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Conversion of Turbojet Engine Jet Cat P200to Turboprop Engine

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Abstract— This paper processes the conversion of Micro Turbojet Engine to Micro Turboprop Engine through describingthe Aerothermodynamics analysis of both engine cycles and design of a Low-pressure single-stage axial flow turbine suitable for driving the propeller using computational methods. The specifications of turbine are based on the typical geometric restrictionsand specifications of small gas turbine engine Jet Cat P200. Baseline design parameters such as flow coefficient, stage-loading coefficient are close to 0.95 and 1.04 respectively with maximum flow expansion in the NGV rows. In the preliminary design mode, the mean line approach is used to generate the turbine flow path. While the turbine blades design is achieved by considering three blade sections at hub mean and tip,using developed code to meet the design constraints based on free vortex law of blading. An average exit swirl angle of less than 5 degrees is achieved leading to minimum losses in the stage. Also, NGV and rotor blade numbers are chosen based on the optimum blade solidity. Detailed design to Free Turbine is introduced which include Bearing, Gearbox selection, shaft oil system. Also Micro Turboprop engine Performanceis evaluation

Index Terms—Small turbojet; Turbojet engine; Turboprop Engine; Free Turbine Design;single-stage axial flow turbine; Aerothermodynamics Design ;Turboprop Performance.

1 Introduction

During recent years, interest on small-sized gas-turbine engines has increased for both ground-based and vehicular uses. Small-size turbojet and Turboprop engines, in particular, are becoming attractive for their potential application on unmanned aerial vehicles (UAVs) because of their extremely high thrust-to-weight ratio [1]. A number of small turbojet design examples are available that develop about 200 N thrust.The lack of knowledge involves almost all the phases of the engine set-up and development:design, manufacturing, operation and testing of small engines. Such phases are regulated by differentconcepts rather that used for large aircraft propulsions and require tailored procedures. Also gas turbines are becoming increasingly complex; the apex ofengine efficiency seems to be fast approaching. In an attempt tofurther, improve gas turbines, new technologies are explored in anever-growing effort to increase thrust-to-weight ratio and minimizethrust specific fuel consumption (TSFC), while reducingthe overall cost of the engine development.

The design of such machines is inevitably influenced by their small size. For a millimeter/centimeter-scale gas turbine [2], designers have to deal with engineering challengescomparable with those, which characterize large conventional machines, plus the fact, thattraditional design criteria do not necessarily apply in the new design space. This involvesparticularly the aero-thermo-mechanical behavior of engine components, since relatively high operating temperatures, component pressure-ratios with lower efficiencies, and high rotational speeds of the core-assembly characterize the thermodynamic cycle.

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The overall design process of a gas turbine engine startswith a given set of specifications, which normally arises frommarket research or an understanding of specific customerrequirements. There are three principle steps involved inturbine aerodynamic design process: preliminary design usingmean line approach, through flow design, and blade design.

In this paper, an attempt is made to carry out theaerodynamic design and analysis of a single stage;Low-pressure turbine and design the blade profilesbased on the computational outcome with detail procedures toachieve proximity to realistic estimate of the aerodynamicparameters.In addition,the detailed design for free low-pressure turbine system is done.

2 Thermodynamic Cycle analysis of Baseline and Converted Engines

2.1 Turbojet Engine Jet Cat P200 Thermodynamic Cycle Analysis.

A non-ideal parametric cycle analysis is done for baseline turbojet engine as a first step in conversion to turboprop engine.The Jet Cat 200 engineis chosen due its relatively low fuel consumption of around 0.575kg/min at maximum engine speed and its relatively high-pressure ratio. The engine P 200 uses radial compressor with annular combustor with ring of 12 vaporizer tubes for fuel delivery and with the addition of a reasonably efficient axial turbine stage. Fig.1 show Scheme diagram for P 200 Jet Cat Engine

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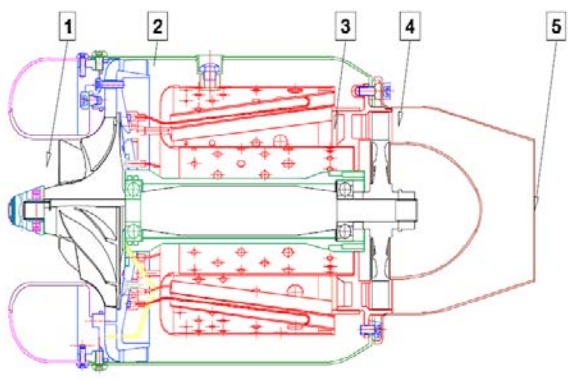

Fig. 1Scheme diagram for P 200 Jet Cat Engine

Jet CAT P200 Technical Specifications published is shown in Table 1 [3].

TABLE 1
JET CAT P200 TECHNICAL SPECIFICATIONS

Technical property	Value
Max RPM	112 000 rpm
Max thrust	230 N
Max EGT	750^{o}C
Pressure ratio	3.7
Mass flow	0.45 kg/s
Max fuel consumption	730 ml/min
Specific fuel consumption	0.152 kg/h. N

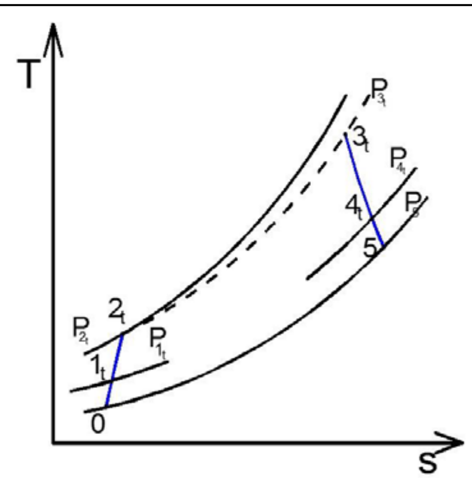

Fig. 2T-s diagram ofTurbojet Engine

This engine model consists of five main components: Inlet, radial compressor, combustion chamber, axial turbine, and exhaust convergent nozzle. In order to study their performance, both energy and mass balances are applied on each component. A complete T-s diagram of the cycle is presented in Figure.2, with numerical value given in Table 2.

A Brayton–Joule cycle own developed code is used to predict the Cycle of the turbojet engine.Such thermodynamic model, the following assumptions areconsidered:

1. Ambient pressure and temperature of air are 288 K and 101.325 kPa, respectively.
2. Air behaves as a semi-ideal gas with specific heats vary with temperature.
3. Fuel/air mixture behaves like a semi-ideal equivalent gas with enthalpy, entropy and specific heats depending on temperature and fuel/air equivalence ratio.
4. Intake total pressure recovery factor is 0.96.
5. Compressor's isentropic-efficiency is 0.70.
6. Combustion chamber efficiency is 0.95.
7. Combustion chamber total pressure recovery factor is 0.95.
8. Turbine's isentropic-efficiency is 0.89.

TABLE 2
TURBOJET ENGINE THERMAL CYCLE ANALYSIS RESULTS

9. Nozzle's isentropic-efficiency is 0.92.

The used fuel is liquid kerosene with a heating value of 43.257 MJ/kg. Using such hypotheses, a parametric analysis is carried out to derive the cycle pressure ratio that guaranteed the maximum engine specific thrust of 512.12 N/ (kg/s). Therefore, a pressure ratio of 3.7is selected and a maximum cycle-temperature of 1023 K is adopted accordingly. Correspondingly, for the design thrust of 230.45 N at the selected point, the air mass flow rate is 0.45 kg/s. The other relevant parameters of the cycle are reported in Table

Parameters		Value	
Inlet Total Temperature	T_{1t}	288	K
Inlet Total Pressure	p_{1t}	97272	Pa
Total Pressure at Compressor Exit	p_{2t}	359906.4	Pa
Total Temperature at Compressor Exit	T_{2t}	474.48	K
Total Temperature difference across Compressor	ΔT_{tc}	186.48	K
Compressor shaft work	L_{ec}	187416.58	J/kg
Outlet Total Pressure from Combustion Chamber	p_{3t}	338312.02	Pa

Total Temperature difference across Turbine	ΔT_{tt}	161.55	K
Outlet Total Temperature from Combustion Chamber	T_{3t}	1184.55	K
Adiabatic Total Temperature at Turbine Exit	T_{4adt}	1033.03	K
Total Pressure at Turbine Exit	p_{4t}	169511.48	Pa
Turbine shaft work	L_{et}	193212.96	J/kg
Pressure at Exhaust Nozzle exit	p_5	101325	Pa
Gas Flow Velocity at Nozzle Exit	C_5	512.12	m/s

TABLE 3
TURBPROP ENGINE THERMAL CYCLE ANALYSIS RESULTS

Specific Thrust	F_s	512.12	N
Thrust of Engine	F	230.45	N

2.2 Turboprop Engine Thermodynamic Cycle Analysis.

The conversion to Turboprop Engine will done by addingLow-pressure single-stage axial flow free turbine. Such Free power turbine is used to extract power required to drive propeller shaft through the gearbox. Non-ideal parametric cycle for Turboprop engine is calculated based on previouscalculation of Turbojet engine by using developed code and the results are presented in Table 3.

Parameters		Value	
Total adiabatic temperature at free power turbine exit	T_{5ad}	921.6	K
Total temperature at free power turbine exit	T_{5t}	932.76	K
Total pressure at free power turbine exit	p_{5t}	109866	Pa
Nozzle exit temperature	T_6	916.03	K
Specific Shaft Power	P_{Prs}	107932.44	W/ kg/s
Specific Equivalent power	P_{eqs}	121265.77	W/ kg/s
Engine Shaft Power	P_{Pr}	48569.6	W
Engine Equivalent power	P_{eq}	54569.6	W
Effective Specific fuel Consumption	C_e	0.7441	kg_f/kW h
Equivalent Specific Fuel Consumption	C_{eq}	0.6623	kg_f/kW h
Engine Thrust	F_{TPE}	818.54	N

2.3 Benefitsfrom Conversion

The concept of turbojet-to-turboprop conversion requires a substantial amount of redesign or considerable changes to the core engine (such as additional low pressure (LP) spools, LP free turbine, etc.). The advantage of such conversion isincreasing of Engine Thrust, which appears in the thermodynamic cycle analysis as it increase from 230.45 N to 818.54 N. Also engine Specific fuel consumption decreases by 72% which increase UAV endurance

TABLE 4
DESIGN REQUIREMENTS AND GEOMETRICAL CONSTRAINTS

3 TURBOPROP ENGINE DESIGN MODIFICATION

3.1 Free Turbine Aerothermodynamics Design

The mean line design & analysis serves as the foundation for gas path preliminary design and Performance estimation in a turbo machinery design cycle [4] [5]. The design of Low-pressure turbine stage is carried out using developed code based on the mean line design approach. Baseline design requirements and geometrical constraints specifications of a typical turbine stage are listed in Table 4.

Parameters		Value	
Diameter at inlet tip section	D_{1e}	0.1	m
Axial Component of Absolute Velocity at inlet section	C_{1am}	195	m/s
Angle of Absolute Flow Velocity at rotor entry	α_{1m}	30	degree
Absolute Velocity Flow Coefficient	φ	0.95	-
Shaft Speed	n	75000	rpm

The preliminary design involves the stage flow path design using

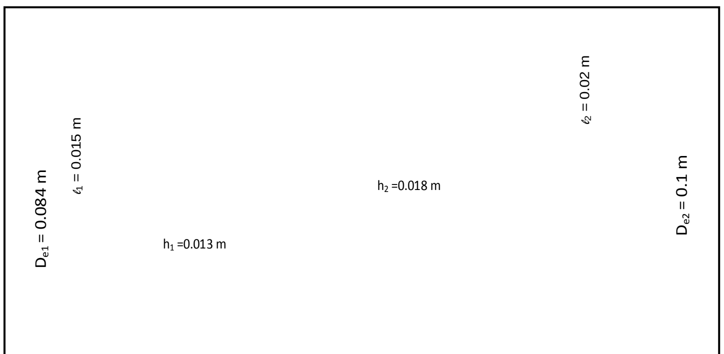

Fig. 3 Compressor Turbine & Free power Turbine meridonal plane

inlet operating conditions of the turbine to meet the given geometric limitations [5]. The working fluid, design speed, mass flow rate, inlet total pressure, temperature, and the reference input data are specified. One of the critical requirements of the preliminary design mode is to explore the design targets. This involves parametric study of the design specifications by choosing the flow coefficient, the stage loading coefficient and absolute flow velocity angle at rotor entry. The flow path constraints are constant tip radius, type of blade sections (constant/variable), flow path limits (maximum hub/tip radius), inlet and exit swirl angles. In Table 5 preliminary design parameters of Turbine is presented and Turbine meridonal plane is shown in Figure 3

TABLE 5
FREE TURBINE PRELIMINARY DESIGN PARAMETERS

Parameters		Value	
Free Turbine shaft Work	L_{et}	107024	J/kg
Air mass flow rate	m	0.45	kg/s
Blade length at inlet Section	l_1	0.02	m
Stage Reaction at Hub section	ρ_i	0.1023	
Stage Reaction at mean section	ρ_m	0.438	
Blade length at Turbine Rotor exit section	l_2	0.02	m
Blade Width	h	0.018	m
Chord of blade	b	0.027	m
Turbine Rotor Pitch Chord Ratio	$t_m/_b$	0.85	
Rotor Number of blades	z	11	
Turbine Stator Pitch Chord Ratio	$t_m/_b$	0.75	
Stator Number of blades	z	13	

3.2 Blade Profiles Design

As pointed out earlier, velocity triangles vary from root to tip of the blade because the blade speed U is not constant and varies from root to tip. For turbine stage blades design, the free vortex law of blading is selected which characterized by:
Constant stagnation enthalpy across the annulus $dh_o/_{dr} = 0$, constant axial velocity $dC_a/_{dr} = 0$ and the whirl component of velocity C_w is inversely proportional to the radius as shown in Equation (1).

$$C_w \times r = Constant \quad (1)$$

Now using subscript m to denote condition at mean diameter, the free vortex variation of nozzle angle α1 at root and tip is shown in Equation (2) & (3) respectively.

$$\alpha_{1i} = \tan^{-1}\left(\frac{r_{1i}}{r_{1m}} \tan\alpha_{1m}\right) \quad (2)$$

$$\alpha_{1e} = \tan^{-1}\left(\frac{r_{1e}}{r_{1m}} \tan\alpha_{1m}\right) \quad (3)$$

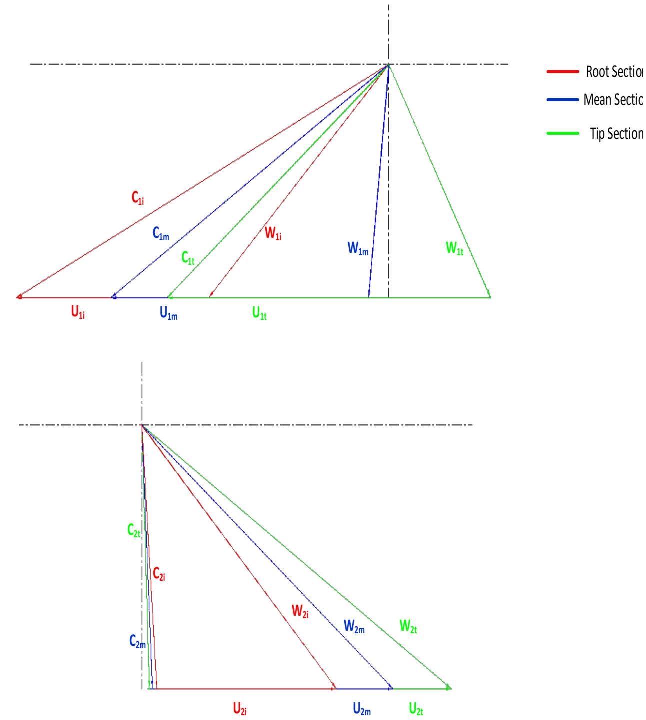

Fig. 4Velocity triangles for hub, mean, and tip

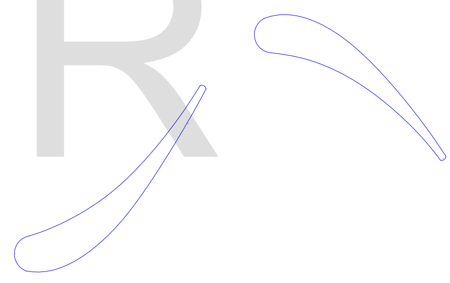

Fig. 5 Stator & Rotor Mean Profile

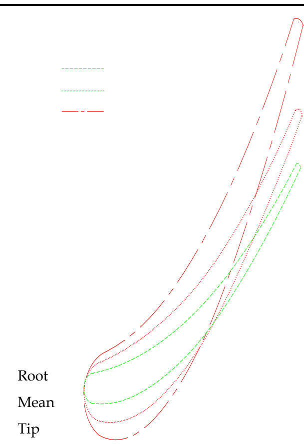

Fig. 6 Stator Blade Profiles

Figure 4 shows the velocity triangles for hub, mean, and tip sections across the rotor row.
Arbitrary blade profiles shown in Figure 5&6&7&8 are devel-

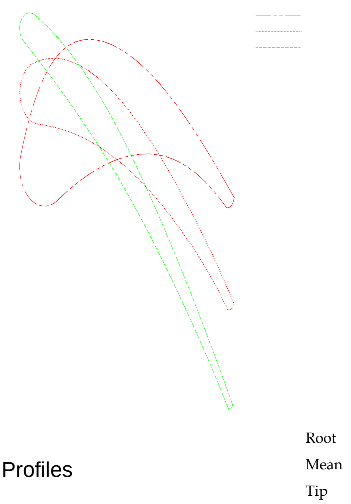

Fig. 7 Rotor Blade Profiles

oped by the graphical control of Bezier curves for both the suction and pressure surface and the camber line thickness distributions. The fifth order polynomial is used to control the profile shape in this method. In Figure 8, 3-D stator and rotor blades are presented

3.3 Detailed design of free turbine

The Detailed design is carried out using the procedure described in section3.2. The Stator row features 13 blades having constant stagger angles with radius, while a free-vortex criterion is used to determine the angles at various radii of the 11 rotor blades. In Figure 9detailed free turbine assembly is presented and parts name and quantity for the assembly are presented in Table 6.

The Free turbine shaft is made of V145 steel, shaft supported by a preloaded ball bearings. The rotor-bearing module is accurately aligned and balanced with all other components in order to control the tip clearances of turbine. Bearings and shaft tunnel are lubricated and cooled with oil fed from the externally mounted tank by a tube through the compressor casing.

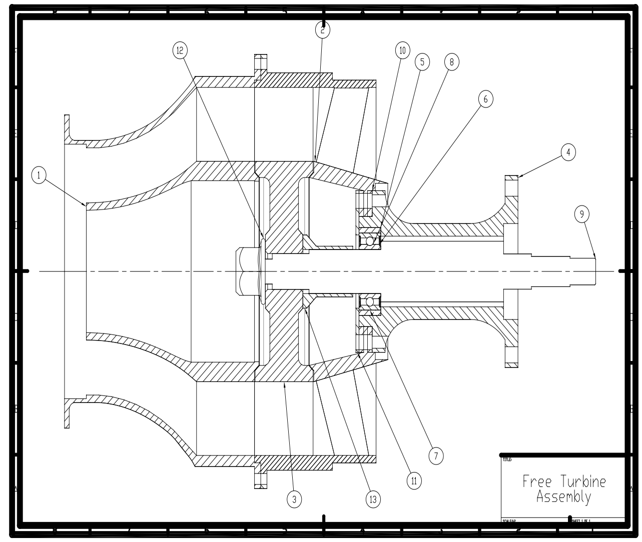

Fig. 9 Free Power Turbine Assembly

TABLE 6
PARTS NAME AND QUANTITY FOR FREE POWER TURBINE ASSEMBLY

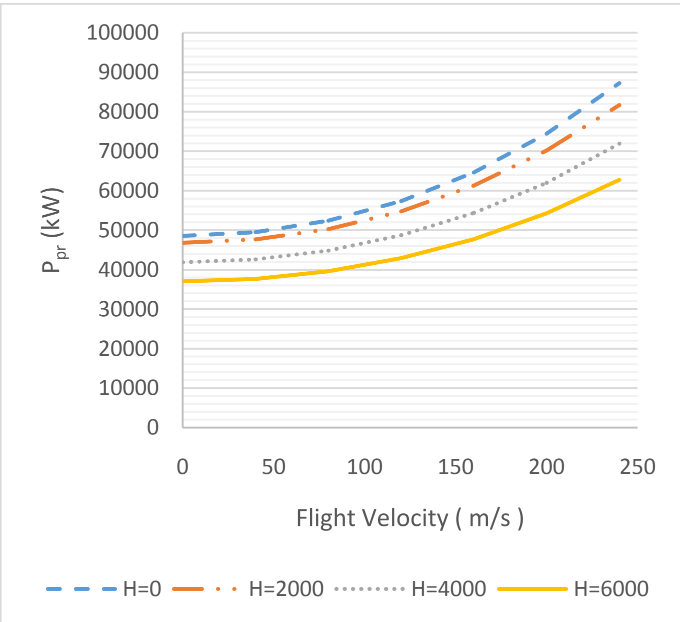

Fig. 10 Change of Shaft Power with Altitude and Flight Velocity

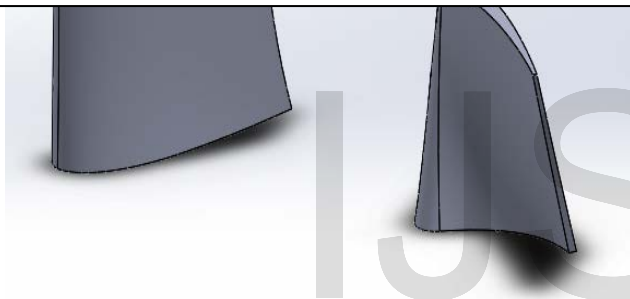

Fig. 8 3-D Statorand Rotor Blade

4 FlightPerformance of Developed Turboprop Engine

The determination of the Turboprop Engine flight performance is an important task during the design phase, where one of the available method is the theoretical one. The theoretical prediction is applied for engine flight performance prediction. Developed method gives the possibility of calculation of flight performance at possible used laws of regulation[5]. The obtained results can be used for aircraft flight performance prediction and navigation calculation, specially the fuel consumption determination.

In Figure 10 & 11 the Change of shaft power and effective specific fuel consumption with altitude and flight velocityis presented

5 Conclusion

Conversion of Micro Turbojet Engine to Micro Turboprop engine through design and analysis of a low-pressure freeaxial gas turbine stage is described using mean line design and Free Vortex technique using developed code. Meridonal plane design of turbine stage is carried out using mean line Aproxtimion to obtain the stage design parameters taking in to account the baseline design requirements and geometrical constraints.

Item No.	Part Name	Quantity
1	Stator	1
2	Rotor Case	1
3	Free Turbine Rotor	1
4	Shaft Tunnel	1
5	Tolerance Ring	1
6	Bearing	1
7	Turbine Shaft	1
8	Gasket	2
9	Clamp Ring	1
10	prevailing torque hex flange nut	1
11	Collar	1

The blade profiles are thenreached using the Bezier profile curves to optimize the blade shape by profile modification.

The theoretical prediction of Turboprop Engine flight performance is presented. The obtained results can be used for aircraft flight performance prediction and navigation calculation, specially the fuel consumption determination.

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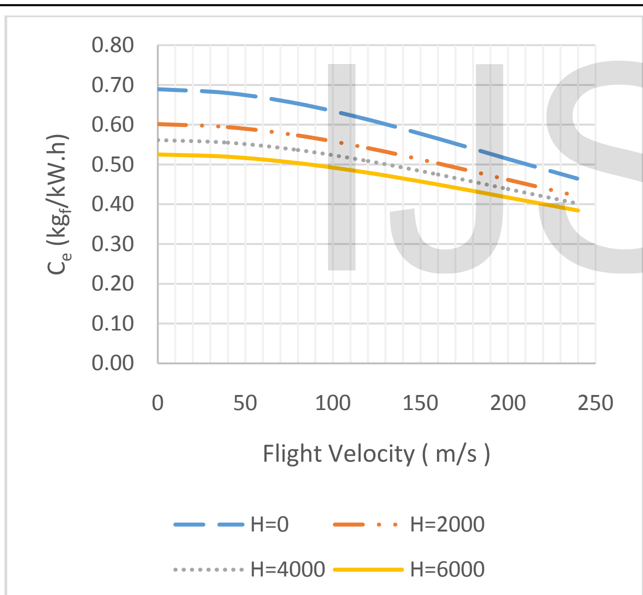

Fig. 10 Change of Effective Specific fuel Consumption with Altitude and Flight Velocity