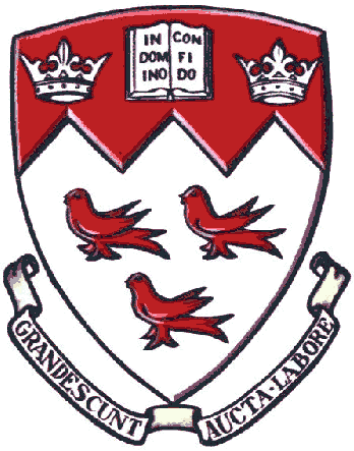
MECH 539: Computational Aerodynamics

Project #3: Murman-Cole Scheme for the Transonic Small Disturbance Equation (Corrected Version)



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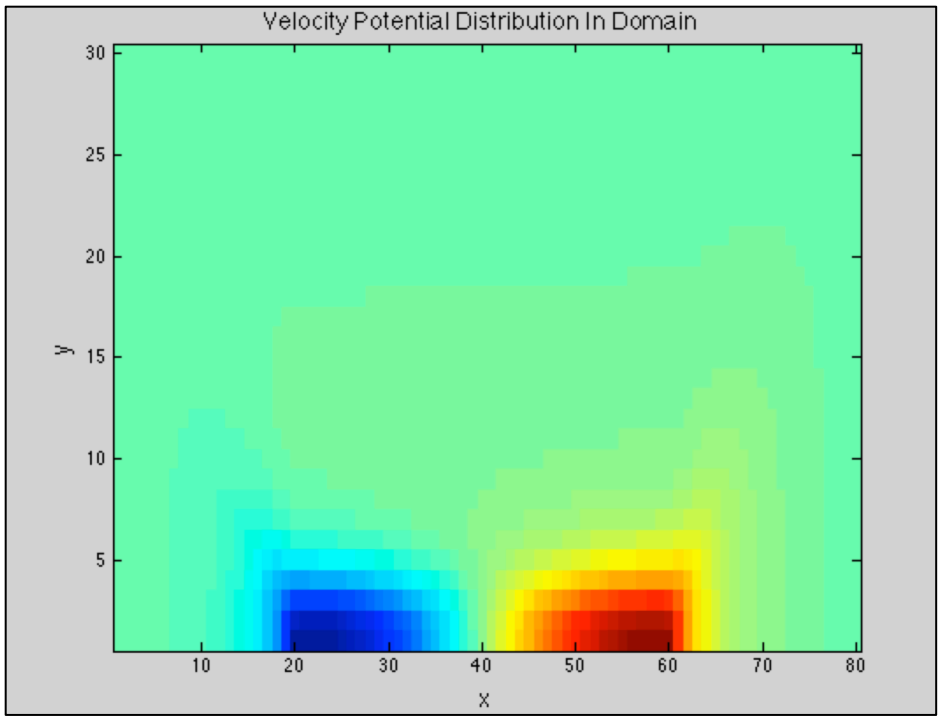
Question 1.

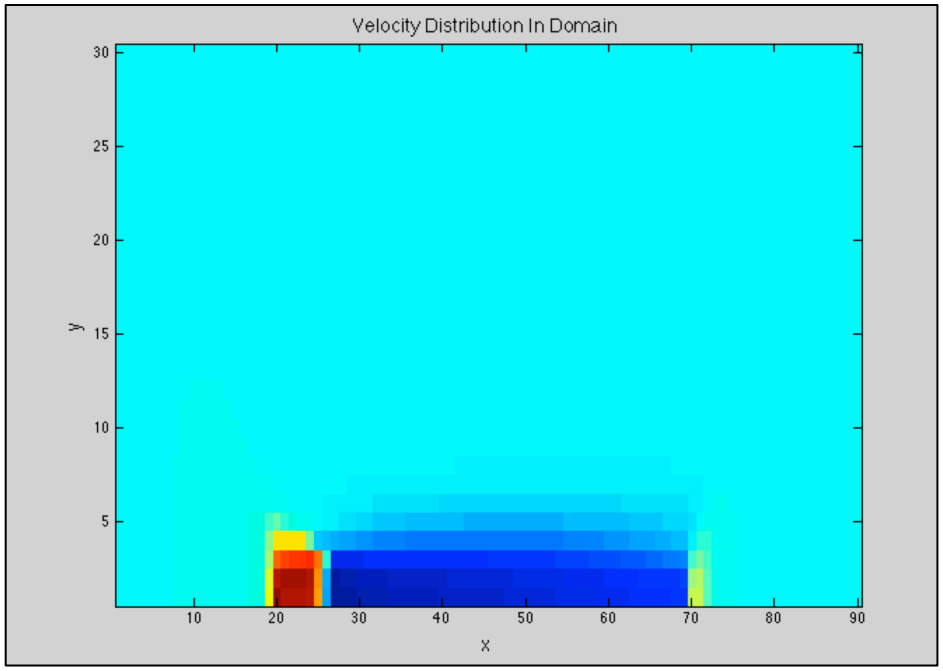
The grid was set up as follow:

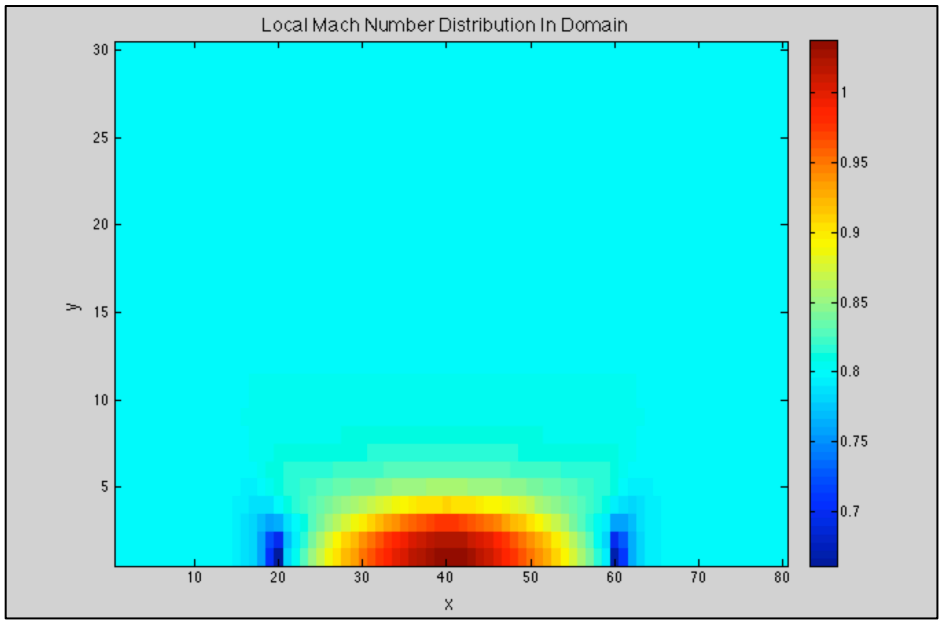
- 20 grid points before and after the wing airfoil in x direction with polynomial stretching
- 40 grid points along the wing airfoil in x direction with constant stretching
- 30 grid points in y direction with polynomial stretching

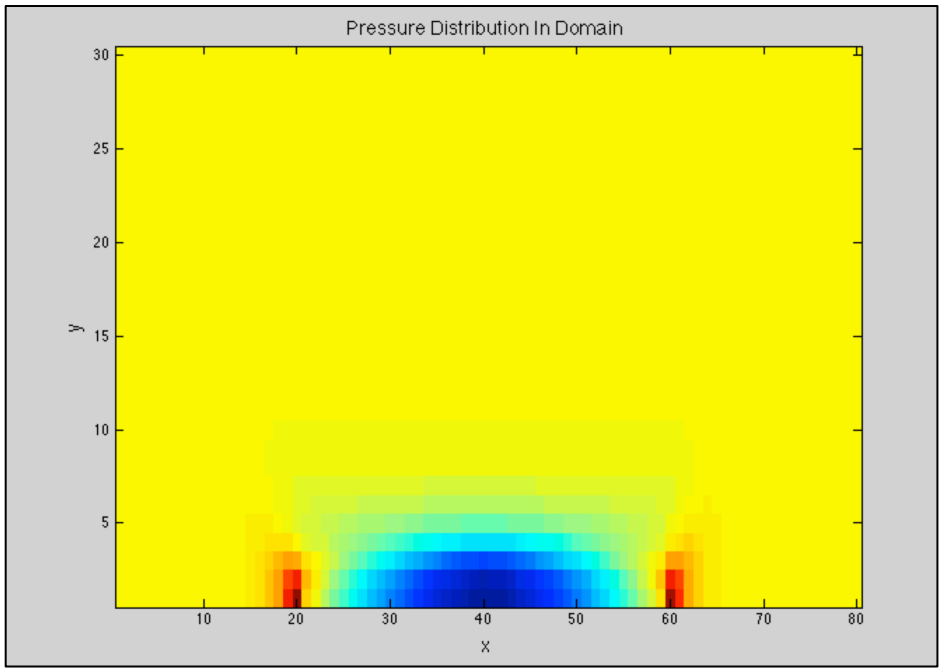
Free stream Mach number was set to 0.8.

Results:





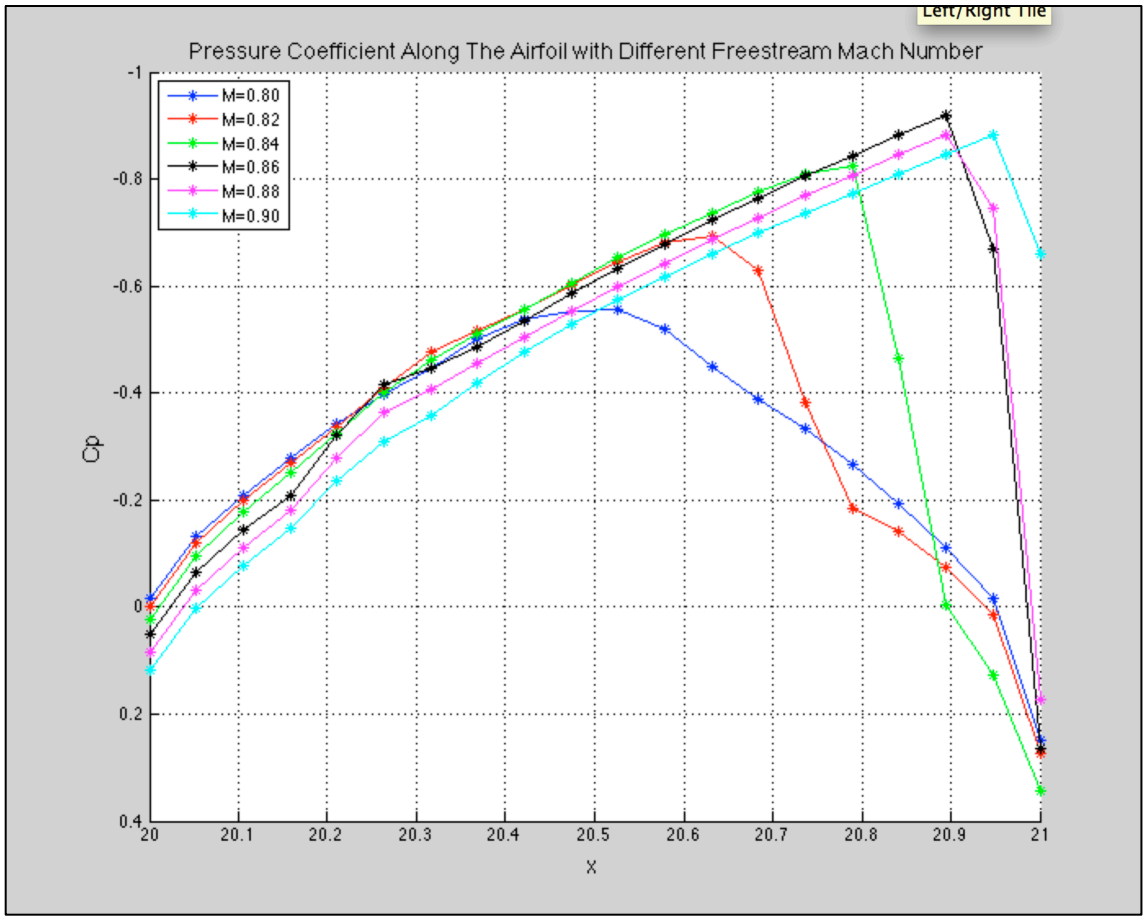

Local Mach Number Distribution In Domain
30
25
20
15
10
5
y
10
20
30
40
50
60
70
80
X
1
0.95
0.9
0.85
0.8
0.75
0.7


Pressure Distribution In Domain
30
25
20
15
10
5
y
10
20
30
40
50
60
70
80
X

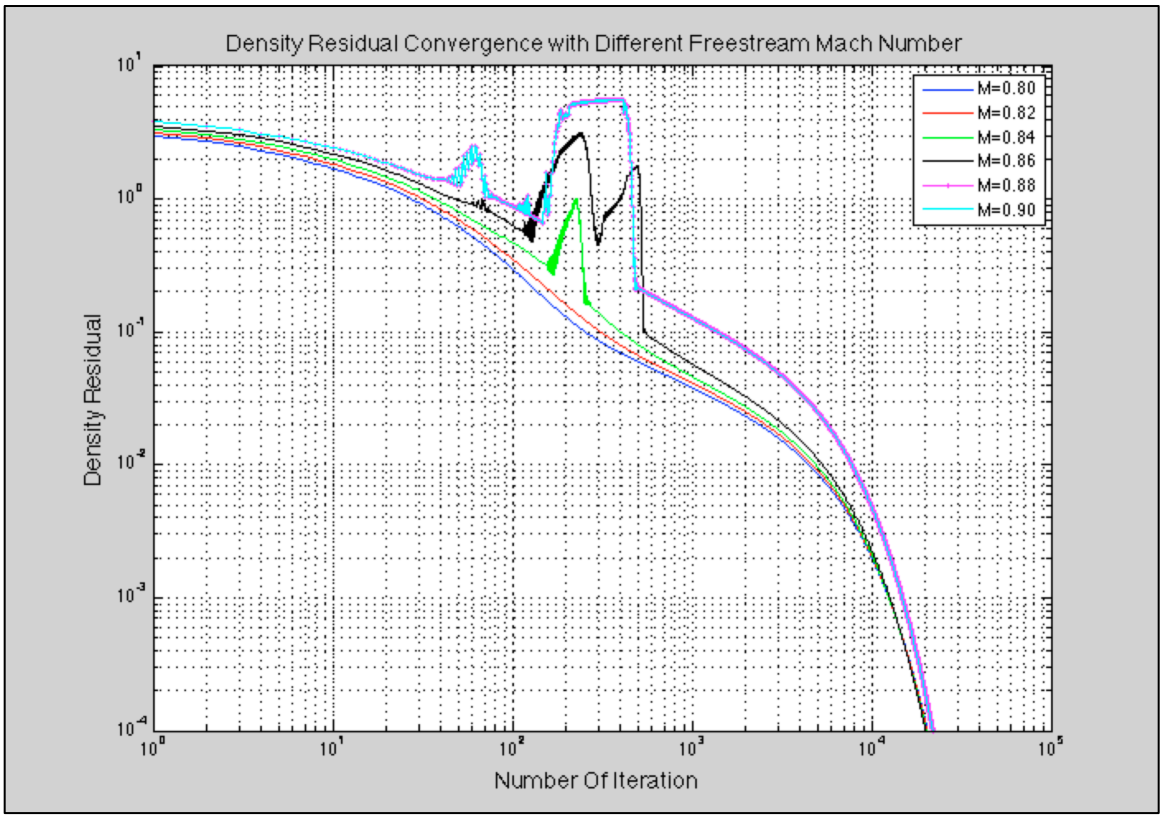
Question 2. (Corrected)

The grid was set up as follow: (x: 60 grid points, y: 20 grid points)

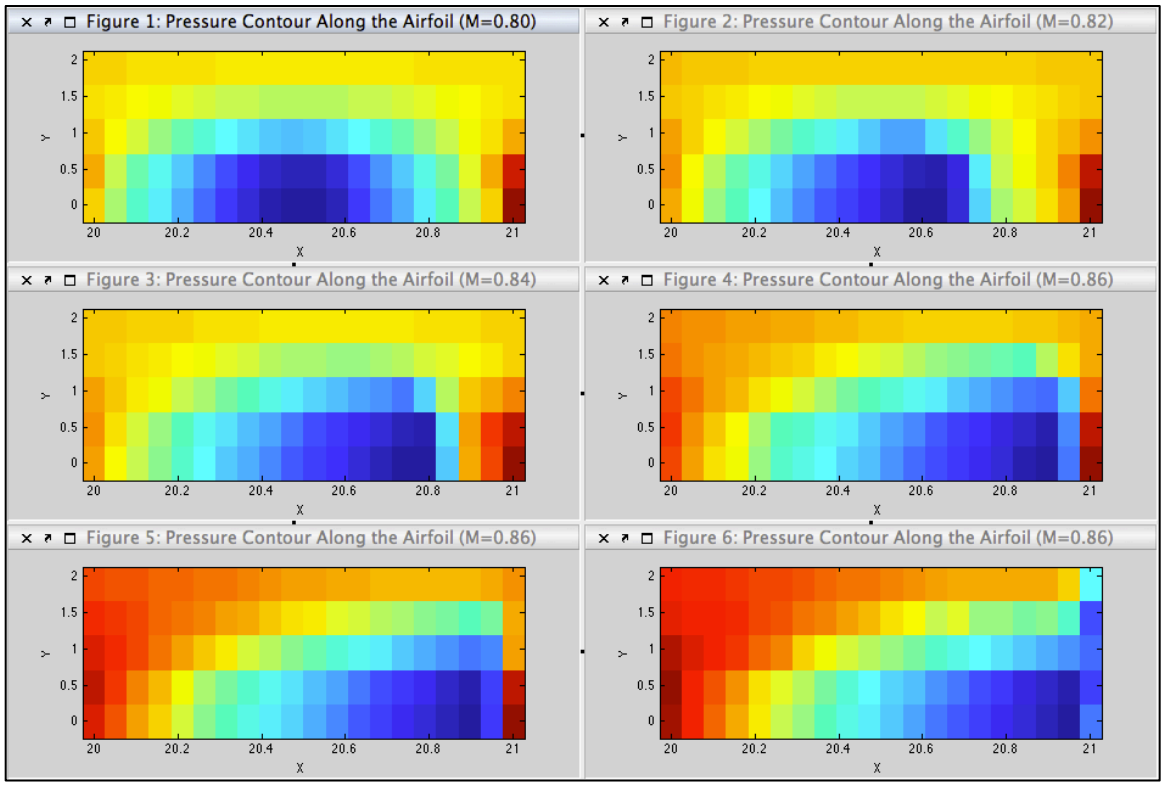
- 20 grid points before and after the wing airfoil in x direction with polynomial stretching
- 20 grid points along the wing airfoil in x direction with constant stretching
- 20 grid points in y direction with polynomial stretching



The plots of pressure coefficient along the airfoil at different freestream Mach number have been plotted. According to the graph, the region where the peak of the graph suddenly drops is where the shock occurs. So, as the freestream Mach number increases, the shock location was shifted right and the peak magnitude of pressure coefficient also increased.



Residual convergences at different freestream Mach number have been plotted with log-log scale. According to the graph, as the freestream Mach number increases, the residual convergence rate decreased and the number of iteration for convergence increased.

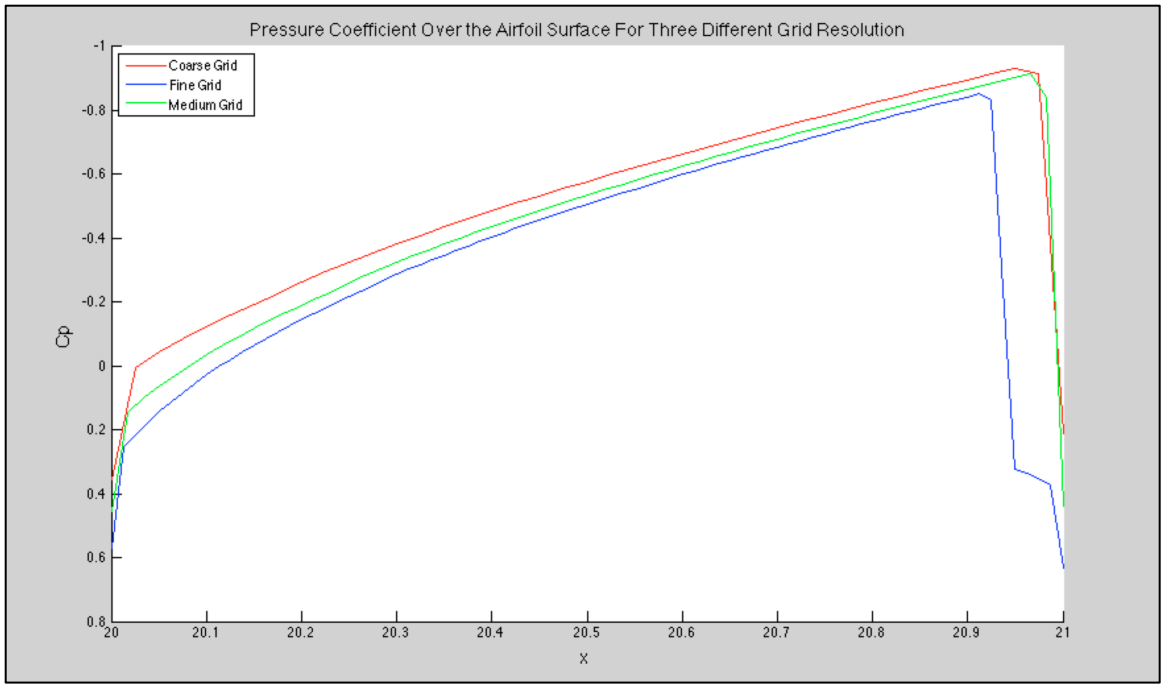


The pressure contours along the airfoil at different freestream Mach number have been plotted separately. According to the plots, the pressure distribution over the airfoil was symmetric at $M_\infty = 0.80$. It was noticeable that the maximum pressure location is shifted to right and the magnitude of pressure cofficient as the freestream Mach number increases. Also, the maximum pressure location corresponds to the shock location.

Question 3. (Corrected)

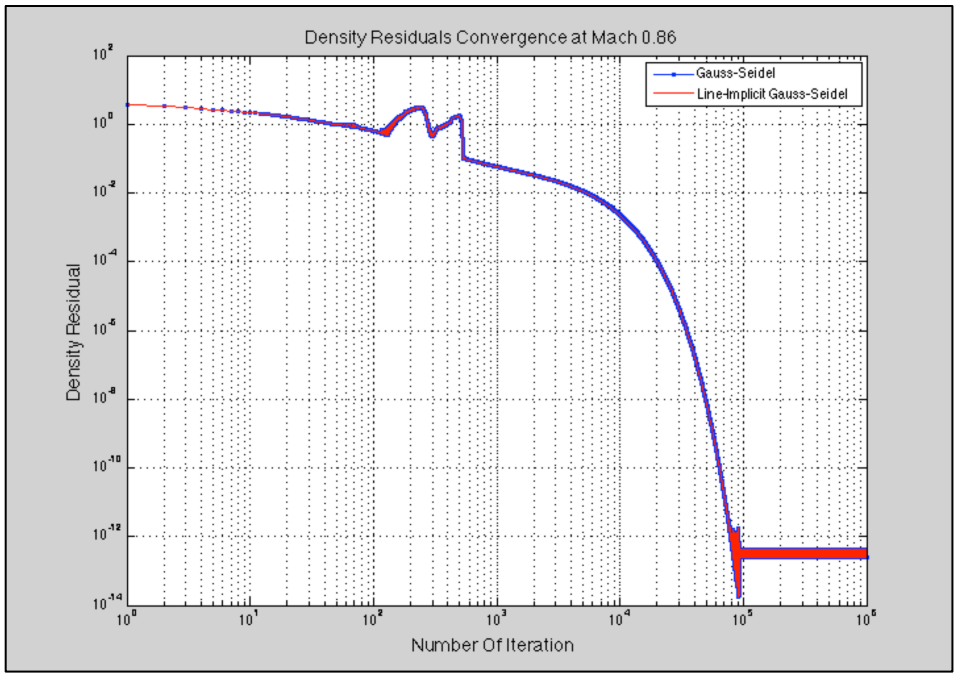
The grid was set up as follow:

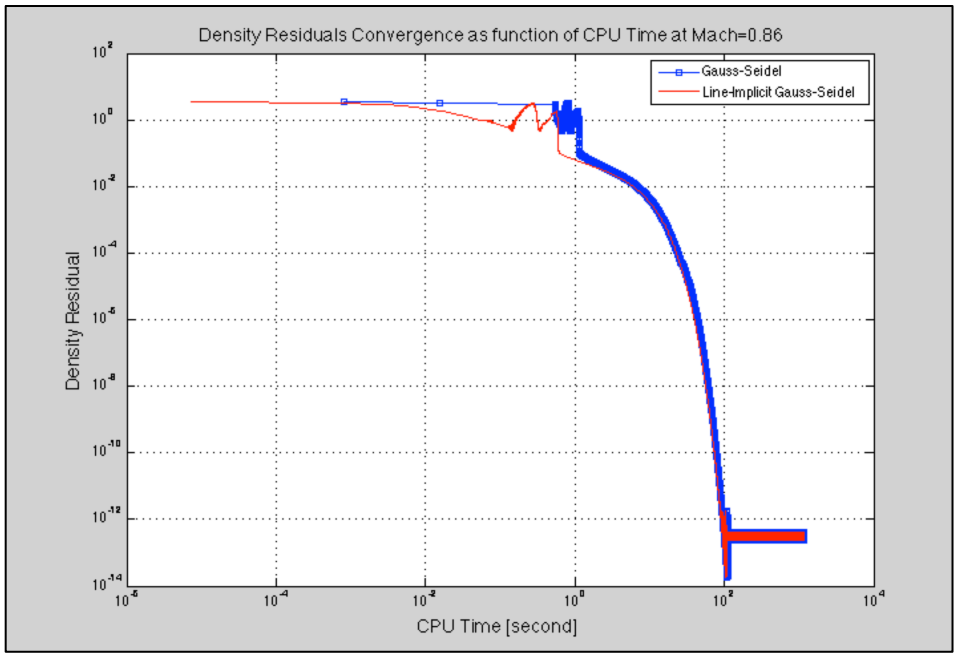
- Coarse Grid (x: 60 grid points, y: 50 grid points)
  - 10 grid points before and after the wing airfoil in x direction with polynomial stretching
  - 40 grid points along the wing airfoil in x direction with constant stretching
  - 50 grid points in y direction with polynomial stretching
- Medium Grid (x: 90 grid points, y: 70 grid points)
  - 15 grid points before and after the wing airfoil in x direction with polynomial stretching
  - 60 grid points along the wing airfoil in x direction with constant stretching
  - 70 grid points in y direction with polynomial stretching
- Fine Grid (x: 120 grid points, y: 100 grid points)
  - 20 grid points before and after the wing airfoil in x direction with polynomial stretching
  - 80 grid points along the wing airfoil in x direction with constant stretching
  - 100 grid points in y direction with polynomial stretching



For the finer grid, the magnitudes of pressure contour were quite the same and also the locations of the shock were quite the same that it was not affected by varying grid resolution.

Question 4. (Corrected)



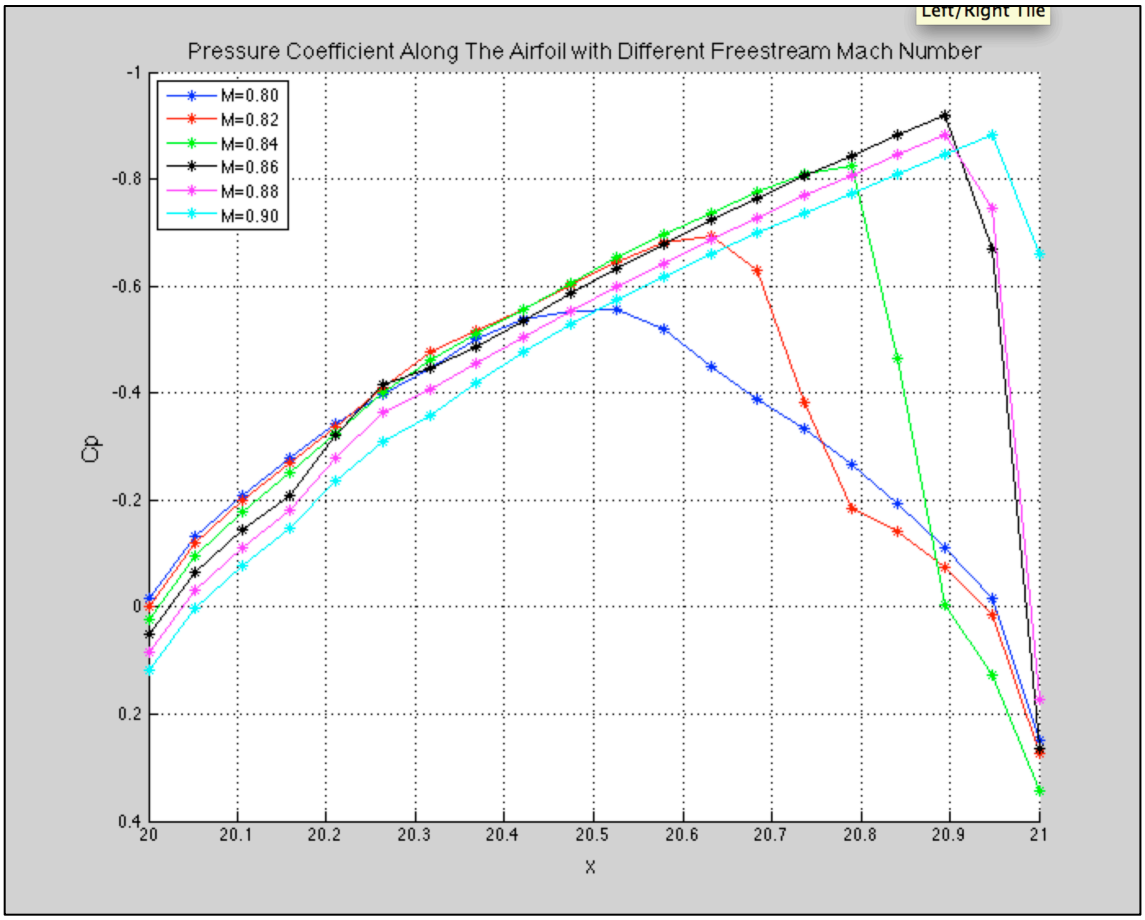


Required Number of iteration for both schemes was the same. For the CPU time, convergence rate of Line-Implicit Gauss-Seidel scheme was faster than that of Gauss-Seidel scheme for the beginning of iteration, but eventually, the convergence rates became the same.

Question 5.

The grid was set up as follow: (x: 60 grid points, y: 20 grid points)

- 20 grid points before and after the wing airfoil in x direction with polynomial stretching
- 20 grid points along the wing airfoil in x direction with constant stretching
- 20 grid points in y direction with polynomial stretching



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