TEST CASE DOCUMENTATION
AND TESTING RESULTS

TEST CASE ID CESE-BENCH-1.1

Inviscid transonic Flow around a NACA0012 Airfoil

Tested with LS-DYNA® v980 Revision Beta

Friday 1st June, 2012

Document Information	
Confidentiality	**external use**
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Contents

1 Introduction **1**
1.1 Purpose of this Document 1

2 Test Case Information **2**

3 Test Case Specification **3**
3.1 Test Case Purpose 3
3.2 Test Case Description 3
3.3 Model Description 3

4 Test Case Results **6**
4.1 Test Case observations 6

1 Introduction

1.1 Purpose of this Document

This document specifies the test case CESE-BENCH-1.1. It provides general test case information like name and ID as well as information to the confidentiality, status, and classification of the test case.

A detailed description of the test case is given, the purpose of the test case is defined, and the tested features are named. Results and observations are stated and discussed. Testing results are provided in section 4.1 for the therein mentioned LS-DYNA® version and platforms.

2 Test Case Information

Test Case Summary	
Confidentiality	external use
Test Case Name	Inviscid transonic Flow around a NACA0012 Airfoil
Test Case ID	CESE-BENCH-1.1
Test Case Status	Under consideration
Test Case Classification	Benchmarking
Metadata	SHOCK WAVES

Table 1: Test Case Summary

3 Test Case Specification

3.1 Test Case Purpose

The purpose of this benchmarking test case is to study the pressure coefficient over a NACA 0012 airfoil in a transonic inviscid flow.

3.2 Test Case Description

The NACA airfoils are airfoil shapes for aircraft wings developed by the National Advisory Committee for Aeronautics (NACA, former NASA) in the 1940s. The four digit series define the profile by describing its maximum camber, the location of the maximum camber and the airfoil's maximum thickness. The NACA 0012 is a symmetrical airfoil frequently used for benchmarking test cases.

In this test case, the Mach number is high enough to reach the so called transonic flow. Transonic flow occurs when there is mixed subsonic and supersonic local flow in the same flow field (typically with free stream Mach numbers from M = 0.7 or 0.8 to 1.3). When flowing over the airfoil's surface, the flow will accelerate and become locally supersonic resulting in potential normal shock waves on the upper camber and lower camber. As in this particular test case, the flow will be considered inviscid, no boundary layer will be developed on the airfoil's surface.

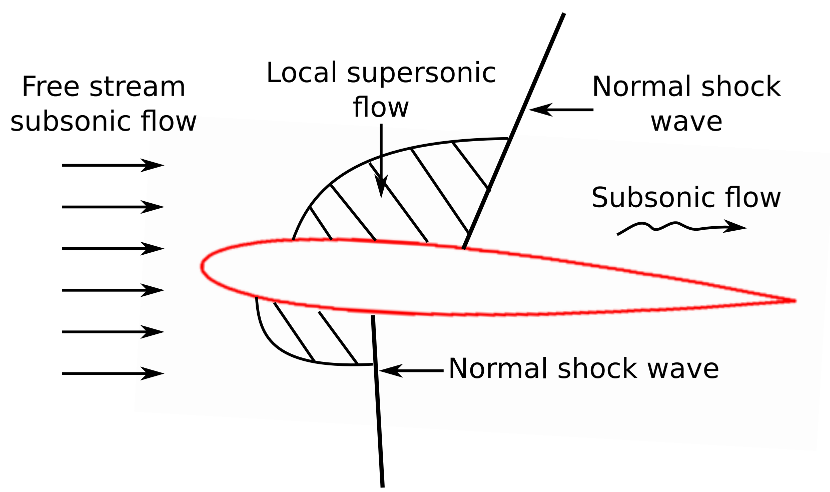

Figure 1: Inviscid transonic flow patterns

3.3 Model Description

The computational domain is [-4.6, 5.7] × [-4, 4] in the x-y plane and 200 elements are used on both upper and lower surfaces of the airfoil. The free stream Mach Number is 0.8, and the angle of attack is 1.25 degree. The chord length in unitary. Figure (2) offers a view of the geometry and mesh used while Table (2) and (3) give some information on the mesh and the parameters used.

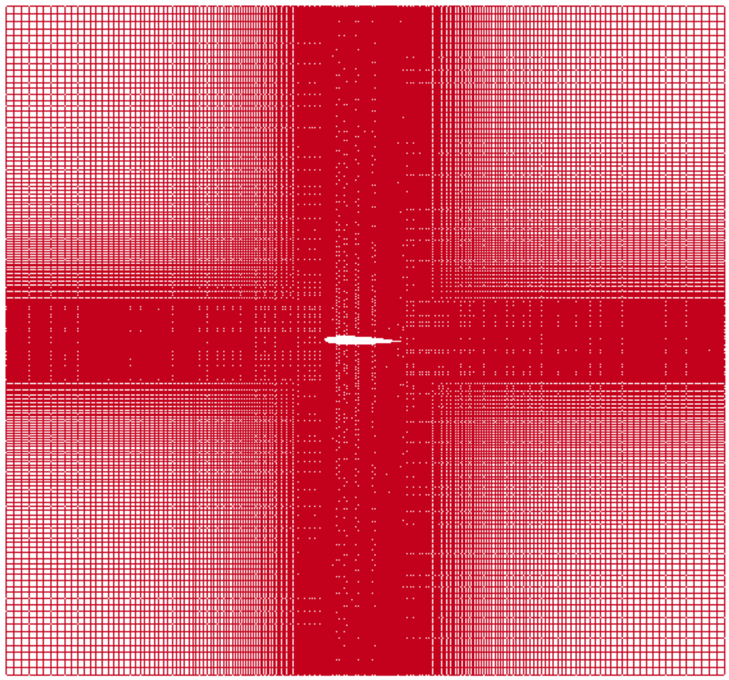

Figure 2: Test Case Geometry and Mesh

Model information	
Total number of nodes	145280
Total number of elements	72000

Table 2: Test Case Mesh information

Model physical parameters	
Specific heat at constant volume	2.790179
Specific heat at constant pressure	3.90625
Incoming velocity in the x-direction	1.0
Incoming velocity in the y-direction	0.0
Pressure	1.1160714
Density	1

Table 3: Test Case Parameters

4 Test Case Results

4.1 Test Case observations

Figure (3) shows the pressure fringes where the two normal shock waves on the upper and lower camber can both be identified. Figure (4) shows the pressure coefficient on the upper and lower surfaces. The results agree very well with those of [1].

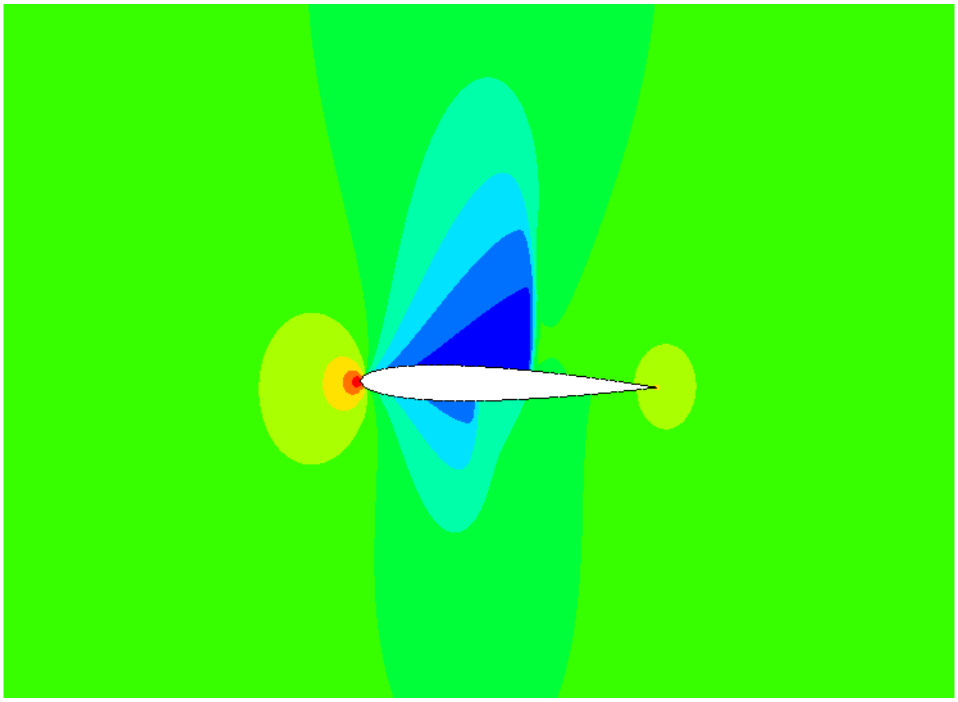

Figure 3: Pressure fringes

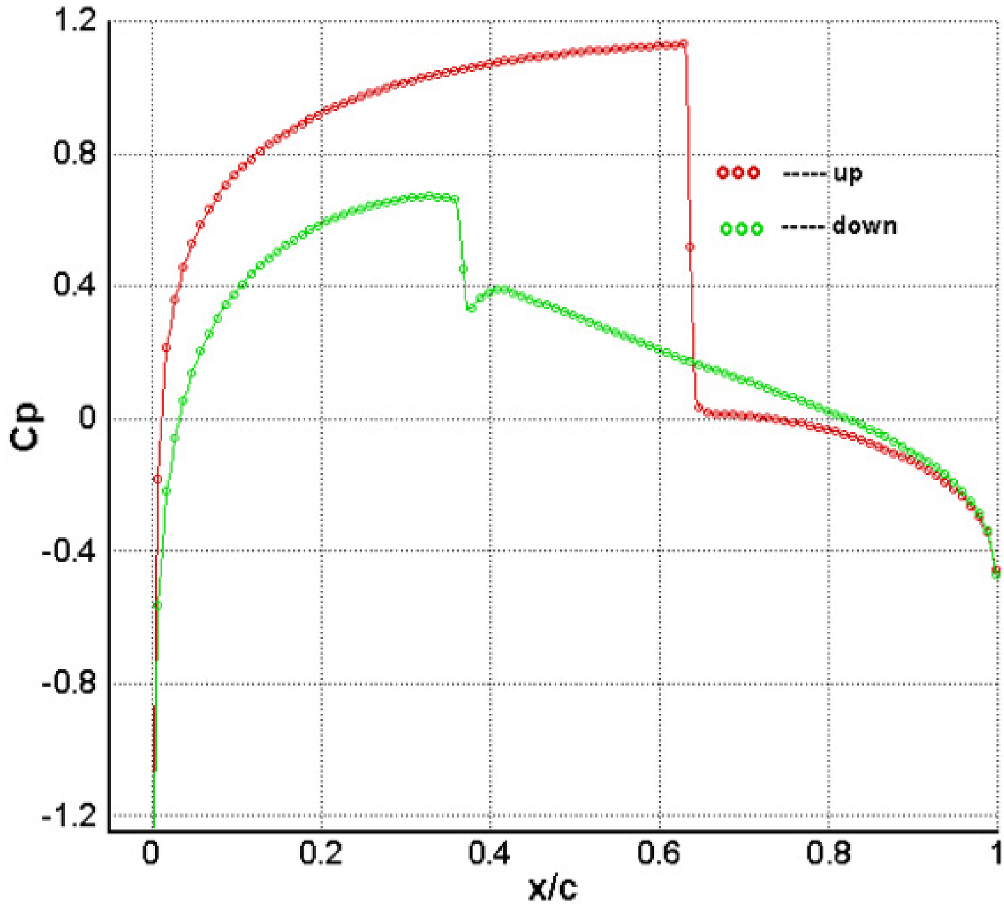

Figure 4: Pressure coefficient on upper and lower surfaces

References

[1] D. L. Bonhaus, *A HIGHER ORDER ACCURATE FINITE ELEMENT METHOD FOR VISCOUS COMPRESSIBLE FLOWS*, PhD thesis, Virginia Polytechnic Institute and State University, 1998.