to withstand torque during operation of flight. By determination of External Loads carried by gear shaft and the torque required to transmit power at the speed specified, the diameter of the shaft is estimated to be d = 76.01 mm and the length to be 504 mm. The overall dimensions of the single spool Turbojet engine intended to produce 4 kN thrust is shown in Fig. 2.

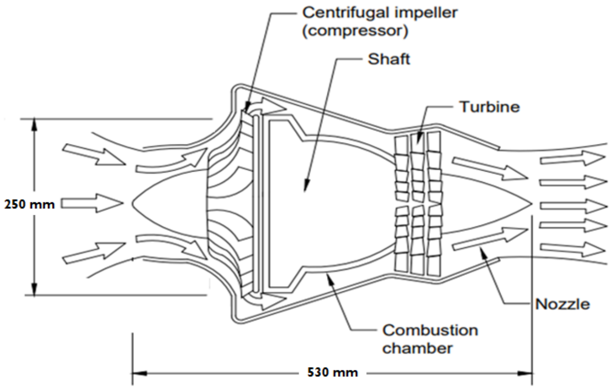

Fig 2. Dimensions of the Engine assembly

V. ESTIMATION OF MASS OF THE ENGINE COMPONENTS

The general form of the equation describing the weight of the micro gas turbine engine design comes from the sum of n-mass described by structural units (building blocks e.g. compressor, combustor, rotor, turbine, jet-pipe). The material assumed is Carbon/Carbon (C/C) with density to be 1800 kg/m^3. The mass of the individual components of the engine and the overall mass of the engine is estimated using standard relations and given in Table 3.

TABLE 3
ESTIMATION OF MASS OF TURBOJET COMPONENTS

Component	**Mass (kg)**	**Total Mass of the engine (kg)**
Compressor	4.359	46.61
Rotor Case	9.0711	
Combustor	11.1438	
Turbine	17.9	
Nozzle	1.32	
External Casing	2.923	

VI. CONCLUSION

The detailed design of the major components of a Turbojet engine is performed and the dimensions of the components are obtained. Also the weight of individual components of the engine is estimated and thus the total mass of the engine. The knowledge derived from this paper will enable engine designers to quickly design engine components and Turbojet engines in the thrust range of 4 kN that can power Medium and High Altitude drones in the near future.

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