Experimental View on Performance of Flow Past Over A Symmetrical Aerofoil at Low Subsonic Speed

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ABSTRACT

This paper brings us the aero dynamics properties of flow over symmetrical aerofoil (NACA-0012) at low subsonic speed (M<0.3). Generally symmetrical aerofoil are used in supersonic aircraft for military purpose like fighter and bomber to give smooth speed at supersonic speed (M>1) and including compressible effect. Here attempt has been made to determine pressure distribution over symmetrical aerofoil at low sub-sonic speed and in viscid incompressible flow. The flow behavior on symmetrical aerofoil of upper and lower side has been investigated. 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). Experiment were out in low subsonic open circuit wind tunnel having speed in test section is M<0.08 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

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

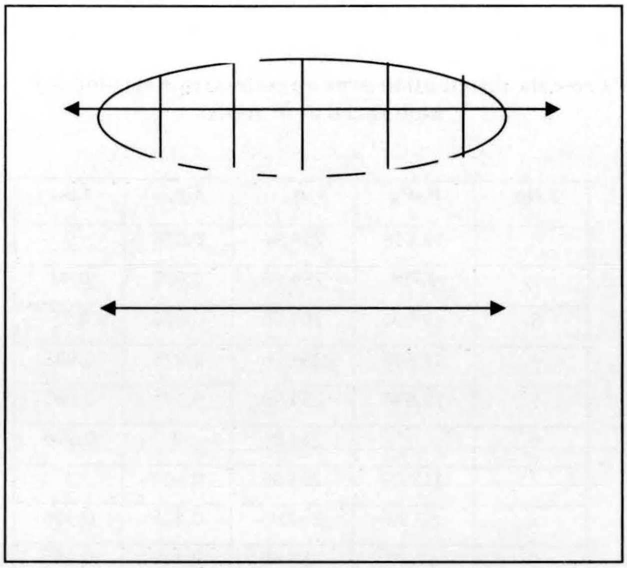

Fig. 1. NACA 0012 AEROFOIL

Table 2:

Pressure distribution over a symmetrical aerofoil for

S.No.	P_i-P_∞	q_∞	Cp_{act}	Cp_{th}
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

low speed at 0° AOA.

IV FORMULATION OF CO-EFFICIENT OF PRESSURE ($C_{P\,(ACTUAL)}$)

$$C_{p(actual)} = \frac{Pi-P\infty}{q\infty}$$

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

$$v_\infty = \sqrt{2gh\text{air}}$$

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

Span area(s) = chord length × span length = 160 × 290 mm^2= 0.0464m^2

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

Where,

δ_{Hg} = 13600 kg/m^3

δ_{Hyd} = 1.23 kg/m^3

V FLOW ANALYSIS

Flow analysis of 0^0, 5^0 and 15^0 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 0^0 AOA, 5^0AOA and 15^0 AOA are following analysis done with respective speeds.

(a) AT 0^0 AOA- As mentioned in early that flow analysis over a symmetrical aerofoil at 0^0 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. .

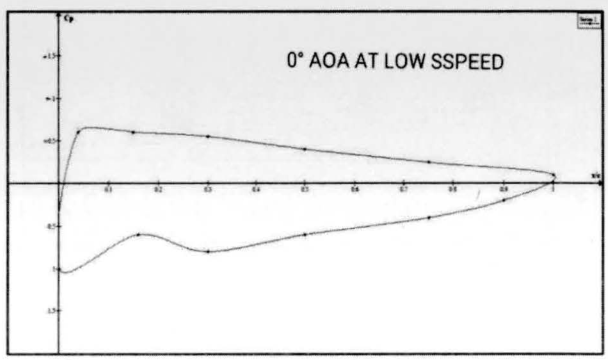

Fig. 2.

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

S.No.	P_i-P_∞	q_∞	Cp_{act}	Cp_{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

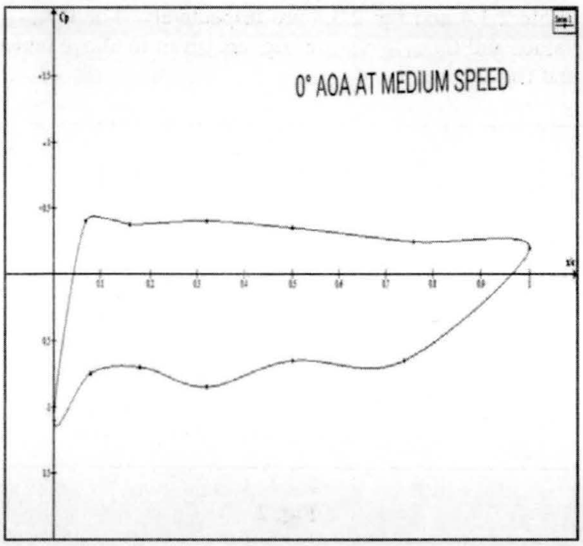

Fig. 3.

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

S.No.	P_i-P_∞	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

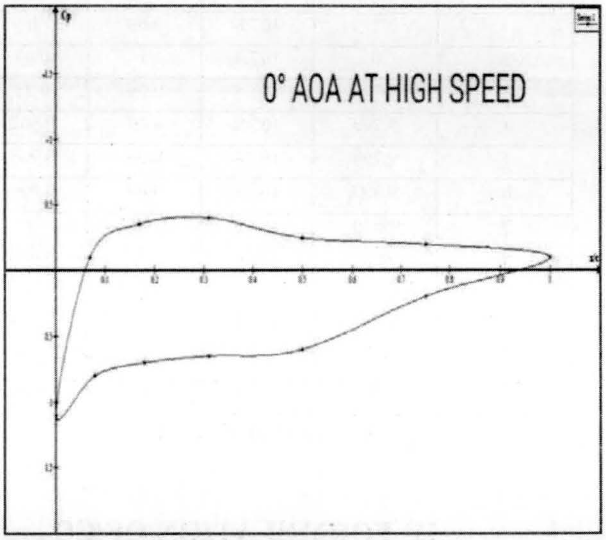

Fig. 4.

(b) AT 5^0 AOA- As mentioned in early that flow analysis over a symmetrical aerofoil at 5^0 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 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_∞	q_∞	Cp_{act}	Cp_{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

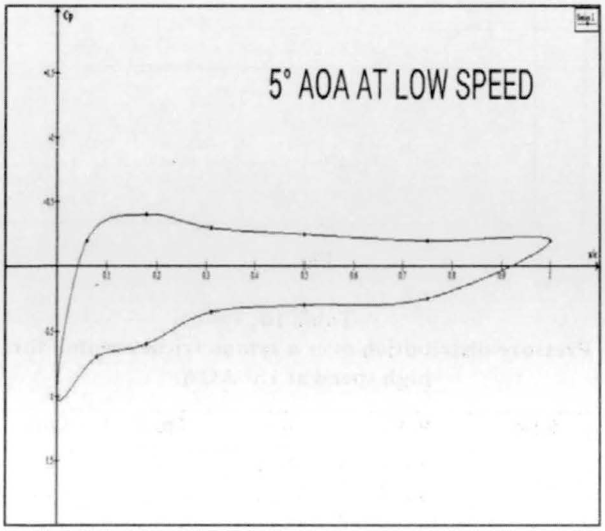

Fig. 5

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

S.No.	P_i-P_∞	q_∞	Cp_{act}	Cp_{th}
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

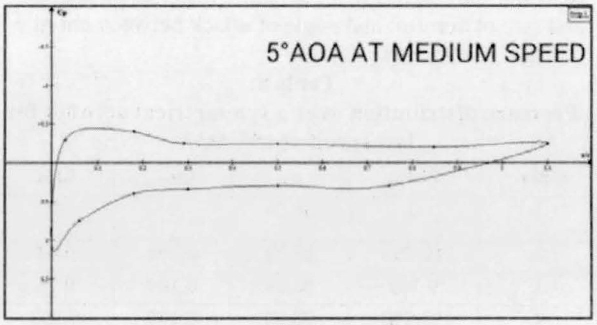

Fig. 6

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

S.No.	Pi-P_∞	q_∞	Cp_{act}	Cp_{th}
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

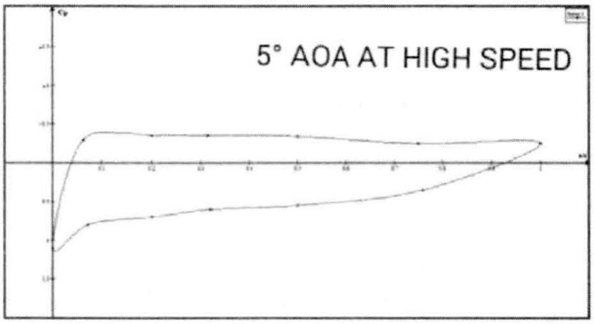

Fig.7

(c) AT 15^0 AOA- As mentioned in early that flow analysis over a symmetrical aerofoil at 15^0 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.

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

S.No.	Pi-P∞	q∞	Cpact	Cpth
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

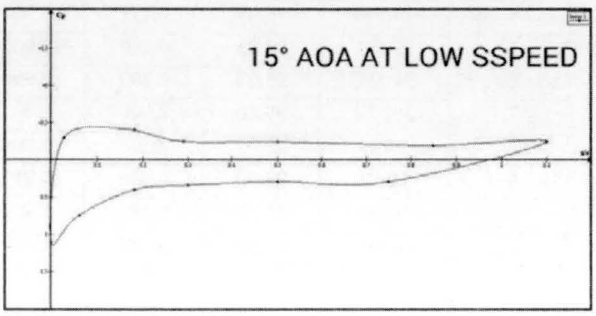

Fig. 8.

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

S.No.	Pi-P∞	q∞	Cpact	Cpth
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

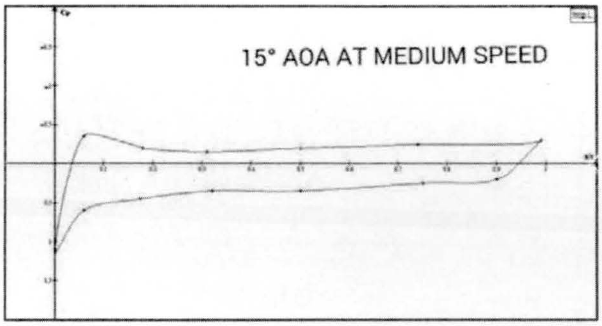

Fig. 9

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

S.No.	Pi-P∞	q∞	Cpact	Cpth
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

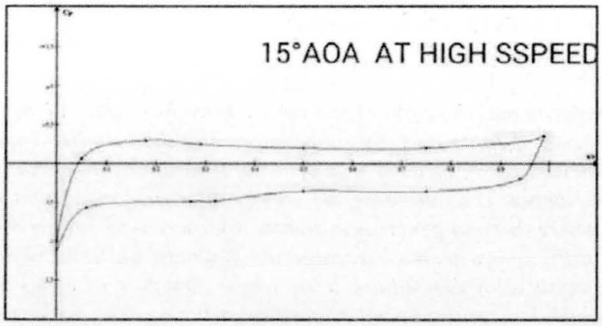

Fig. 10.

VI RESULT AND DISCUSSION

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

The graph and table of experimentally verified value depict the 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 to calculated first coefficient of pressure, so coefficient of pressure is calculated experimentally with given aerofoil(NACA-0012) with set up of instrument .

VII CONCLUSION

The experimentally carried out low subsonic open tunnel gives 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 sequencly 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).

VIII FURTHER WORK

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

(a) CFD analysis.
(b) Flow visualization at low speed (incompressible low) subsonic.
(c) Compare experimentally with CFD flow analysis with different contour.

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