ONERA M6 Wing Test-Case, Original and TMR

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1 Introduction: the fortune of M6

The first transonic wing test-case addressed in [1] (1977) with the Flo22 full potential equation solver was M6. Several flow conditions taken from [2] (1973, 1975) were considered, including those which will be preferred over the years by the majority of the CFD community, $M_\infty = 0.84, \alpha = 3.06°$.

The time of a paper, M6 was granted a fuselage to investigate fuselage effects on the flow past a wing [3] (1979), the authors being well aware that M6 is not even a full wing, but is only "representative of the tip panel of a relatively simple wing of conventional hight speed section shape" [1]. However, the typical double shock on the upper surface of the wing in the said aerodynamic conditions was considered a challenging feature for transonic simulations. In 1982, the editor of the Proceedings of IMA Conference on Numerical Methods in Aeronautical Fluid Dynamics could append to a paper entitled "Transonic Airfoil Calculations Using the Euler Equations" [4] Jameson's first published results of an Euler simulation past a wing, M6 of course, $M_\infty = 0.84, \alpha = 3.06°$, with a comparison to a full potential simulation.

In the late 80's and in the 90's, the ONERA-M6 test-case was used to validate (or invalidate) early 3D separated viscous flow simulations. A second set of conditions, $M_\infty = 0.84, \alpha = 6.06°, Re_{\overline{C}} = 11.7\ 10^6$, providing wide separation on the outer part of the wing, was selected by several authors [5] [6] [7] [8].

2 The M6 AGARD database

2.1 Summary of the 1979 AGARD database

In 1979, AGARD (NATO's Advisory Group for Aerospace Research & Development) issued Report AR-138 entitled "Experimental Data Base for Computer Program Assessment". The two most enduring test-cases of this rich and accurate compendium are the RAE2822 airfoil and the ONERA-M6 wing.

Schmitt and Charpin described the M6 test-case [10]. M6 borrows the shape of the outer third of the ONERA calibration model M5 wing. The generating airfoil is of the peaky type. The wing planform is defined by the layout reproduced here in figure 1. The wing is produced through conical generation from a unique symmetrical airfoil: all sections are homothetical. The upper surface wing root section is given in 72 point (x, z) non-dimensional coordinates in a table. Note that the data are those of a longitudinal section, not of the generating airfoil (ONERA-D) in the usual aerodynamical sense. Conventional mean aerodynamic chord for Reynolds number expression and reference area for aerodynamic coefficients are respectively $\overline{C} = 0.64607\,m$ and $A_{ref} = 0.7532\,m^2$.

Experimental data gathered in [10] were produced in the ONERA pressurized S2MA wind tunnel, figure 4. The height and width of the test section were $1.770\,m$ and $1.750\,m$ for a model of (semi)span $1.1963\,m$. The model was assumed to be indeformable. Transition was free. The setting included a boundary layer diverter (half model).

Data consist exclusively of pressure coefficient distributions: no lift, drag or moment coefficients, no flow visualizations, no unsteady data. Pressure distributions are given in 7 spanwise sections, for 30 aerodynamic conditions, ranging from $M_\infty \simeq 0.70, \alpha = 0°$ to $M_\infty \simeq 0.93, \alpha = 6°$, the Reynolds number being of the order of 11.7 Million.

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2.2 The original ONERA M6 wing geometry

The planform is sketched in figure 1 [10]. Let us denote A and B the root leading edge point and trailing edge point, and C the apex of the prolonged trapezoidal planform. Following the figure, one can adopt

$$\begin{aligned}(X_A, Y_A) &= (0.0000\,m, 0.0000\,m)\\ (X_B, Y_B) &= (0.8059\,m, 0.0000\,m)\\ (X_C, Y_C) &= (1.5787\,m, 2.7345\,m)\end{aligned}$$

and cut the triangle at

$$Y_{CUT} = b = 1.1963\,m$$

The leading edge and trailing edge sweep angles are 30°, 15.8° [10].

The reduced coordinates of the wing section printed in [10] are gathered in table 1 (upper half section, non-dimensional). The trailing edge of the original ONERA M6 wing has a 0.14104% base thickness.

The form of the wing tip is: "truncation parallel to the wing root and addition of a half body of revolution" [10].

2.3 Aerodynamic conditions

The 30 sets of aerodynamic conditions of [10] are listed in table 2.

Since [1], tradition has established $M_\infty = 0.84$ instead of $M_\infty = 0.8395$ for test number 2308 and since [5] $M_\infty = 0.84$ instead of $M_\infty = 0.8372$ for test number 2565. Although the difference would be very small, I suggest to follow tradition and to forget the value printed between brackets in the table.

2.4 Pressure coefficient distributions

The AGARD document contains chordwise pressure distributions in 7 spanwise sections for all 30 aerodynamic conditions. Section locations are given in table 3.

3 Proposed test-case for NASA-LRC Turbulent Modeling Resource

3.1 M6 CAD model creation

A CAD model based on the initial discrete data described in section 2.2 was created. CATIA, Dassault System software, was used to generate the CAD model. The strategy adopted during the surface generation was to keep as much points from the initial discrete model. In order to generate a smooth shape, we had to remove few points that were responsible for oscillation. The process is described in details in [9]. The model is presented in figure 2.

3.2 M6sharp model with closed trailing edge

A shape derived from the original ONERA M6 wing has been requested for the NASA-TMR with a zero thickness trailing edge. The CAD model of this shape was generate from the model with base presented in section **??**. A prolongation of all surfaces joining the wing base was done as can be seen on figure 3. The wing chordwise extension is of 0.55 percent of local chord. The trailing edge created is a straight line. This CAD model will be available on the TMR website, as well as the model with base.

3.3 Proposed aerodynamic conditions

In order to avoid inefficient scatter of the CFD contributions, it has been proposed to limit the number of aerodynamic conditions to the two cases mentioned in the introduction:
1) the well known attached case with two shocks merging, $M_\infty = 0.84, \alpha = 3.06°, Re_{\overline{C}} = 11.72\,10^6$, test

x/c	z/c	x/c	z/c
0.0000000	0.0000000	0.3761446	0.0489296
0.0000165	0.0006914	0.4018567	0.0488202
0.0000696	0.0014416	0.4274223	0.0484833
0.0001675	0.0022554	0.4528441	0.0479351
0.0003232	0.0031382	0.4781197	0.0471661
0.0005508	0.0040959	0.5032514	0.0461903
0.0008657	0.0051343	0.5282426	0.0450209
0.0012868	0.0062598	0.5530937	0.0436741
0.0018364	0.0074784	0.5778043	0.0421684
0.0025441	0.0087958	0.6023757	0.0405241
0.0034428	0.0102163	0.6268104	0.0387613
0.0045704	0.0117419	0.6511093	0.0368990
0.0059751	0.0133708	0.6752726	0.0349542
0.0077112	0.0150951	0.6993027	0.0329402
0.0098413	0.0168984	0.7231995	0.0308662
0.0124479	0.0187537	0.7469658	0.0287365
0.0156171	0.0206220	0.7705998	0.0265505
0.0194609	0.0224545	0.7941055	0.0243027
0.0241067	0.0242004	0.8174828	0.0219842
0.0297008	0.0258245	0.8407324	0.0195838
0.0364261	0.0273317	0.8638564	0.0170915
0.0444852	0.0287912	0.8868235	0.0145051
0.0541248	0.0303278	0.9061905	0.0122389
0.0656303	0.0320138	0.9225336	0.0102727
0.0793366	0.0338372	0.9363346	0.0085827
0.0956354	0.0357742	0.9479946	0.0071423
0.1149796	0.0377923	0.9578511	0.0059224
0.1378963	0.0398522	0.9661860	0.0048907
0.1649976	0.0419089	0.9732361	0.0040180
0.1919327	0.0436214	0.9792020	0.0032796
0.2187096	0.0450507	0.9842508	0.0026547
0.2453310	0.0462358	0.9885252	0.0021257
0.2717978	0.0471987	0.9921438	0.0016778
0.2981113	0.0479494	0.9952080	0.0012985
0.3242726	0.0484902	0.9978030	0.0009773
0.3502830	0.0488183	1.0000000	0.0007052

Table 1: M6 wing section non dimensional coordinates

2308,
2) and a case with extended flow separation on the outer part of the wing, $M_\infty = 0.84, \alpha = 6.06°, Re_{\overline{C}} = 11.71\,10^6$, test 2565.

Chordwise pressure coefficient distributions in the 7 aboved-mentioned spanwise sections have been converted from [10] into an ASCII Tecplot format: see files case_2308.dat and case_2565.dat on the TMR site and, here, figures 5 and 6.

References

[1] JAMESON, A. and CAUGHEY, D.A., "Numerical Calculation of the Transonic Flow Past a Swept Wing", *ERDA Research and Development Report, Mathematics and Computing*, New York University, June 1977.

[2] MONNERIE, B. and CHARPIN, F., "Essais de Buffeting d'une Aile en Flèche en Transsonique,"

test number	M_∞	α (°)	$Re_{\overline{C}}$ (Million)
2309	0.6998	0.04	11.74
2551	0.6977	0.06	11.67
2310	0.7003	1.08	11.74
2311	0.7001	2.06	11.74
2312	0.6990	3.06	11.74
2313	0.7009	4.08	11.77
2541	0.7019	5.06	11.66
2542	0.6971	6.09	11.63
2305	0.8399	0.04	11.72
2396	0.8371	0.03	11.69
2306	0.8398	1.07	11.71
2307	0.8386	2.06	11.72
2308	**0.8400** [0.8395]	3.06	11.72
2563	0.8359	4.08	11.81
2564	0.8447	5.06	11.78
2565	**0.8400** [0.8372]	6.06	11.71
2300	0.8840	0.03	11.71
2301	0.8833	1.08	11.77
2302	0.8803	2.05	11.78
2304	0.8809	3.06	11.77
2591	0.8831	4.07	11.78
2592	0.8808	5.07	11.78
2593	0.8868	6.07	11.83
2296	0.9207	0.03	11.78
2297	0.9208	1.07	11.79
2298	0.9180	2.05	11.76
2299	0.9190	3.07	11.77
2583	0.9262	4.08	11.73
2584	0.9181	5.06	11.69
2585	0.9298	6.07	11.71

Table 2: M6 test-case aerodynamic conditions (from [10]).

section number	$\eta = Y/b$
1	0.20
2	0.44
3	0.65
4	0.80
5	0.90
6	0.96
7	0.99

Table 3: Spanwise locations of chordwise pressure coefficient distributions

10ème Colloque d'Aérodynamique Appliquée, Institut de Mécanique des Fluides, Lille, Novembre 1973 (also in *L'Aéronautique et l'Astronautique*, n° 50 (1975-1), pp.3-16).

[3] JAMESON, A. and CAUGHEY, D.A., "Recent Progress in Finite-Volume Calculations for Wing-Fuselage Combinations", AIAA Paper 79-1513, *AIAA 12th Fluid and Plasma Dynamics Conference,* Williamsburg, VA, July 23-25, 1979.

[4] JAMESON, A., "Transonic Airfoil Calculations Using the Euler Equations", in *Proceedings of IMA Conference on Numerical Methods in Aeronautical Fluid Dynamics*, March 1981, P.L. Roe editor,

Academic Press, 1982, pp.289-308.

[5] VATSA, V.N., "Accurate Numerical Solutions for Transonic Viscous Flow over Finite Wings", *Journal of Aircraft,* Vol. 24, No. 6, June 1987, pp. 377-385.

[6] ABID, R., VATSA, V.N., JOHNSON, D.A., WEDAN, B.W., "Prediction of Sepatared Transonic Wing Flows with a Non-Equilibrium Agebraic Model", AIAA Paper 89-0558, *AIAA 27th Aerospace Sciences Meeting,* Reno, NV, January 9-12, 1989.

[7] RADESPIEL, R., ROSSOW, C., SWANSON, R.C., "An Efficient Cell-Vertex Multigrid Scheme for the Three-Dimensional Navier-Stokes Equations", AIAA Paper 89-1953, *AIAA 9th Computational Fluid Dynamics Conference,* Buffalo, NY, June 14-16, 1989.

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[9] MAYEUR, J., DUMONT, A., DESTARAC, D., GLEIZE, V., "RANS simulations on TMR test cases and M6 wing with the Onera elsA flow solver", AIAA Paper 2015-1745, AIAA Scitech 2015, Kissimmee, Florida, USA

[10] SCHMITT, V. and CHARPIN, F., "Pressure Distributions on the ONERA-M6-Wing at Transonic Mach Numbers," *Experimental Data Base for Computer Program Assessment*, AGARD Avisory Report AR-138, May 1979.

Figures

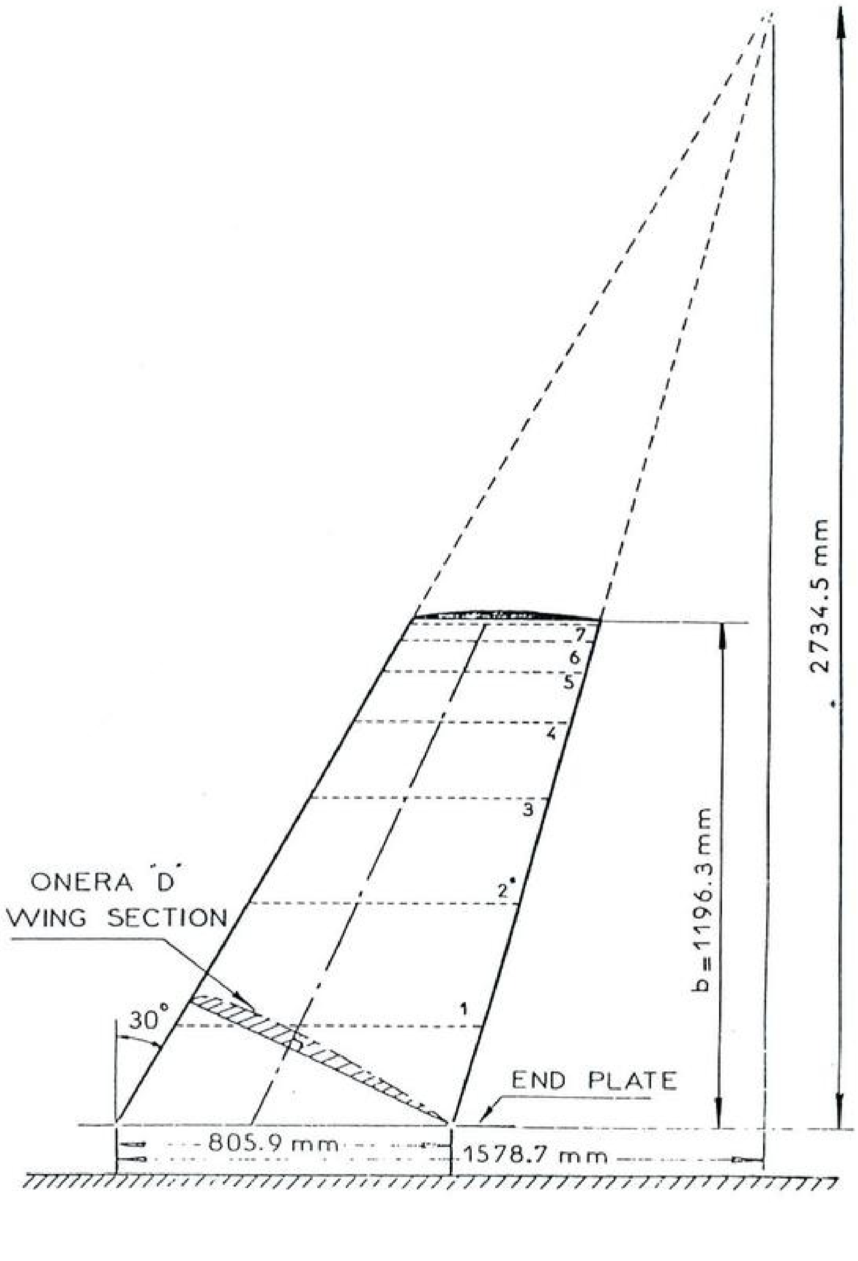

Figure 1: Layout of the M6 wing planform (from [10]).

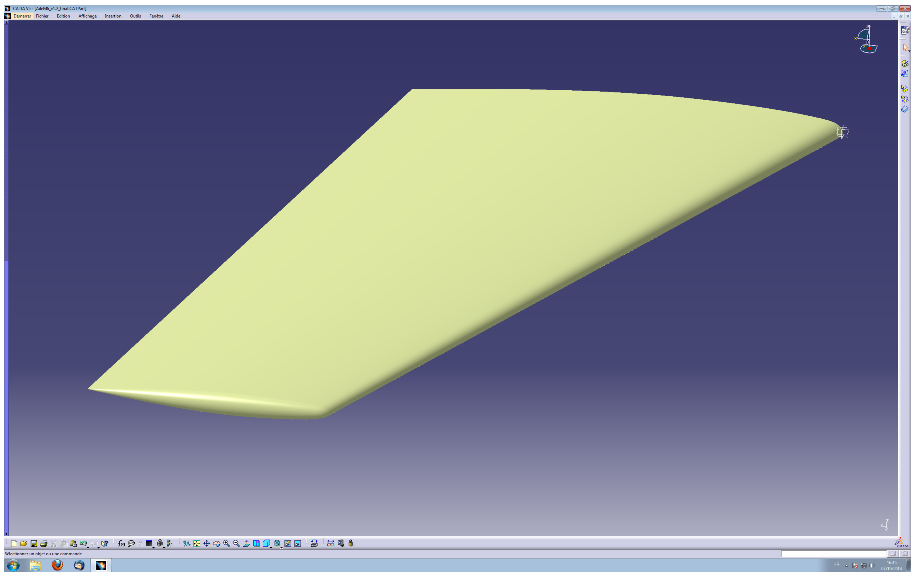

Figure 2: M6 wing CAD model.

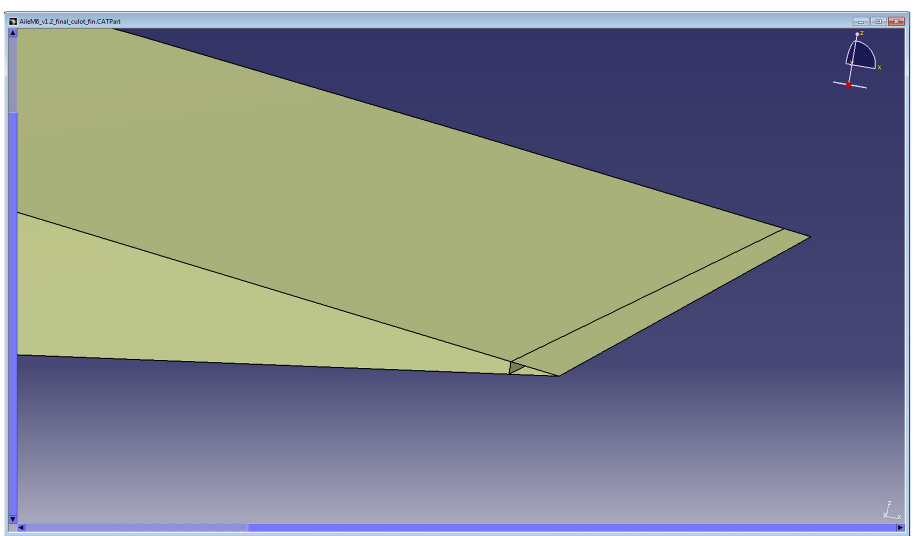

Figure 3: CAD creation of M6 wing trailing edge.

Figure 4: The M6 wing model in the ONERA-S2MA wind tunnel (from [10]).

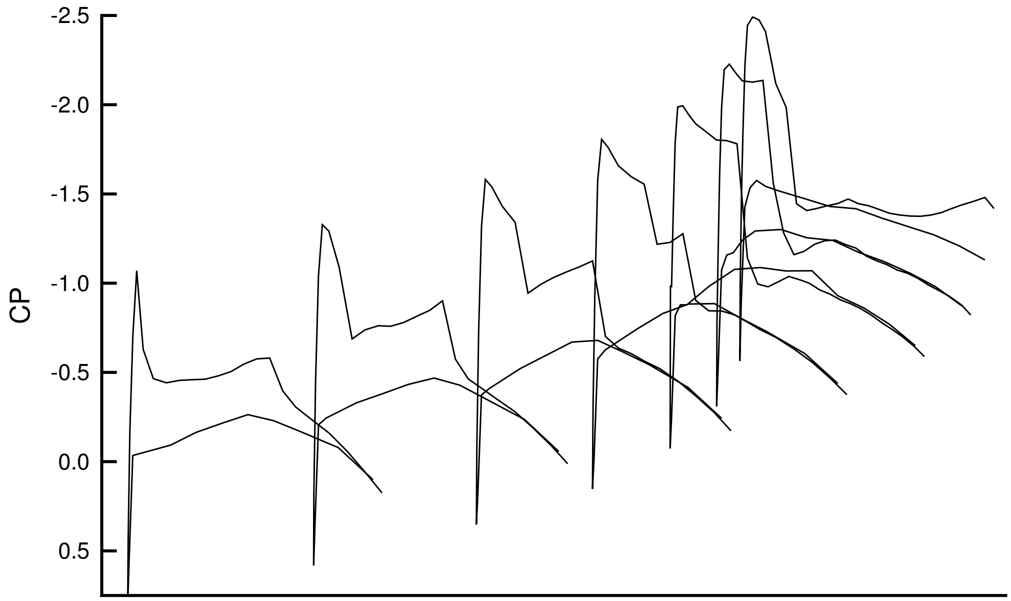

Figure 5: Test number 2308; $M_\infty = 0.84, \alpha = 3.06°, Re_{\overline{C}} = 11.72\,10^6$; pressure coefficient distributions.

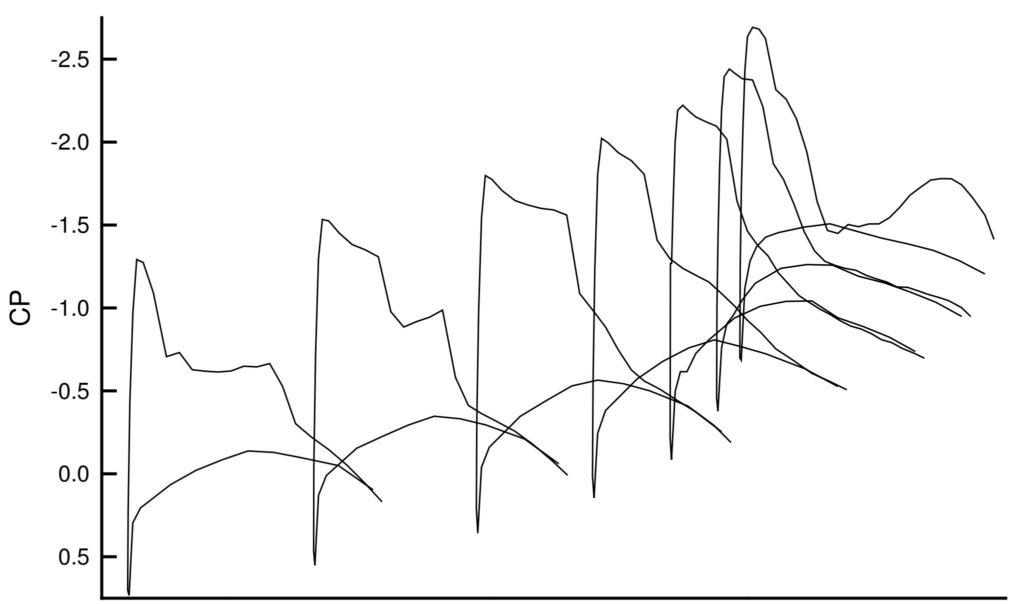

Figure 6: Test number 2565; $M_\infty = 0.84, \alpha = 6.06°, Re_{\overline{C}} = 11.73\,10^6$; pressure coefficient distributions.