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Investigation on Pressure Distribution of Flow Past Over A Symmetrical Aerofoil at Low Subsonic Speed

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ABSTRACT: The main objective to aerodynamics properties of flow over symmetrical aerofoil (NACA-0012) at low subsonic speed ($M < 0.3$). Generally symmetrical aerofoil used in supersonic aircraft as military purpose like fighter and bomber to give smooth speed at supersonic speed ($M > 1$) and including compressible effect.

Here try to determine pressure distribution over symmetrical aerofoil at low sub-sonic speed and inviscid incompressible flow. Also we wanted to know about flow behavior on symmetrical aerofoil of upper and lower side. So we get behavior of flow over a wing whose wing span and chord length is given, like we can predict about pressure distribution over any type of aerodynamics shape used in any subsonic and supersonic speed of vehicle (aircraft). This paper fully carried out in low subsonic open circuit wind tunnel having speed in test section is $M < 0.08$ as experimentally and theoretically analyzed for its performance.

KEYWORDS: Pressure distribution over aerofoil, flow past body, fluid dynamics, aerodynamics, applied aerodynamics.

I. INTRODUCTION

This paper brings you to performance analysis of presence distribution over flow past symmetrical aerofoil (NACA-0012) at ($M < 0.3$) low subsonic speed in open circuit incompressible low speed subsonic wind tunnel.

We are considering $M = 0.08$ and $M = 0.03$ over symmetrical aerofoil (NACA-0012), its chord length 160mm and using span = 260mm. We know that at zero angle of attack, zero lift produced or at low angle of attack, low lift would be produced. So we wanted to calculate " C_p " value at different flow speed [low, medium, high rpm of fan of wind tunnel] with constant angle of attack [like 0° , 5° , 15°].

C_p will say about upper and lower surface flow distribution over aerofoil or wing, so they predict about circulation over an aerofoil due to that can predict about lift a drag over whole aircraft. So there an aim is concentrate only pressure distribution over the symmetrical aerofoil at low subsonic speed.

II. SPECIFICATIONS OF WIND TUNNEL

TYPE: -Open Circuit Incompressible Subsonic

MACH: 0.08

TEST SIZE: - 1000*300*300mm with FRP body

MOTOR: - single phase 5 HPDC MOTOR

RPM: -2800, Thyristor Drive

BLOWER: - Cast Aluminium Axial Flow Fan

PITOT TUBE: -Prandi Pitot Tube

SMOKE GENERATOR: -Kerosine Smoke Generator

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III. GEOMETRY & SPCIFICATION OF AEROFOIL

As we know NACA-0012, its chord length is 160 mm and consisting 12 ports (points or holes on aerofoil) in it, shown in the figure as X-X axis plane. The port no. 7th is stagnation point or leading edge point and port 1st is trailing edge point. Port no. 8,9,10,11,12 and port no. 6,5,4,3,2 are upper port(suction port) and lower port (pressure port) respectively. On stagnation point is maximum value of C_p either theoretical or actual and least value of trailing edge port. The C_p (actual) value of upper ports (suction ports) are -ve and lower ports (pressure ports) are +ve but less than 1 because subsonic incompressible. The given table no:1 Specification is mentioned. The aerofoil x-axis and y-axis value are given in this table and numbering of all the ports given their respective position. Fig:1 also mentioned about specification of NACA-0012 aerofoil which is placed on test section in wind tunnel.

Table 1
Specification of aerofoil

PORTS/ POINTS	Y, UPPER (mm)	Y, BOTTOM (mm)	X (mm)	X/C
1	0	0	160	1
2	9	-9	120	0.75
3	12	-12	90	0.5
4	15	-15	55	0.3125
5	14	-14	30	0.1875
6	5	-5	10	0.0625
7	0	0	0	0
8	5	-5	10	0.0625
9	14	-14	30	0.1875
10	15	-15	55	0.3125
11	12	-12	90	0.5
12	9	-9	120	0.75

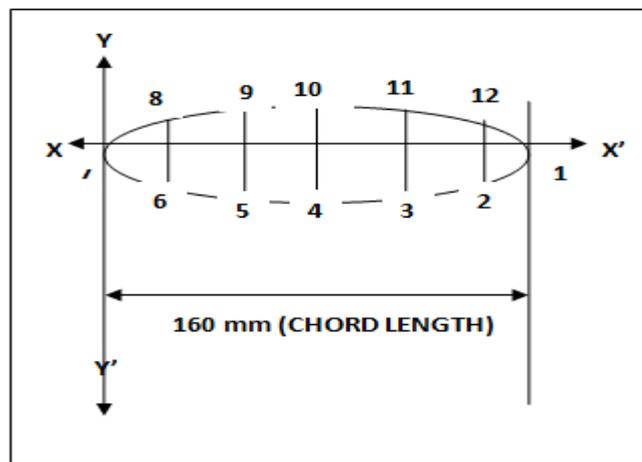


Fig. 1. NACA 0012 AEROFOIL

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IV. FORMULATION OF CO-EFFICIENT OF PRESSURE ($C_{P(Actual)}$)

$$C_{p(actual)} = \frac{P_i - P_\infty}{q_\infty}$$

$$q_\infty = \frac{1}{2} \rho v^2$$

$$v_\infty = \sqrt{2gh_{air}}$$

$$\text{equivalent to air value} = P_i - P_\infty = \frac{101325}{0.76} \times h_{Hg}$$

$$\text{Span area(s)} = \text{chord length} \times \text{span length} = 160 \times 290 \text{ mm}^2 = 0.0464\text{m}^2$$

$$C_{p(th)} = 1 - (x/c)^2$$

Where,

$$\rho_{Hg} = 13600 \text{ kg/m}^3$$

$$\rho_{Hyd} = 1.23 \text{ kg/m}^3$$

V. FLOW ANALYSIS

Flow analysis of 0° , 5° and 15° AOA with vary of speeds of wind tunnel respectively. In a given AOA with different speed like low speed (600-1199 rpm), medium speed (1200-1999 rpm) and high speed (2000-2800 rpm).

At 0° AOA, 5° AOA and 15° AOA are following analysis done with respective speeds.

(a) AT 0° AOA

As mentioned in early that flow analysis over a symmetrical aerofoil at 0° AOA in different speed like low speed ,medium speed and high speed observe through practical value and fig. It is in Table:2,3,4 and fig: 2,3,4 are respectively. The $C_{p(actual)}$ values and $C_{p(actual)}$ vs x/c fig. are given in above tables and fig. .

Table 2:
Pressure distribution over a symmetrical aerofoil for low speed at 0° AOA.

S.No.	$P_i - P_\infty$	q_∞	C_{pact}	C_{pth}
1.	9.799	107.8	0.09	0
2.	0	107.8	0	0.43
3.	9.799	107.8	0.09	0.75
4.	9.799	107.8	0.09	0.902
5.	9.799	107.8	0.09	0.965
6.	9.799	107.8	0.09	0.996
7.	78.39	107.8	0.727	1
8.	78.39	107.8	0.727	0.996
9.	78.39	107.8	0.727	0.965
10.	39.19	107.8	0.363	0.902
11.	9.59	107.8	0.181	0.75
12.	9.799	107.8	0.09	0.43

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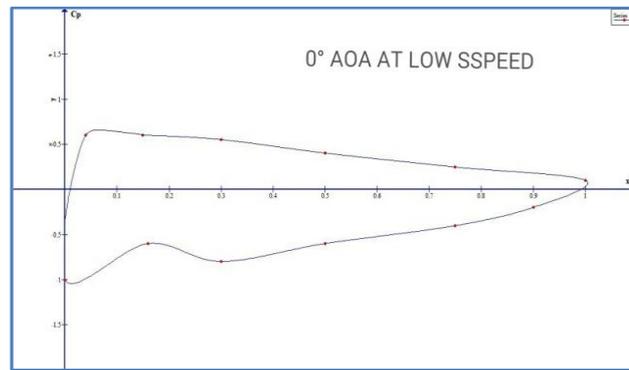


Fig. 2.

Table 3:
Pressure distribution over a symmetrical aerofoil for medium speed at 0° AOA.

S.No.	$P_i - P_\infty$	q_∞	Cp_{act}	Cp_{th}
1	19.598	254.96	0.076	0
2.	9.799	254.96	0.038	0.43
3.	19.598	254.96	0.076	0.75
4.	19.598	254.96	0.076	0.902
5.	19.598	254.96	0.076	0.965
6.	0	254.96	0	0.996
7.	117.89	254.96	0.461	1
8.	87.99	254.96	0.325	0.996
9.	87.99	254.96	0.325	0.965
10.	78.39	254.96	0.307	0.902
11.	9.799	254.96	0.038	0.75
12.	9.799	254.96	0.038	0.43

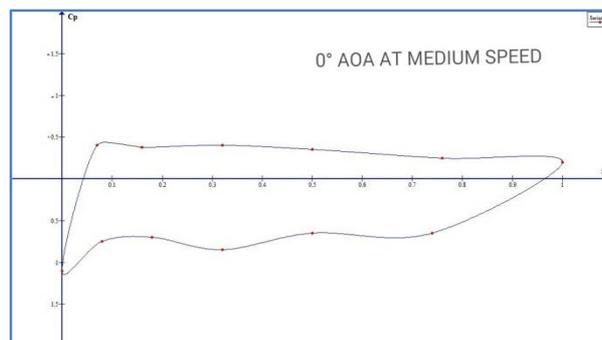


Fig. 3.

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Table 4:
Pressure distribution over a symmetrical aerofoil for high speed at 0° AOA.

S.No.	$P_i - P_\infty$	q_∞	$C_{p_{act}}$	$C_{p_{th}}$
1	48.92	68.65	0.108	0
2.	29.33	68.65	0.065	0.43
3.	48.92	68.65	0.108	0.75
4.	48.92	68.65	0.108	0.902
5.	68.52	68.65	0.151	0.965
6.	154.98	68.65	0.345	0.996
7.	254.64	68.65	0.564	1
8.	9.799	68.65	0.021	0.996
9.	215.98	68.65	0.478	0.965
10.	97.99	68.65	0.217	0.902
11.	39.19	68.65	0.086	0.75
12.	0	68.65	0	0.43

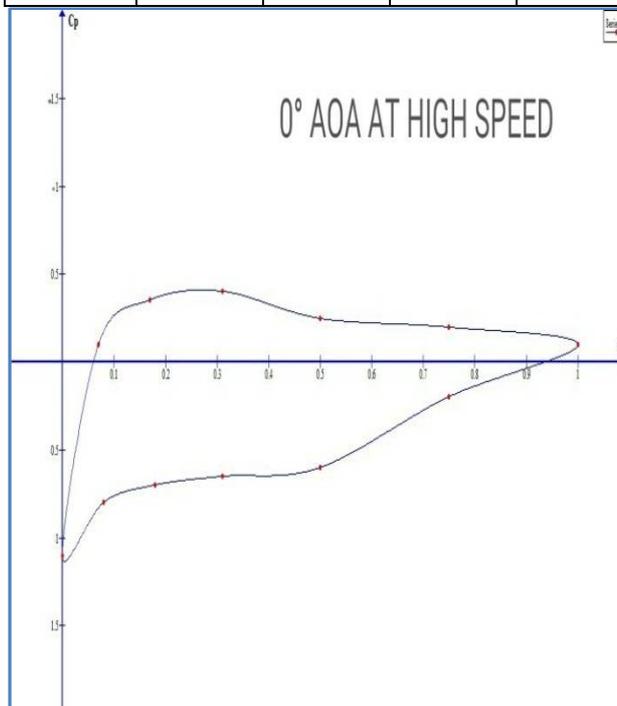


Fig. 4.

(b) AT 5° AOA

As mentioned in early that flow analysis over a symmetrical aerofoil at 5° AOA in different speed like low speed ,medium speed and high speed observe through practical value and fig. It is in Table:5, 6, 7 and fig: 5, 6, 7 are

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respectively. The $C_{p (actual)}$ values and $C_{p (actual)}$ vs x/c fig. are given in above tables and fig. .If AOA increases graph will converse.

Table 5:
Pressure distribution over a symmetrical aerofoil for low speed at 5° AOA.

S.No.	$P_i - P_\infty$	q_∞	$C_{p_{act}}$	$C_{p_{th}}$
1	0	68.65	0	0
2.	0	68.65	0	0.43
3.	9.799	68.65	0.142	0.75
4.	19.598	68.65	0.285	0.902
5.	9.799	68.65	0.142	0.965
6.	29.33	68.65	0.427	0.996
7.	78.39	68.65	0.142	1
8.	48.92	68.65	0.712	0.996
9.	29.33	68.65	0.427	0.965
10.	29.33	68.65	0.427	0.902
11.	0	68.65	0	0.75
12.	0	68.65	0	0.43

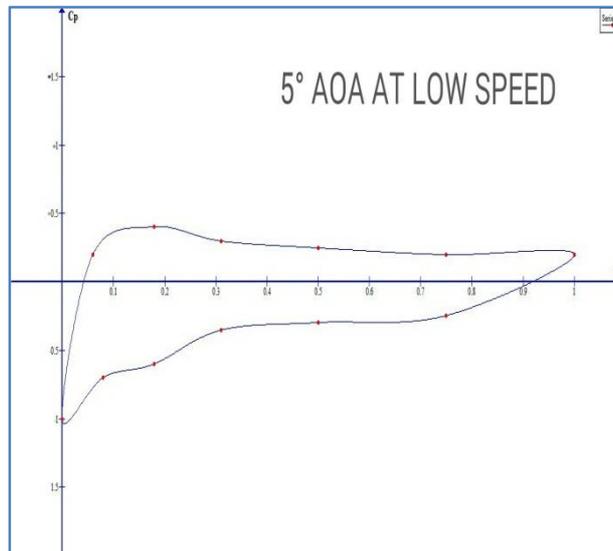


Fig. 5.

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Table 6:
Pressure distribution over a symmetrical aerofoil for medium speed at 5° AOA.

S.No.	$P_i - P_\infty$	q_∞	C_{Dact}	C_{Dth}
1.	9.799	166.75	0.058	0
2.	9.799	166.75	0.058	0.43
3.	9.799	166.75	0.058	0.75
4.	9.799	166.75	0.058	0.902
5.	0	166.75	0	0.965
6.	68.52	166.75	0.41	0.996
7.	97.99	166.75	0.587	1
8.	58.66	166.75	0.351	0.996
9.	97.99	166.75	0.587	0.965
10.	117.59	166.75	0.705	0.902
11.	19.59	166.75	0.117	0.75
12.	19.59	166.75	0.117	0.43

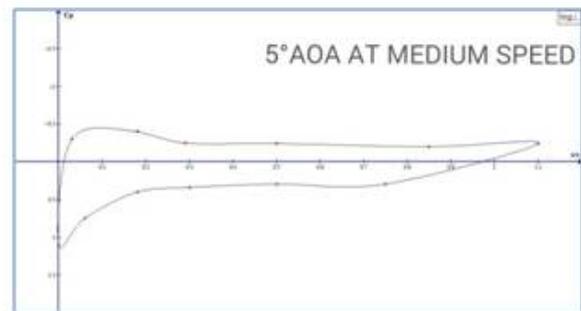


Fig. 6.

Table 7:
Pressure distribution over a symmetrical aerofoil for high speed at 5° AOA.

S.No.	$P_i - P_\infty$	q_∞	C_{Dact}	C_{Dth}
1.	0	98.09	0	0
2.	19.598	98.09	0.199	0.43
3.	9.799	98.09	0.199	0.75
4.	19.598	98.09	0.199	0.902
5.	19.598	98.09	0.199	0.965
6.	39.196	98.09	0.399	0.996
7.	107.72	98.09	1.009	1
8.	9.799	98.09	0.09	0.996
9.	68.52	98.09	0.698	0.965
10.	58.66	98.09	0.598	0.902
11.	9.799	98.09	0.099	0.75
12.	0	98.09	0	0.43

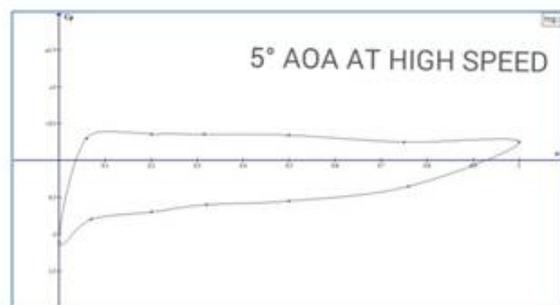


Fig. 7.

(c) AT 15° AOA

As mentioned in early that flow analysis over a symmetrical aerofoil at 15° AOA in different speed like low speed ,medium speed and high speed observe through practical value and fig. It is in Table:8, 9, 10 and fig: 8, 9, 10 are respectively. The $C_{p(Actual)}$ values and $C_{p(Actual)}$ vs x/c fig. are given in above tables and fig.

Now the stall region will start but it depends on shape and size of aerofoil and angle of attack between chord line and free stream.

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Table 8:
Pressure distribution over a symmetrical aerofoil for low speed at 15° AOA.

S.No	$P_i - P_\infty$	q_∞	$C_{p_{act}}$	$C_{p_{th}}$
1.	29.33	205.96	0.142	0
2.	19.598	205.96	0.095	0.43
3.	39.196	205.96	0.19	0.75
4.	29.33	205.96	0.142	0.902
5.	39.196	205.96	0.19	0.965
6.	87.99	205.96	0.427	0.996
7.	205.31	205.96	0.996	1
8.	9.799	205.96	0.047	0.996
9.	137.19	205.96	0.66	0.965
10.	68.52	205.96	0.33	0.902
11.	19.598	205.96	0.095	0.75
12.	19.598	205.96	0.095	0.43

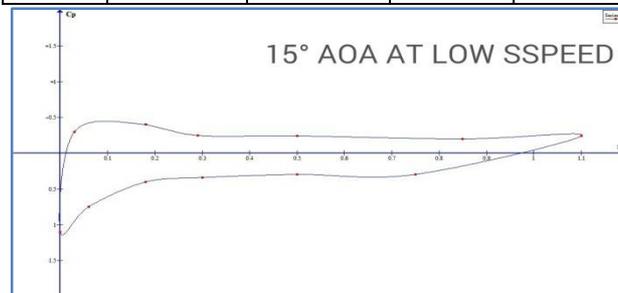


Fig. 8.

Table 9:
Pressure distribution over a symmetrical aerofoil for medium speed at 15° AOA.

S.No.	$P_i - P_\infty$	q_∞	$C_{p_{act}}$	$C_{p_{th}}$
1	29.33	205.96	0.142	0
2.	19.598	205.96	0.095	0.43
3.	39.196	205.96	0.19	0.75
4.	29.33	205.96	0.142	0.902
5.	39.196	205.96	0.19	0.965
6.	87.99	205.96	0.427	0.996
7.	205.31	205.96	0.996	1
8.	9.799	205.96	0.047	0.996
9.	137.19	205.96	0.66	0.965
10.	68.52	205.96	0.33	0.902
11.	19.598	205.96	0.095	0.75
12.	19.598	205.96	0.095	0.43

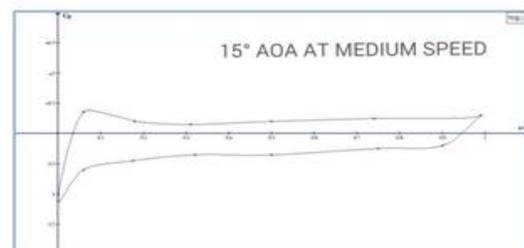


Fig.9.

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Table 10:
Pressure distribution over a symmetrical aerofoil for high speed at 15° AOA.

S.No.	$P_1 - P_{\infty}$	q_{∞}	$C_{p,act}$	$C_{p,th}$
1.	48.92	382.5	0.127	0
2.	29.33	382.5	0.076	0.43
3.	48.92	382.5	0.128	0.75
4.	48.92	382.5	0.128	0.902
5.	68.52	382.5	0.179	0.965
6.	156.78	382.5	0.409	0.996
7.	254.74	382.5	0.665	1
8.	9.799	382.5	0.025	0.996
9.	231.35	382.5	0.556	0.965
10.	97.99	382.5	0.256	0.902
11.	39.196	382.5	0.102	0.75
12.	0	382.5	0	0.43

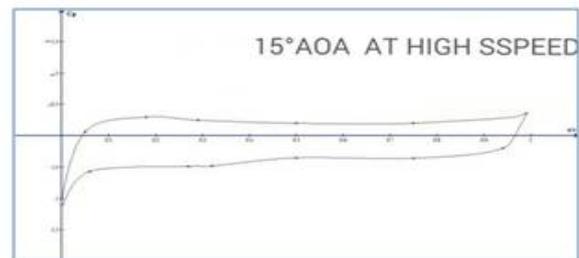


Fig. 10.

VI. FURTHER WORK

This is purely experimental work and its analysis but following to be done in future:-

- CFD analysis.
- Flow visualization at low speed (incompressible low) subsonic.
- Compare the experimentally with CFD flow analysis with different contour.

VII. RESULT AND DISCUSSION

After different AOA with respective speeds, we got corresponding $C_{p(actual)}$ values with multi-tube manometer as experimentally with their table and graphs.

The graph and table of experimentally verified value will says about behavior of flow over a aerofoil that given lift on different AOA with respective speeds.

The best ways to calculate lift over flow past body is calculate first coefficient of pressure, so coefficient of pressure is calculated experimentally of given aerofoil(NACA-0012) with set up of instrument .

VIII. CONCLUSION

The experimentally carried out in low subsonic open circuit wind tunnel given maximum speed as mach no. less than and equal to 0.08. Due to pressure distribution over NACA-0012 aerofoil, taking AOA constant with varying the speed like low, medium and high speed, so taken three sequency AOA repeating with another AOA got $C_{p(actual)}$ values and drawn graphs with corresponding values.

The analysis of $C_{p(actual)}$ values of corresponding AOA and its graph easy to predict next AOA but before stall limit of AOA. We can design an aerofoil either symmetrical or cambered ,this experiment will going to help and to develop design of aerodynamic of aircraft (any shape of other aerodynamic vehicles).

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