Select ADO-DG Case Studies in Aerodynamic Design Optimization

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OUTLINE

- **NACA0012-ADO INVISCID AIRFOIL**
  - ADO-DG Case 1
  - SYN83 Optimizations
  - FLO82 Cross Analyses
  - GSA Study
  - Case Summary
- **ADO-CRM-WING**
  - ADO-DG Cases 4.1-4.3
  - SYN107 Optimizations
  - Case Summary

NACA0012-ADO MODEL PROBLEM

- **Optimization Statement**
  - Minimize Drag
  - $M = 0.85$, $\alpha = 0^\circ$, Inviscid Flow
  - Maintain or Exceed Thickness Distribution of Baseline
- NACA0012-ADO Airfoil Equation
  - Closed Trailing-Edge at $x = 1$

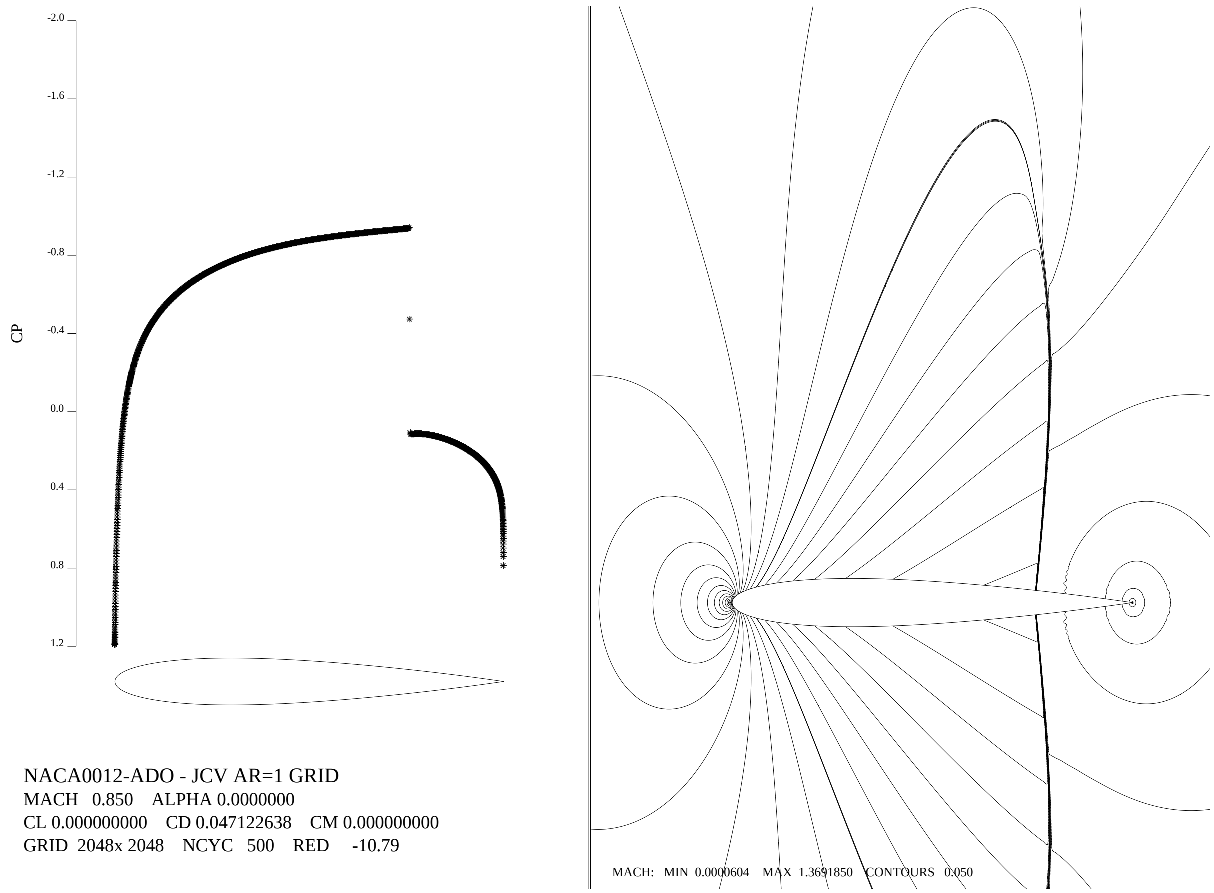
$$y_A(x) = \pm\frac{0.12}{0.2}\left(0.2969\sqrt{x} - 0.1260x - 0.3516x^2 + 0.2843x^3 - 0.1036x^4\right)$$

ADO-DG CASE 1: NACA0012-ADO

- **Optimization Methods**
  - SYN83
  - GSA Design Space Survey

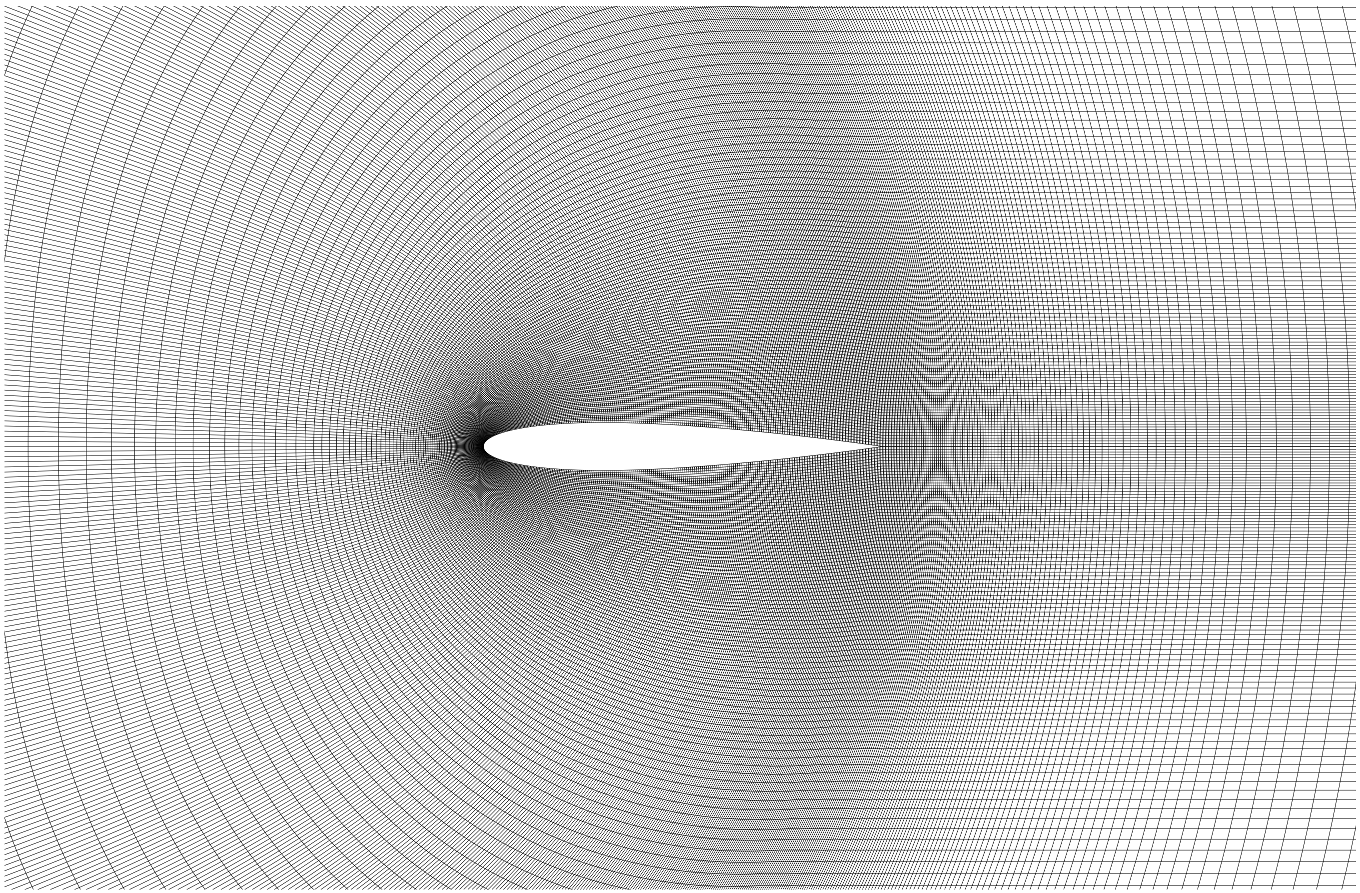
- **FLO82 Cross-Analysis**
  - Vassberg & Jameson, "In Pursuit of Grid Convergence for Two-Dimensional Euler Solutions," AIAA Journal of Aircraft, Vol.47, No.4, pp.1152-1166, July-August, 2010.
  - High-Quality O-Mesh with Aspect-Ratio-1 Cells
  - Family of Meshes (32x32)-to-(2048x2048) Cells
  - Converge Residuals to Machine-Level Zero
  - Richardson Extrapolation

BASELINE NACA0012-ADO SOLUTION



FLO82 Solution for NACA0012-ADO Airfoil at $M = 0.85$, $\alpha = 0^\circ$.

SYN83 OPTIMIZATION



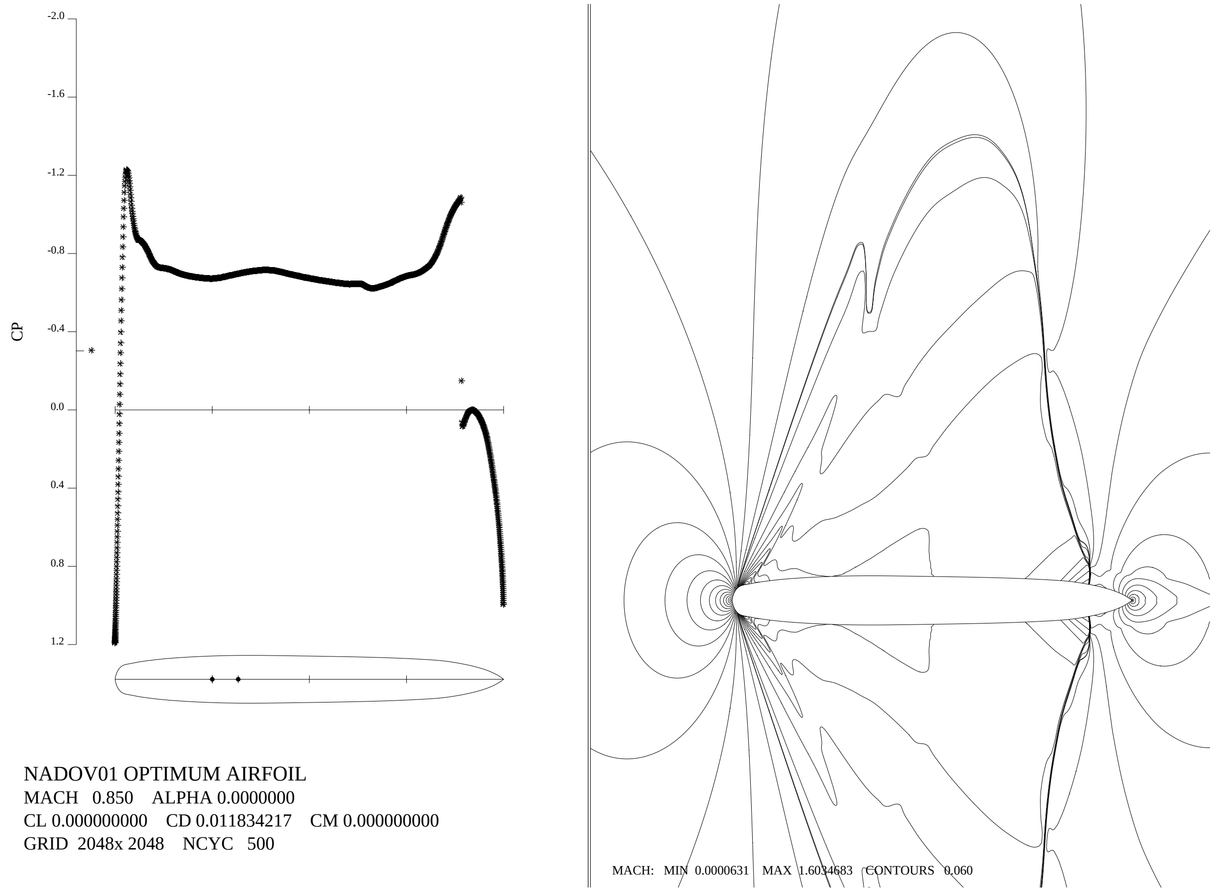
SYN83 C-mesh (768x128) about NACA0012-ADO.

SYN83 OPTIMIZATION

SYN83 Results (C_d in counts).

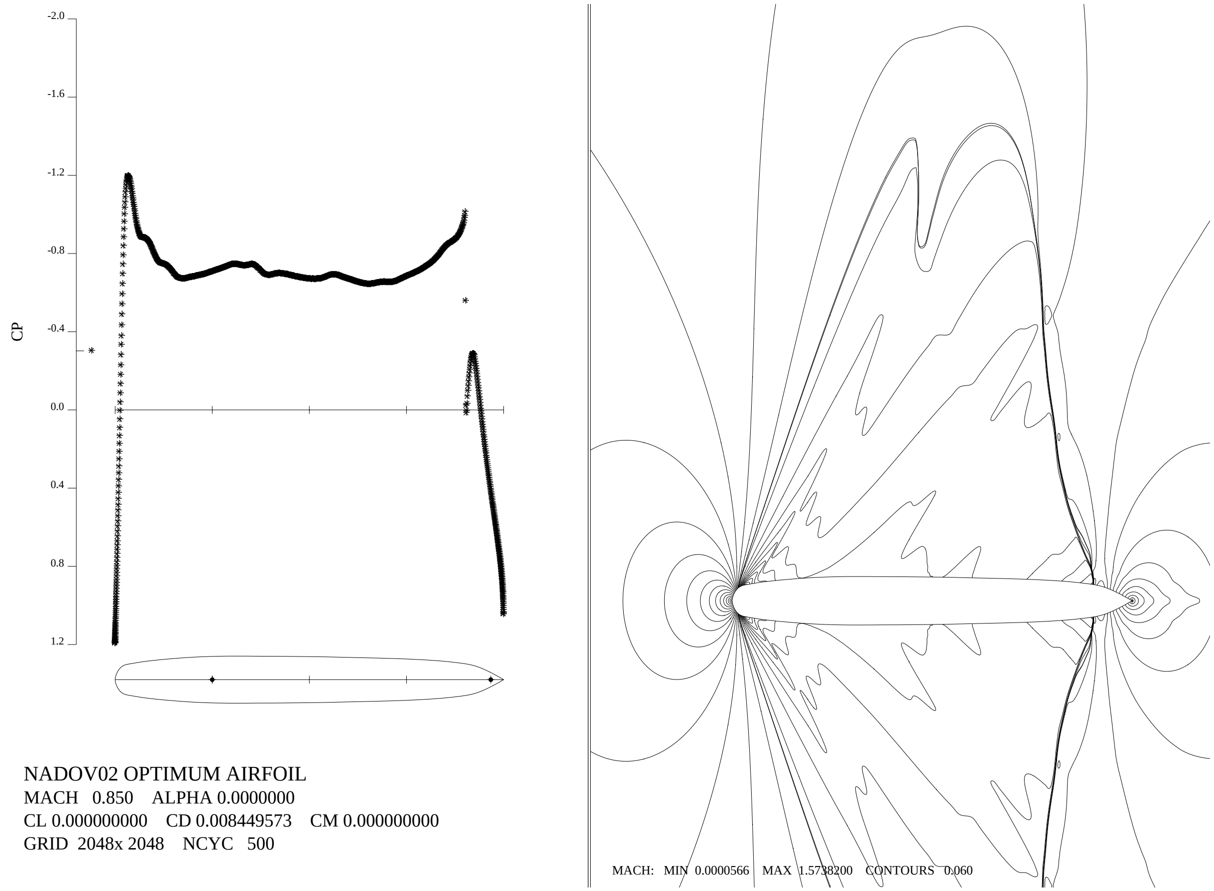
Airfoil		C_d	ΔC_d
Seed	NACA0012-ADO	456.34	-
Design	SYNA	103.71	-352.63
Seed	SEEDB	101.79	-354.55
Design	SYNB	79.31	-377.03

SYN83 OPTIMIZATION



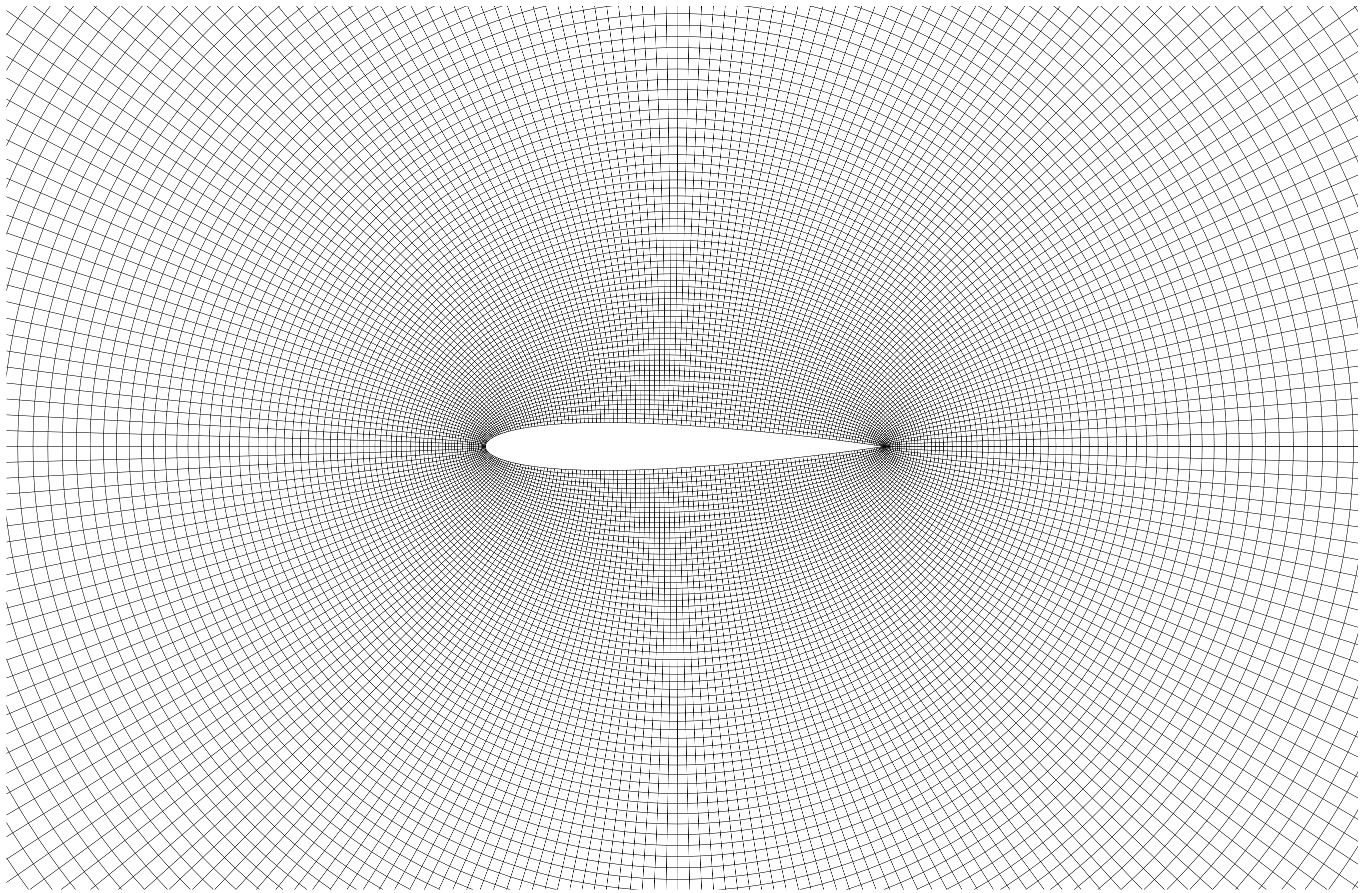
FLO82 Solution for SYNA Airfoil at $M = 0.85$, $\alpha = 0^\circ$.

SYN83 OPTIMIZATION



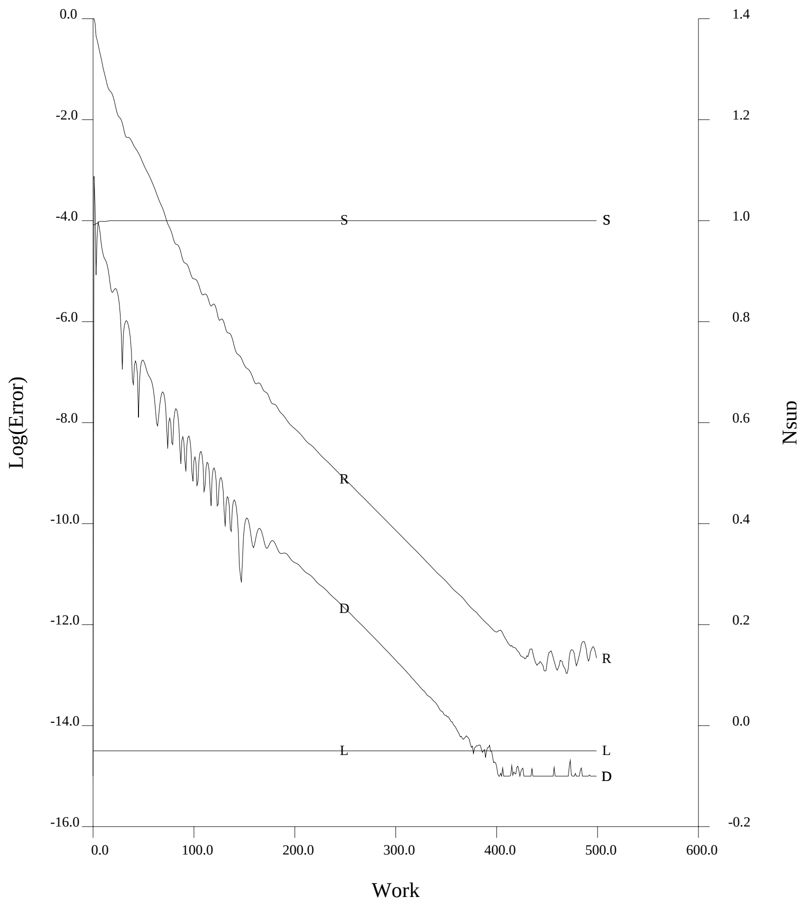
FLO82 Solution for SYNB Airfoil at $M = 0.85$, $\alpha = 0^{\circ}$.

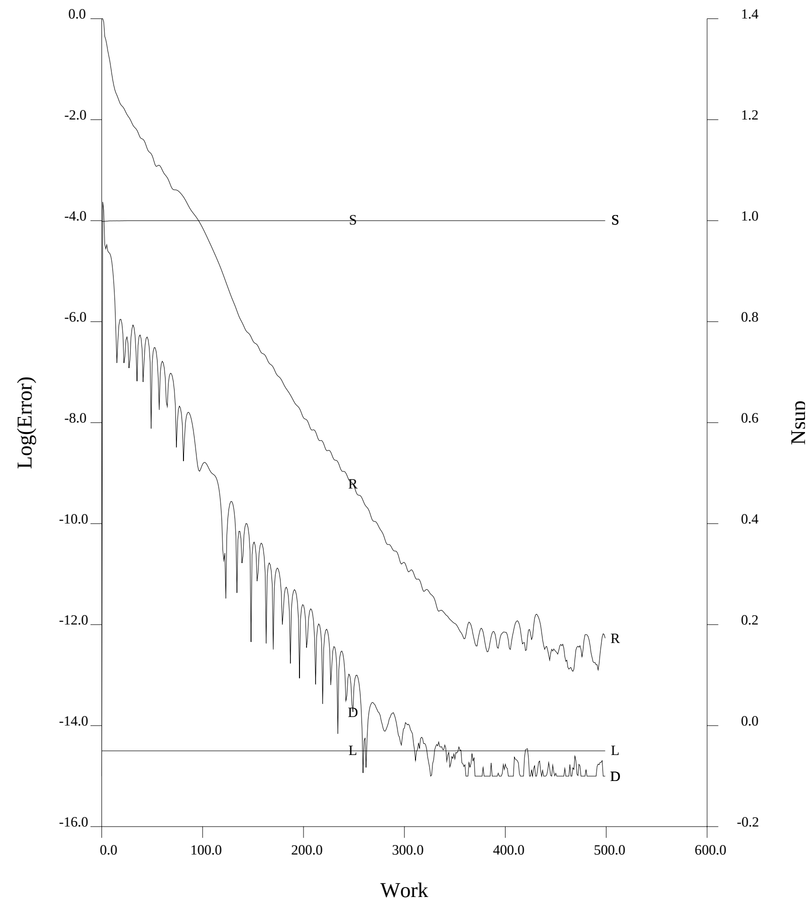
FLO82 O-MESH



Close-up view of the NACA0012-ADO 256x256 O-mesh.

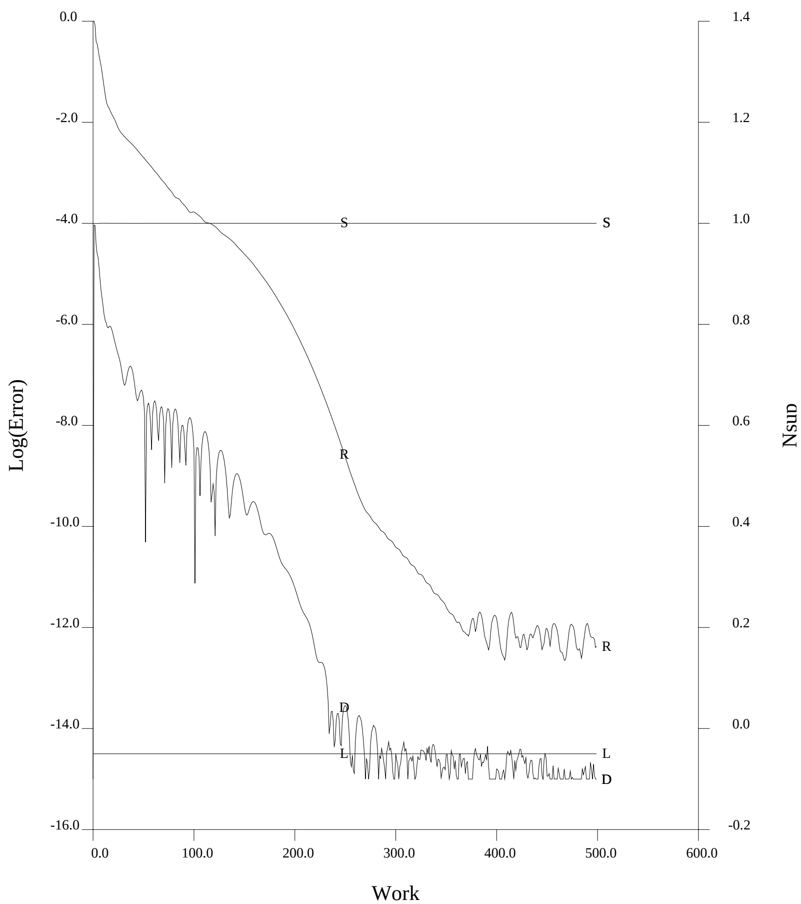
(256 x 256)   (512 x 512)



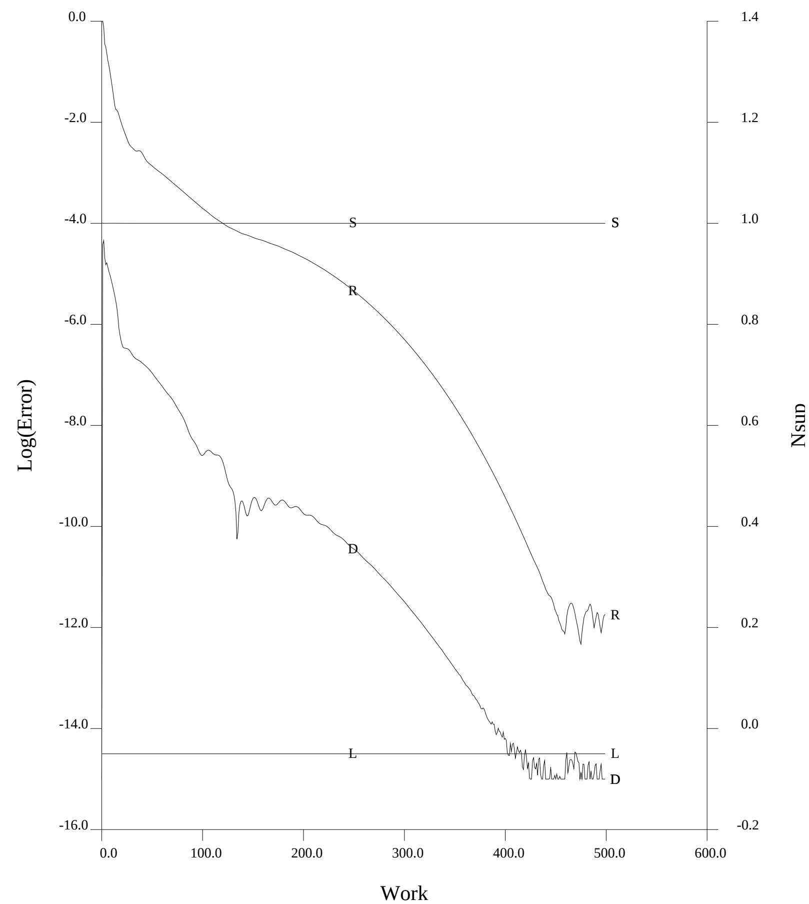


FLO82 Convergence Histories at $M = 0.85$, $\alpha = 0^\circ$.

(1024 x 1024)



(2048 x 2048)



FLO82 Convergence Histories at $M = 0.85$, $\alpha = 0^\circ$.

FLO82 CROSS-ANALYSIS

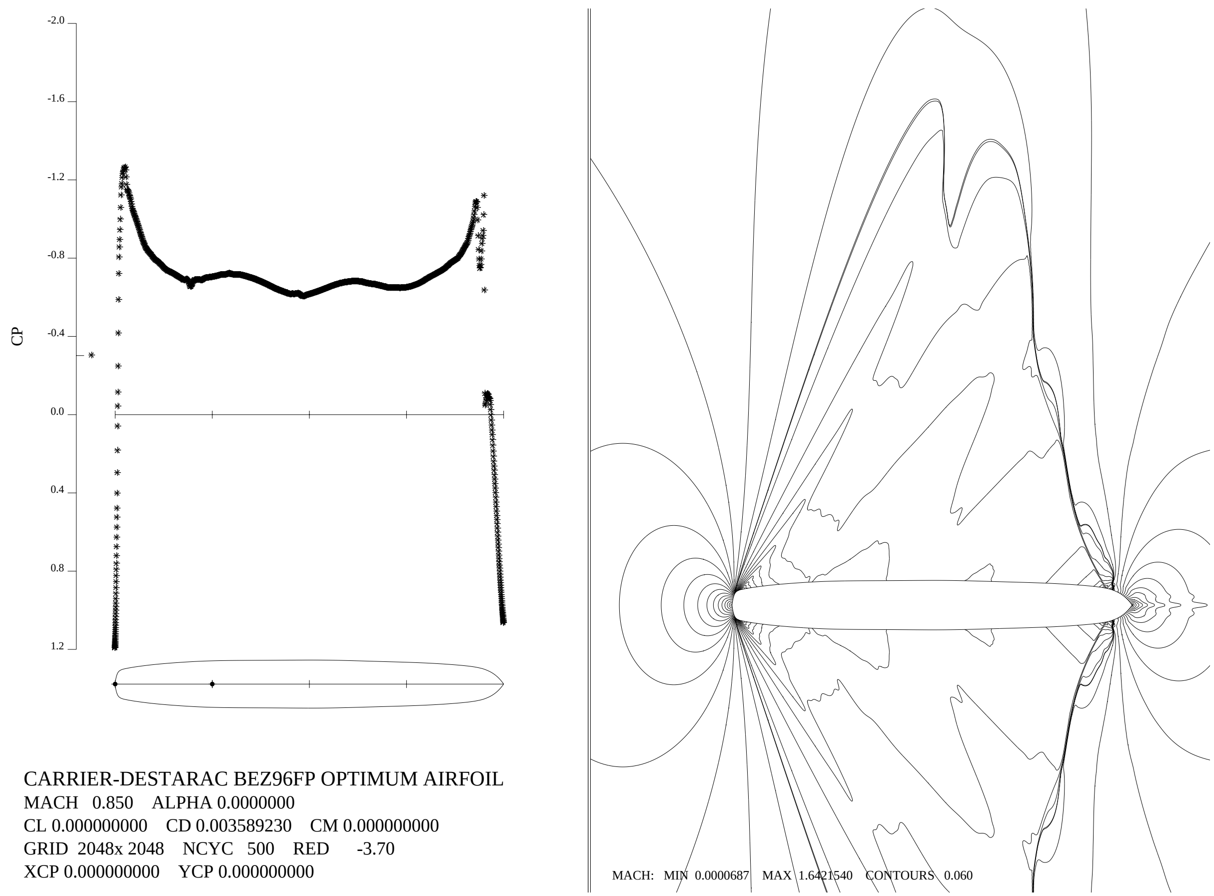
FLO82 Drag Assessment (C_d in counts).

Airfoil	N256	N512	N1024	N2048	∞	ΔC_d
NACA0012-ADO	470.19	470.09	471.13	471.23	471.27	-
SYNA	153.78	123.24	119.03	118.34	118.21	-353.06
SEEDB	111.04	98.98	97.68	96.85	95.42	-375.85
SYNB	109.25	86.87	84.68	84.50	84.48	-386.79
NADOT101	487.50	488.20	488.48	488.58	488.62	+17.35
SYNBT101	122.22	99.20	96.75	96.64	96.63	-374.64

Note: 1% Increase in Thickness Increases Drag:

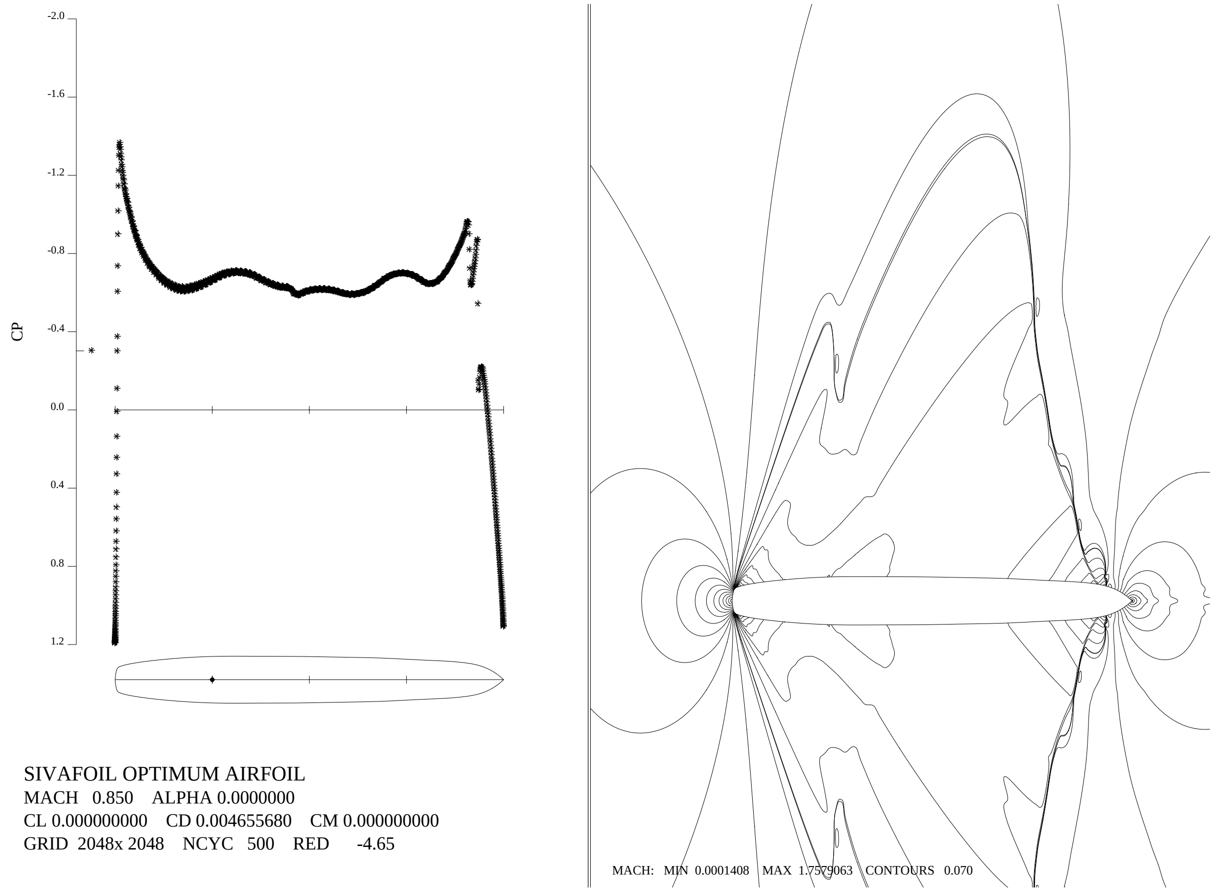
- NACA0012-ADO Airfoil by 3.68%
- SYNB Design by 14.38%.

CARRIER-DESTERAC AIRFOIL (G)



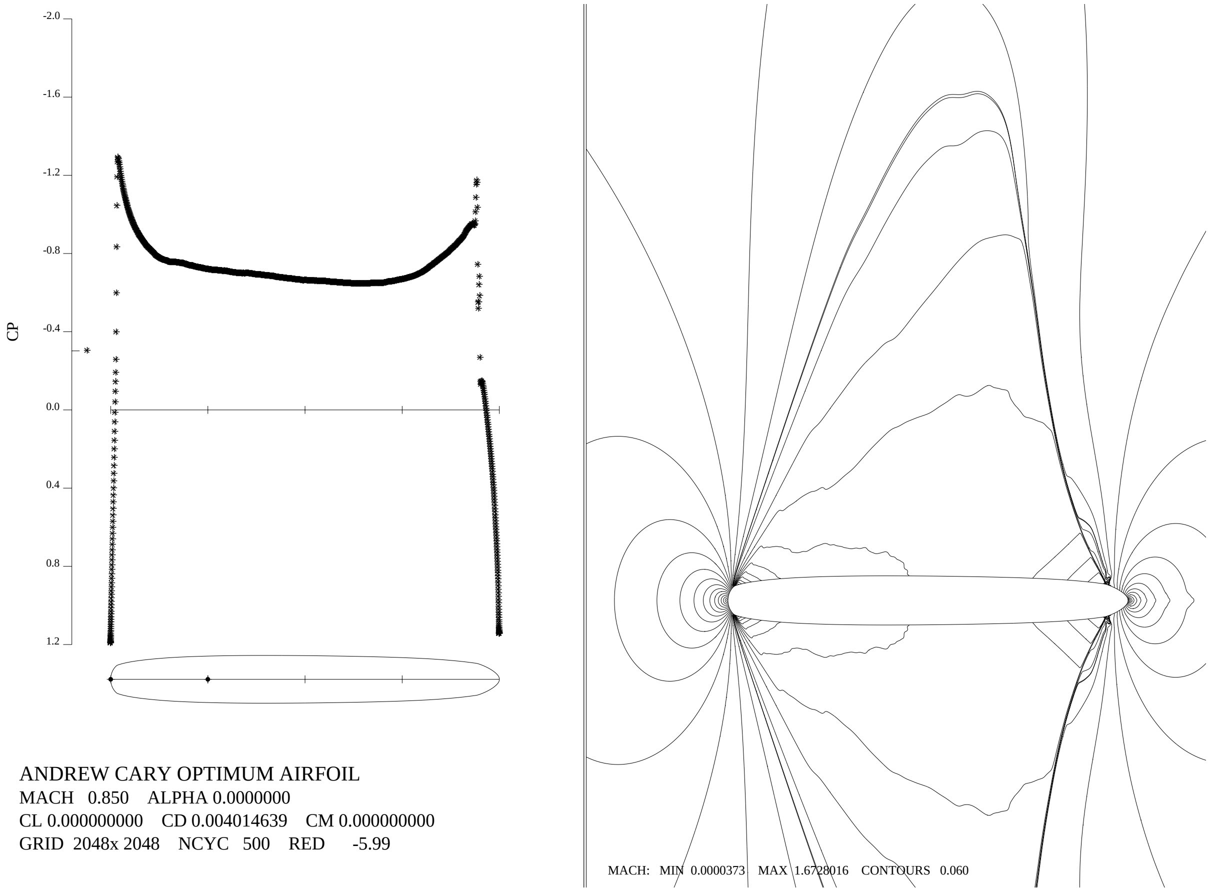
FLO82 Solution, Carrier-Desterac Airfoil (G), $M = 0.85$, $\alpha = 0^\circ$.

BISSON-NADARAJAH AIRFOIL (S)



FLO82 Solution, Bisson-Nadarajah Airfoil (S), $M = 0.85$, $\alpha = 0^\circ$.

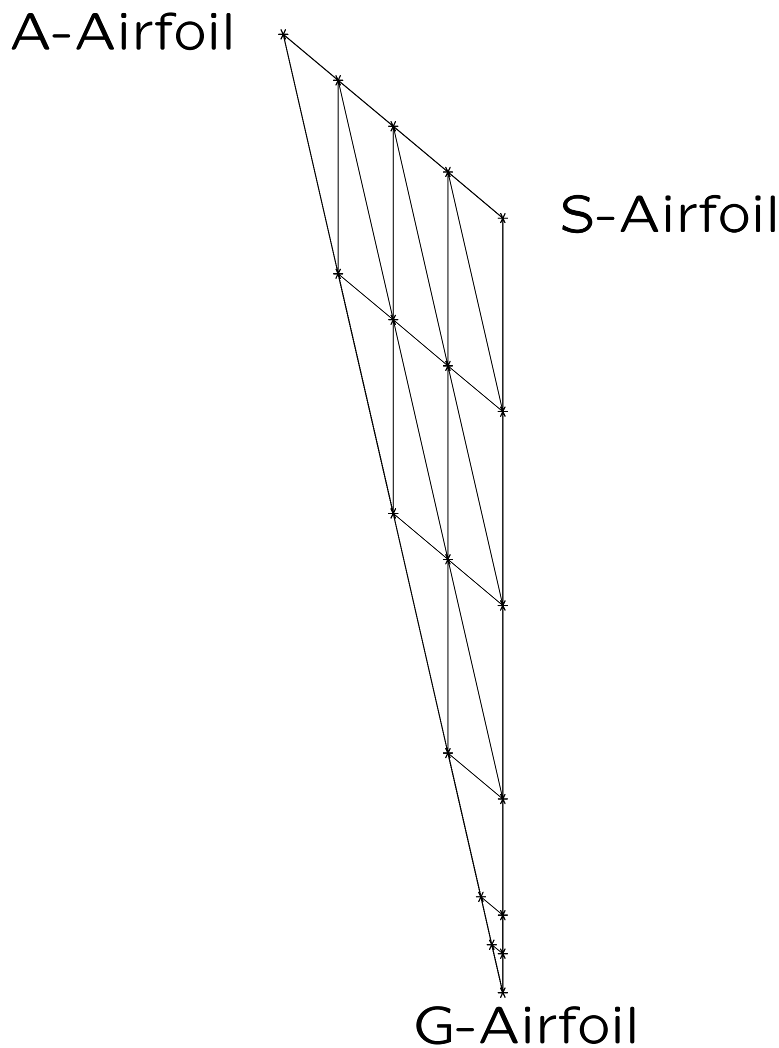
CARY AIRFOIL (A)



FLO82 Solution, Cary Airfoil (A), $M = 0.85$, $\alpha = 0^\circ$.

GSA STUDY

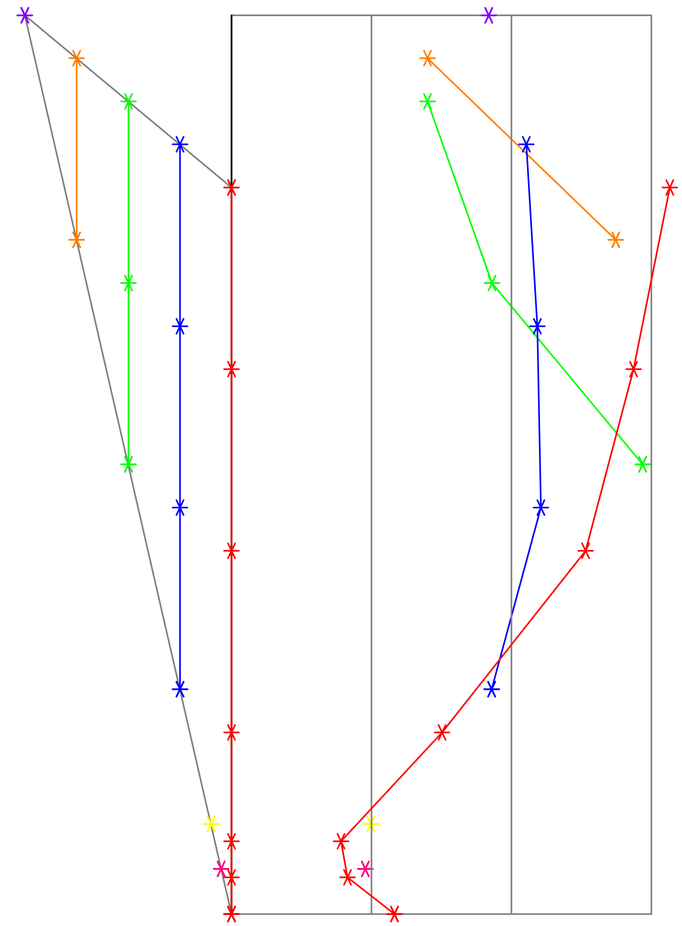
- GSA Sub-Space Spanned By 3 Airfoils
  - Gerald's, Siva's & Andrew's Optimimum Airfoils
- Geometric RMS Distances Between GSA Airfoils
  - GS: 0.0258890
  - GA: 0.0328760
  - SA: 0.0096097
- GSA Triangle Interior Angles
  - $\angle$G: 12.9886°
  - $\angle$S: 129.7516°
  - $\angle$A: 37.2598°



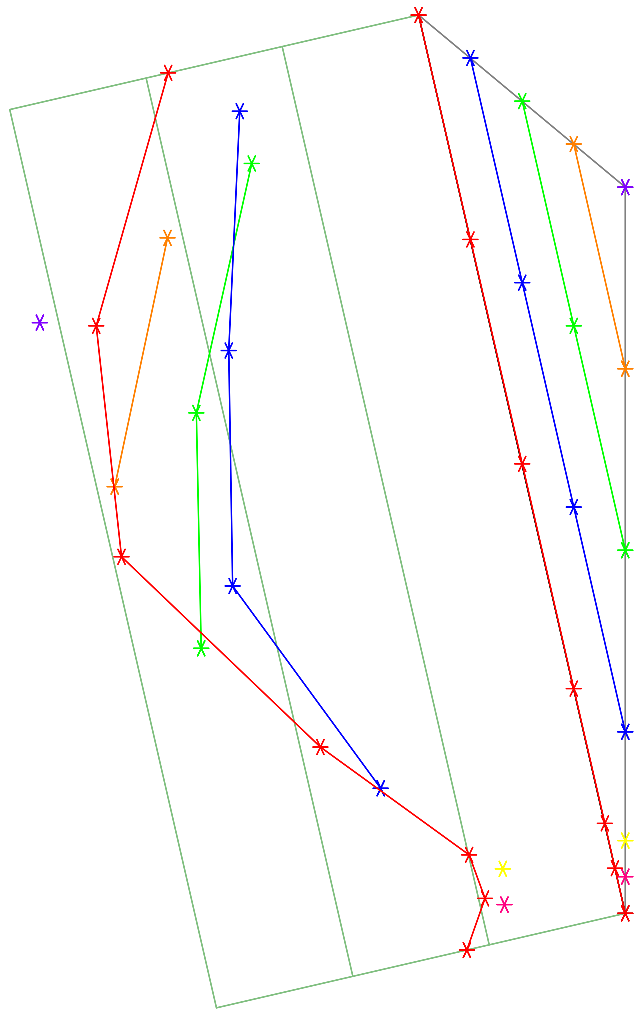
GSA Triangle & FLO82 Survey Locations.

GSA FLO82 Results, $M = 0.85$, $\alpha = 0°$.

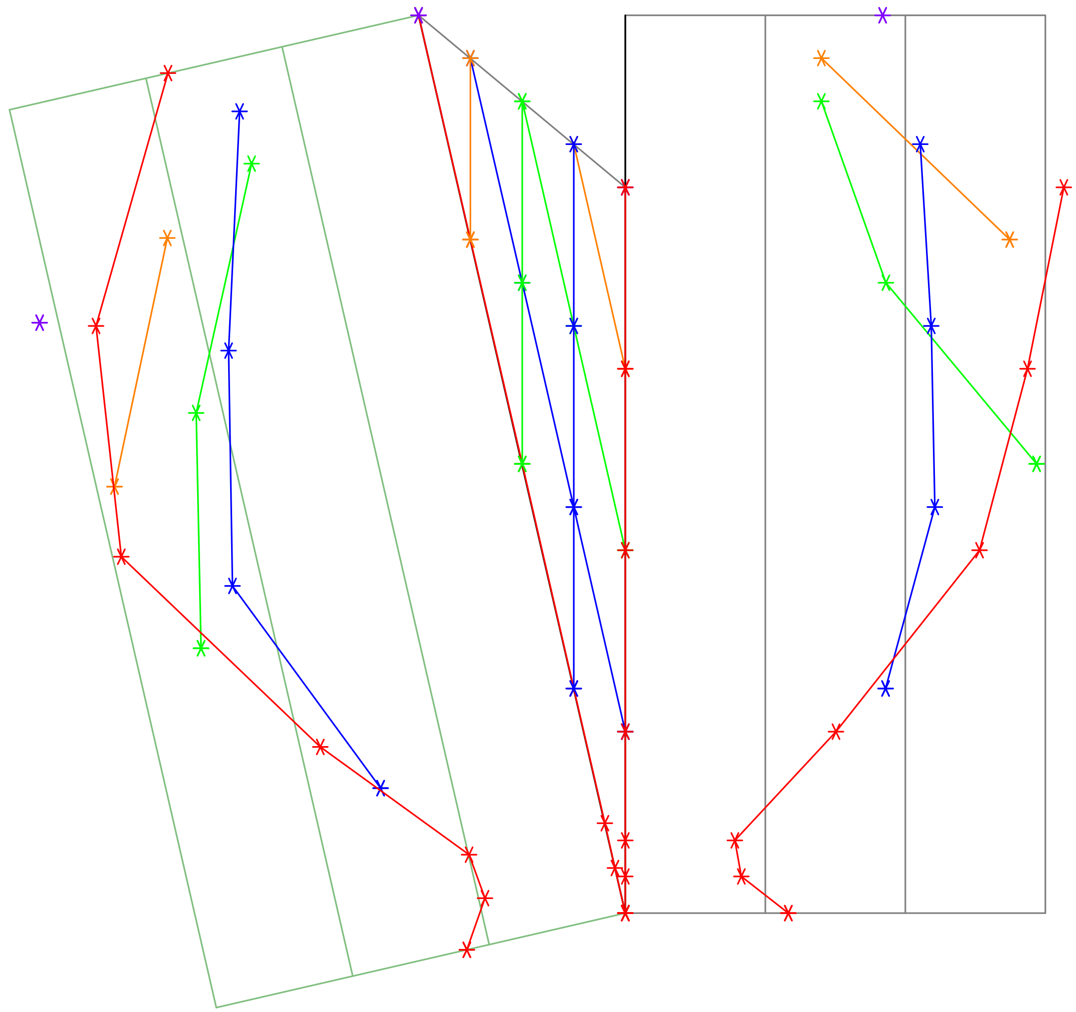
	G_{WT}	S_{WT}	A_{WT}	$C_{d\infty}$	X_{TRI}	Y_{TRI}
A	0.00	0.00	1.00	39.189621	0.03203	0.00739
	0.00	0.25	0.75	$\sim$37.0	0.03050	0.00554
	0.00	0.50	0.50	$\sim$37.0	0.02896	0.00369
	0.00	0.75	0.25	40.531373	0.02743	0.00185
S	0.00	1.00	0.00	45.656136	0.02589	0.00000
	0.25	0.00	0.75	43.724493	0.02403	0.00554
	0.25	0.25	0.50	39.300977	0.02249	0.00369
	0.25	0.50	0.25	40.930957	0.02095	0.00185
	0.25	0.75	0.00	44.367019	0.01942	0.00000
	0.50	0.00	0.50	44.688405	0.01602	0.00369
	0.50	0.25	0.25	41.051787	0.01448	0.00185
	0.50	0.50	0.00	42.651239	0.01294	0.00000
	0.75	0.00	0.25	39.288289	0.00801	0.00185
	0.75	0.25	0.00	37.520335	0.00647	0.00000
	0.90	0.00	0.10	34.978658	0.00320	0.00074
*	0.90	0.10	0.00	33.911086	0.00259	0.00000
	0.95	0.00	0.05	34.777423	0.00160	0.00037
	0.95	0.05	0.00	34.142484	0.00130	0.00000
G	1.00	0.00	0.00	35.818186	0.00000	0.00000



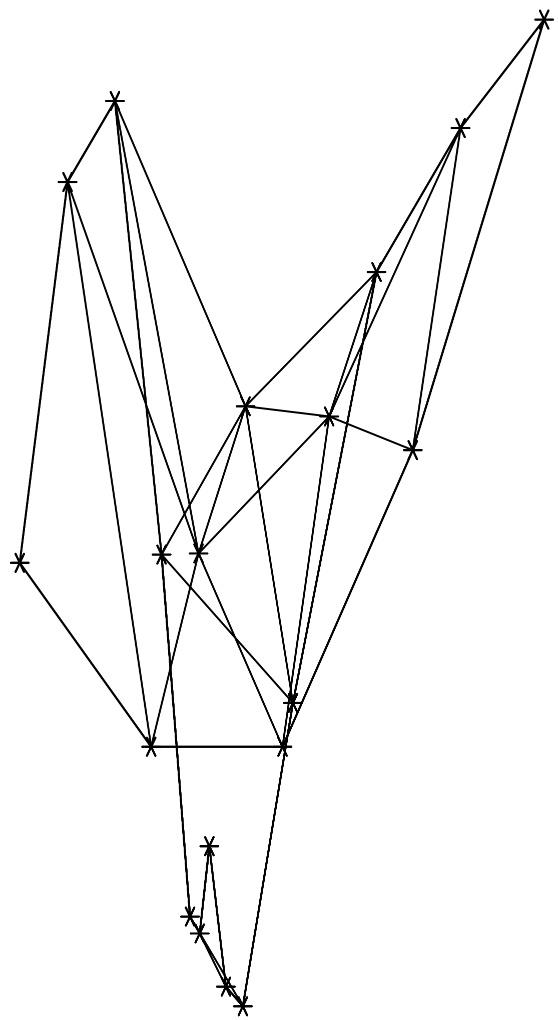
GSA FLO82 Solutions, Iso-A Contours, $M = 0.85$, $\alpha = 0^\circ$.



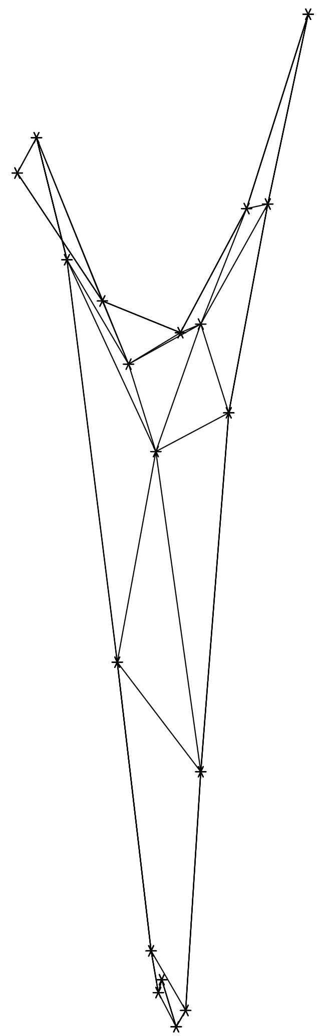
GSA FLO82 Solutions, Iso-S Contours, $M = 0.85$, $\alpha = 0^{\circ}$.



FLO82 Solutions, $M = 0.85$, $\alpha = 0^\circ$.



FLO82 Solutions, Level View of Valley, $M = 0.85$, $\alpha = 0^\circ$.



FLO82 Solutions, Elevated View of Valley, $M = 0.85$, $\alpha = 0^\circ$.

ADO-DG CASE-1 SUMMARY

- Optimizations Yield Pathological Designs
  - Small Geometric Changes $\Rightarrow$ Large Solution Deltas
- Non-Unique Solutions at Design Point
  - Lifting ($\pm$) Solutions for Symmetric Airfoils at $\alpha = 0°$
- Convergence Issues with Flow & Adjoint Solvers
  - Forcing Symmetric Solutions Helps
- Shock-Free Design Still Not Found
  - The GSA Airfoils Are Close

ADO-CRM-WING MODEL PROBLEM

- **ADO-DG Case 4 General Statement**
  - Minimize Drag at Fixed Lifting Condition(s)
  - $Re = 5.0 \times 10^6$
  - $C_M \geq -0.17$ at $C_L = 0.50$ & $M = 0.85$
  - Maintain or Exceed Internal Volume of Baseline Wing

- **Case 4.1 Single-Point**
  - $M = 0.85$, $C_L = 0.50$

- **Case 4.2 Triple-Point C_L Sweep**
  - $M = 0.85$, $C_L = [0.45 , 0.50 , 0.55]$, $WT = [1 , 2 , 1]$

- **Case 4.3 Triple-Point Mach Sweep**
  - $M = [0.84 , 0.85 , 0.86]$, $C_L = 0.50$, $WT = [1 , 2 , 1]$

ADO-CRM-WING MODEL PROBLEM

- **ADO-CRM-Wing Reference Quantities**
  - $Sref/2 = 3.407014$, $Cref = 1.0$, $b/2 = 3.75820$
  - $Xref = 1.2077$, $Yref = 0.0$, $Zref = 0.007669$

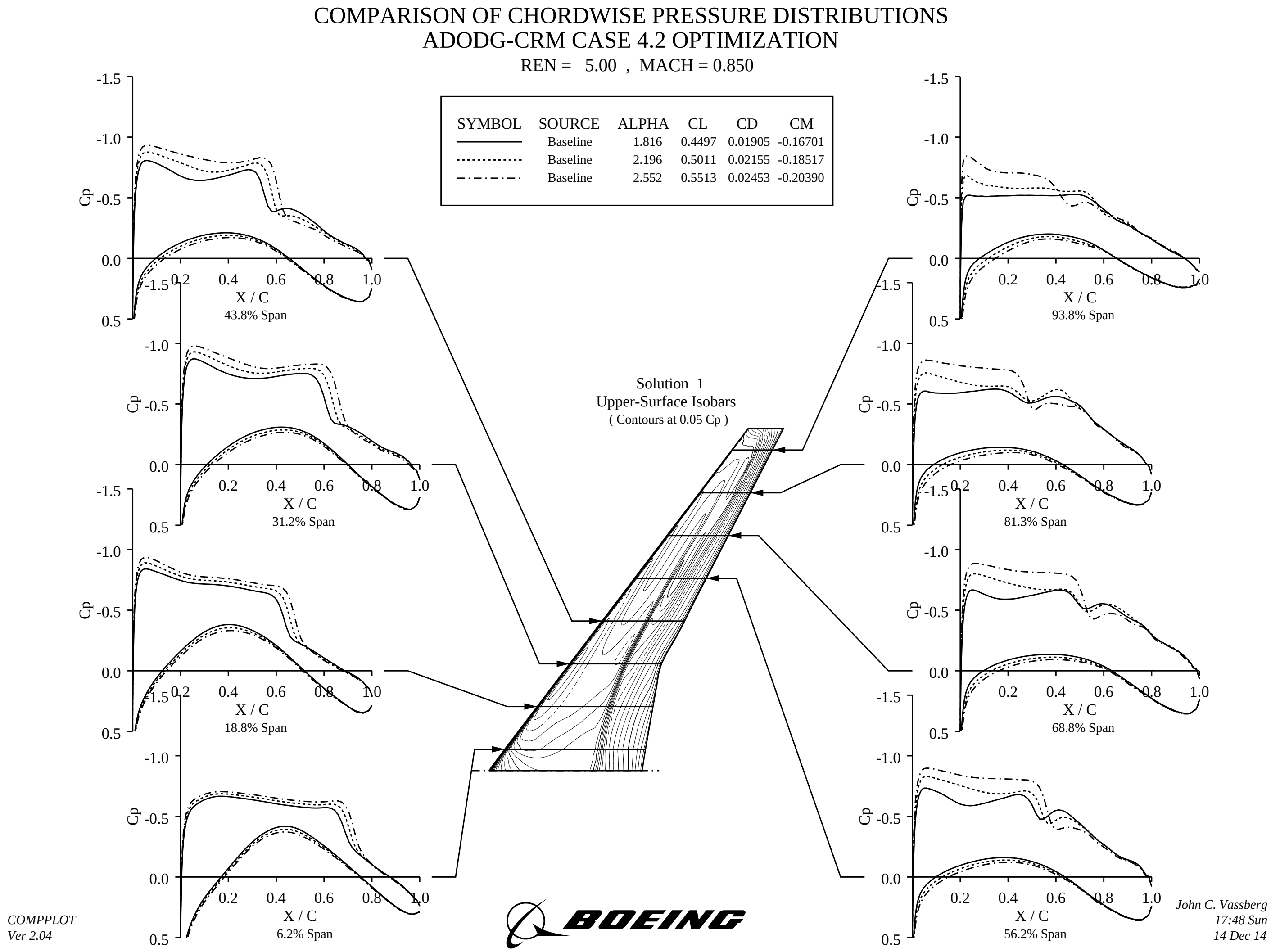
- **SYN107 Optimizations**
  - Constrained Lifting Condition(s)
    - Active Adjustment of α During Convergence
    - Includes dC_D/dC_L Terms in Gradient Formulation
  - Added Pitching-Moment Penalty
    - Sufficient to Achieve C_M Constraint
  - Constrained Airfoil Area Distribution
    - Over-Constraint on Wing Volume
    - More Representative of Practice
  - Omits Eddy Viscousity Derivatives

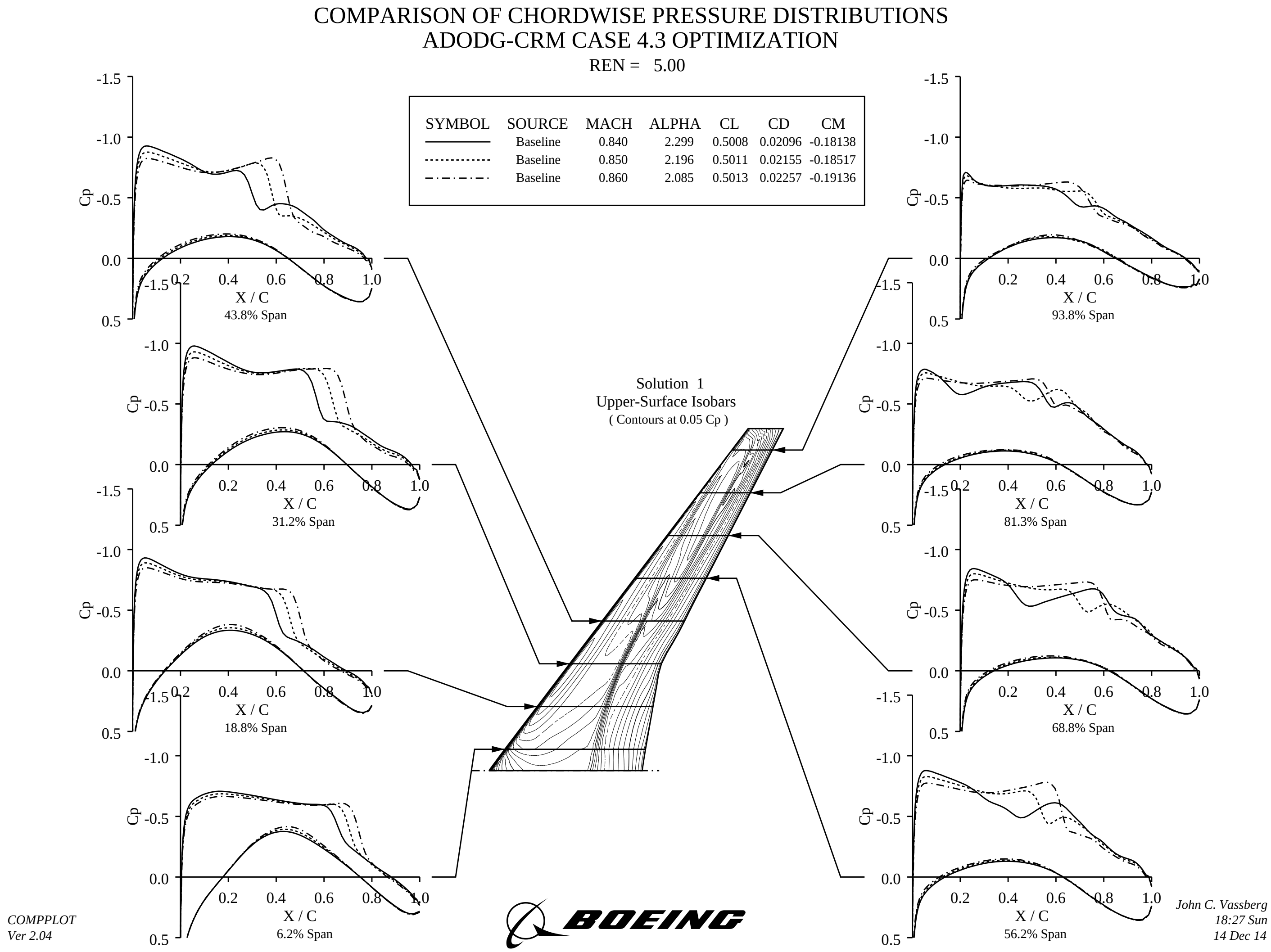
SYN107 OPTIMIZATIONS

- SYN107 Optimization Require $\sim$ 2.3 Hours
  - Per Design Point, Per 100 Design Cycles
  - 2046 Design Variables; (2x31x33) Cubic B-Spline CPs
  - Initial & Every 10^{th} Design Cycle
    - 240 Iters for Analysis
    - 160 Iters for dC_D/dC_L, dC_M/dC_L & $dC_L/d\alpha$
    - 240 Iters for Adjoint
  - Otherwise
    - 20 Iters for Analysis & Adjoint
- SYN107 Analyses Require $\sim$ 4.2 Minutes
  - Per Flow Condition (200 Iters)
  - Run as Alpha or Mach Sweeps
  - Grid: (256x64x48) C-Mesh
- Parallel Execution on 4 Cores Deskside Computer
  - Intel i7-970 CPU at 3.2 GHz; 2011-Q1
- Current Computers $\sim$ 5X Faster

ADO-DG CASE 4: BASELINE WING

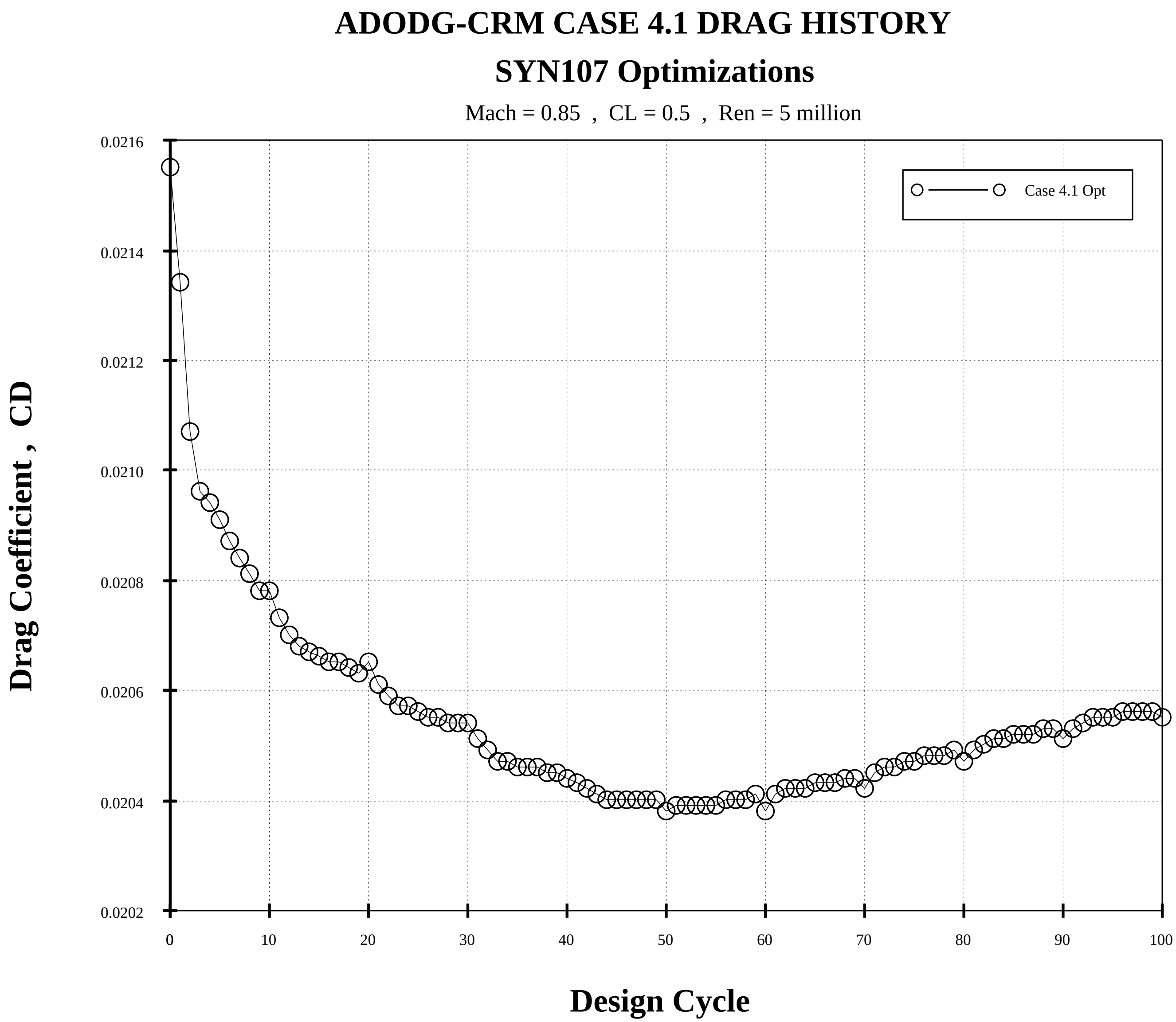
- **Design Pt.:** $M = 0.85$, $C_L = 0.50$, $Re = 5.0 \times 10^6$
- **Polar:** $C_L = [0.45\ ,\ 0.50\ ,\ 0.55]$
- **DragRise:** $M = [0.84\ ,\ 0.85\ ,\ 0.86]$
- **Comparisons**
  - Overlaid Pressure Distributions

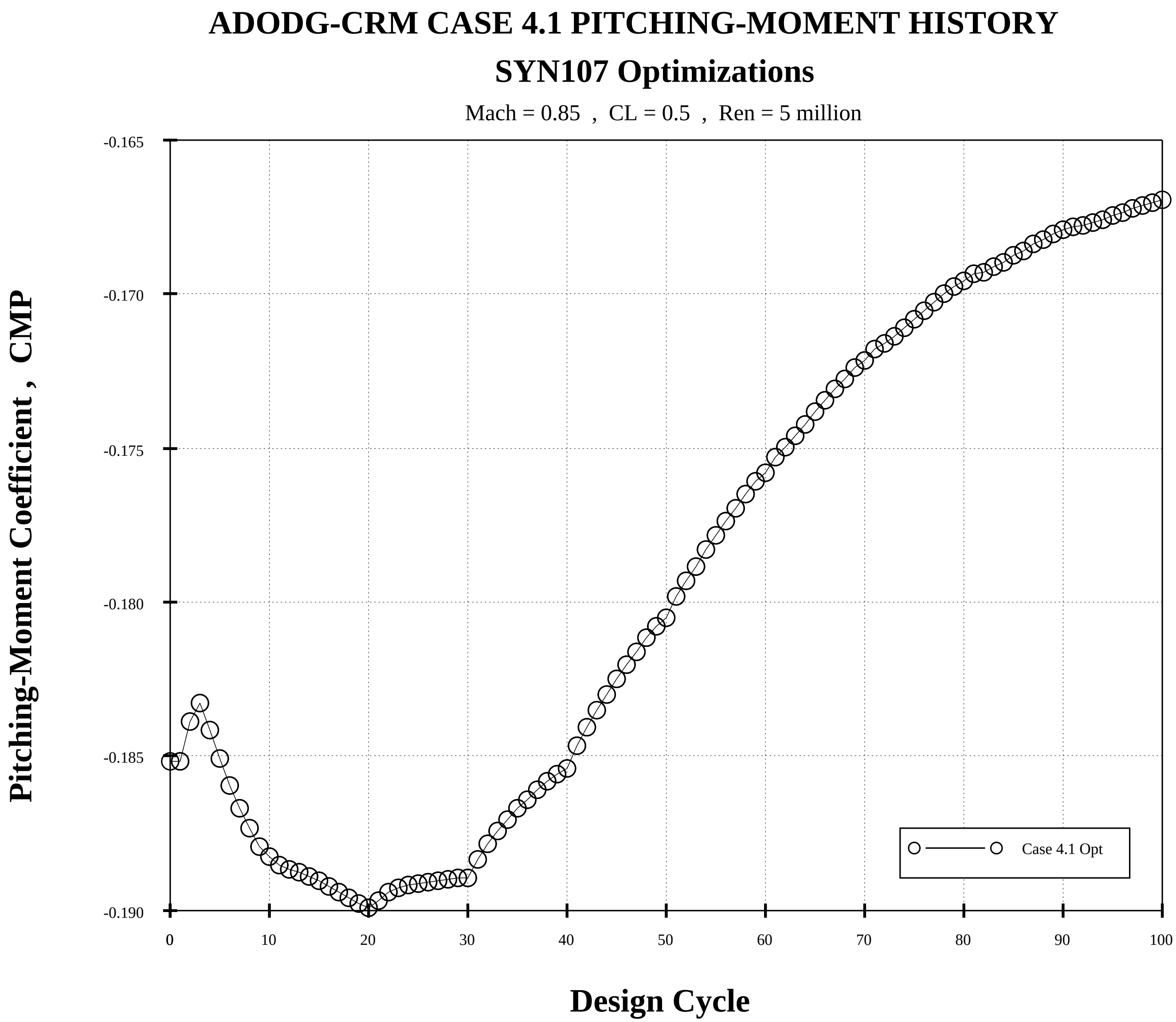

COMPARISON OF CHORDWISE PRESSURE DISTRIBUTIONS
ADODG-CRM CASE 4.2 OPTIMIZATION
REN = 5.00 , MACH = 0.850
SYMBOL SOURCE ALPHA CL CD CM
Baseline 1.816 0.4497 0.01905 -0.16701
Baseline 2.196 0.5011 0.02155 -0.18517
Baseline 2.552 0.5513 0.02453 -0.20390
Solution 1
Upper-Surface Isobars
(Contours at 0.05 Cp)
Cp
X / C
43.8% Span
31.2% Span
18.8% Span
6.2% Span
93.8% Span
81.3% Span
68.8% Span
56.2% Span
COMPPLOT
Ver 2.04
BOEING
John C. Vassberg
17:48 Sun
14 Dec 14

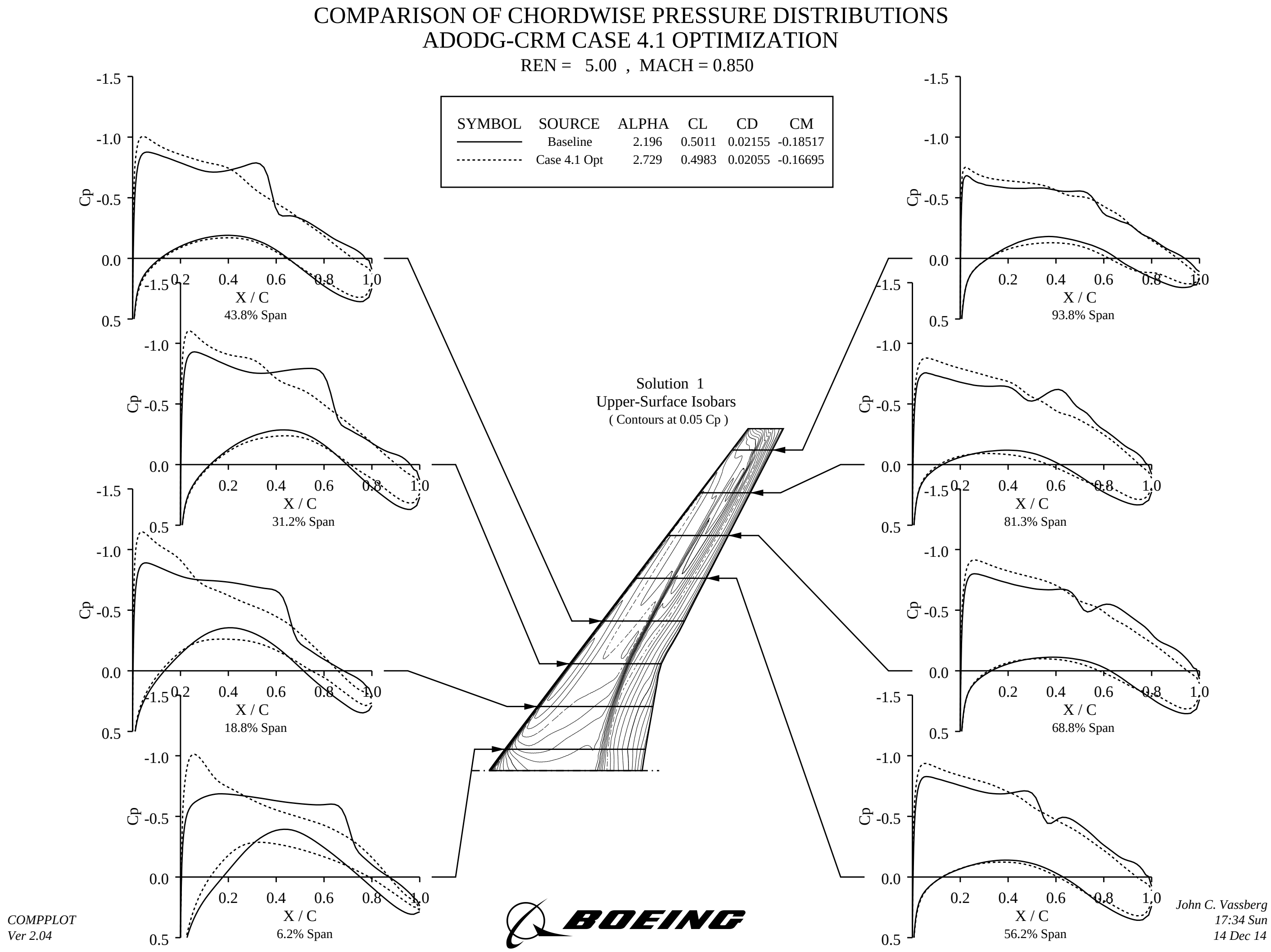

COMPARISON OF CHORDWISE PRESSURE DISTRIBUTIONS
ADODG-CRM CASE 4.3 OPTIMIZATION
REN = 5.00
SYMBOL SOURCE MACH ALPHA CL CD CM
Baseline 0.840 2.299 0.5008 0.02096 -0.18138
Baseline 0.850 2.196 0.5011 0.02155 -0.18517
Baseline 0.860 2.085 0.5013 0.02257 -0.19136
Solution 1
Upper-Surface Isobars
(Contours at 0.05 Cp)
Cp
X / C
43.8% Span
31.2% Span
18.8% Span
6.2% Span
93.8% Span
81.3% Span
68.8% Span
56.2% Span
COMPPLOT
Ver 2.04
John C. Vassberg
18:27 Sun
14 Dec 14

ADO-DG CASE 4.1: Single-Point

- $M = 0.85$, $C_L = 0.50$, $Re = 5.0 \times 10^6$
- **Design Cycle Histories**
  - Drag & Pitching Moment
- **Baseline .vs. Final Design Comparisons**
  - Overlaid Pressure Distributions
  - Side-by-Side Upper-Surface Isobars
  - Overlaid Spanload Distributions


ADODG-CRM CASE 4.1 DRAG HISTORY
SYN107 Optimizations
Mach = 0.85 , CL = 0.5 , Ren = 5 million
Case 4.1 Opt
Drag Coefficient , CD
Design Cycle
0.0216
0.0214
0.0212
0.0210
0.0208
0.0206
0.0204
0.0202
0
10
20
30
40
50
60
70
80
90
100


ADODG-CRM CASE 4.1 PITCHING-MOMENT HISTORY
SYN107 Optimizations
Mach = 0.85 , CL = 0.5 , Ren = 5 million
-0.165
-0.170
-0.175
-0.180
-0.185
-0.190
0
10
20
30
40
50
60
70
80
90
100
Pitching-Moment Coefficient , CMP
Design Cycle
Case 4.1 Opt


COMPARISON OF CHORDWISE PRESSURE DISTRIBUTIONS
ADODG-CRM CASE 4.1 OPTIMIZATION
REN = 5.00 , MACH = 0.850
SYMBOL SOURCE ALPHA CL CD CM
Baseline 2.196 0.5011 0.02155 -0.18517
Case 4.1 Opt 2.729 0.4983 0.02055 -0.16695
Solution 1
Upper-Surface Isobars
(Contours at 0.05 Cp)
Cp
X / C
43.8% Span
31.2% Span
18.8% Span
6.2% Span
93.8% Span
81.3% Span
68.8% Span
56.2% Span
COMPPLOT
Ver 2.04
BOEING
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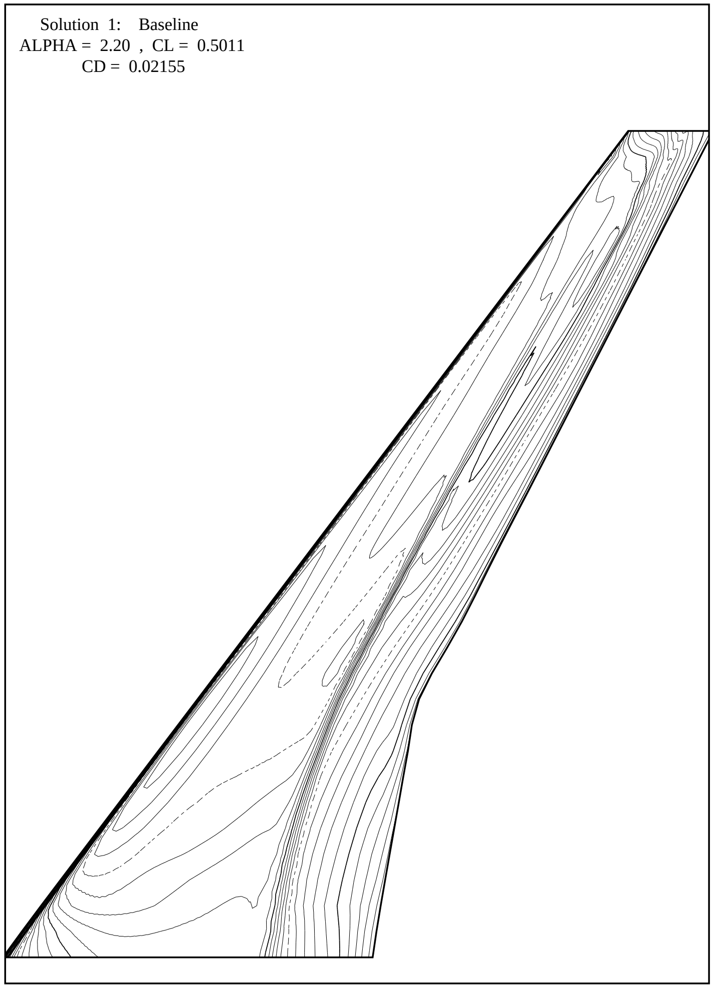
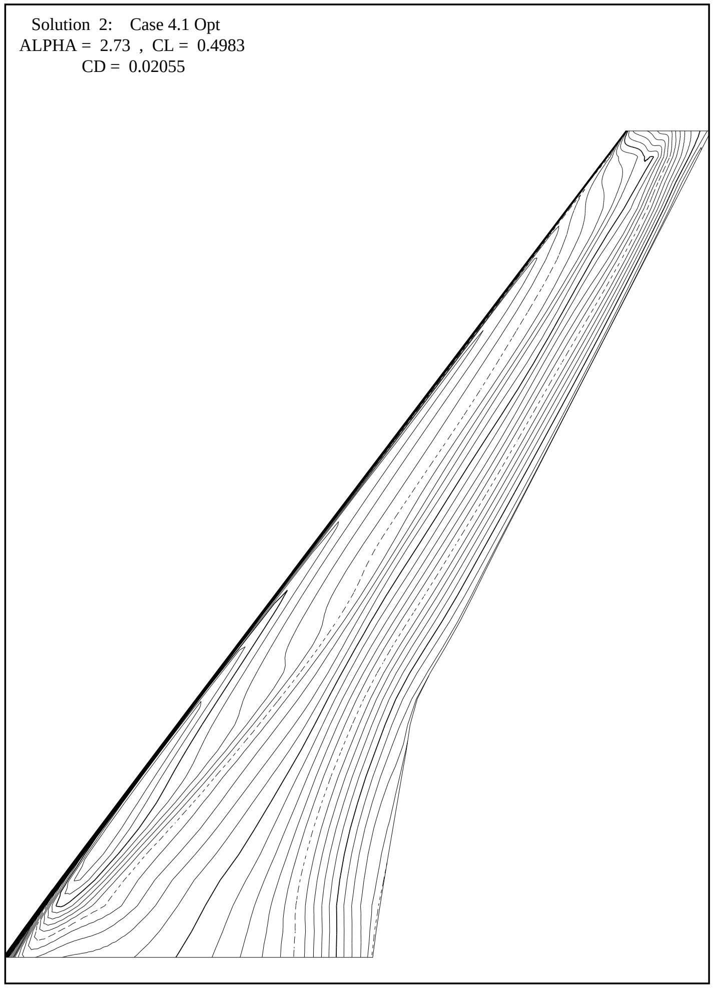
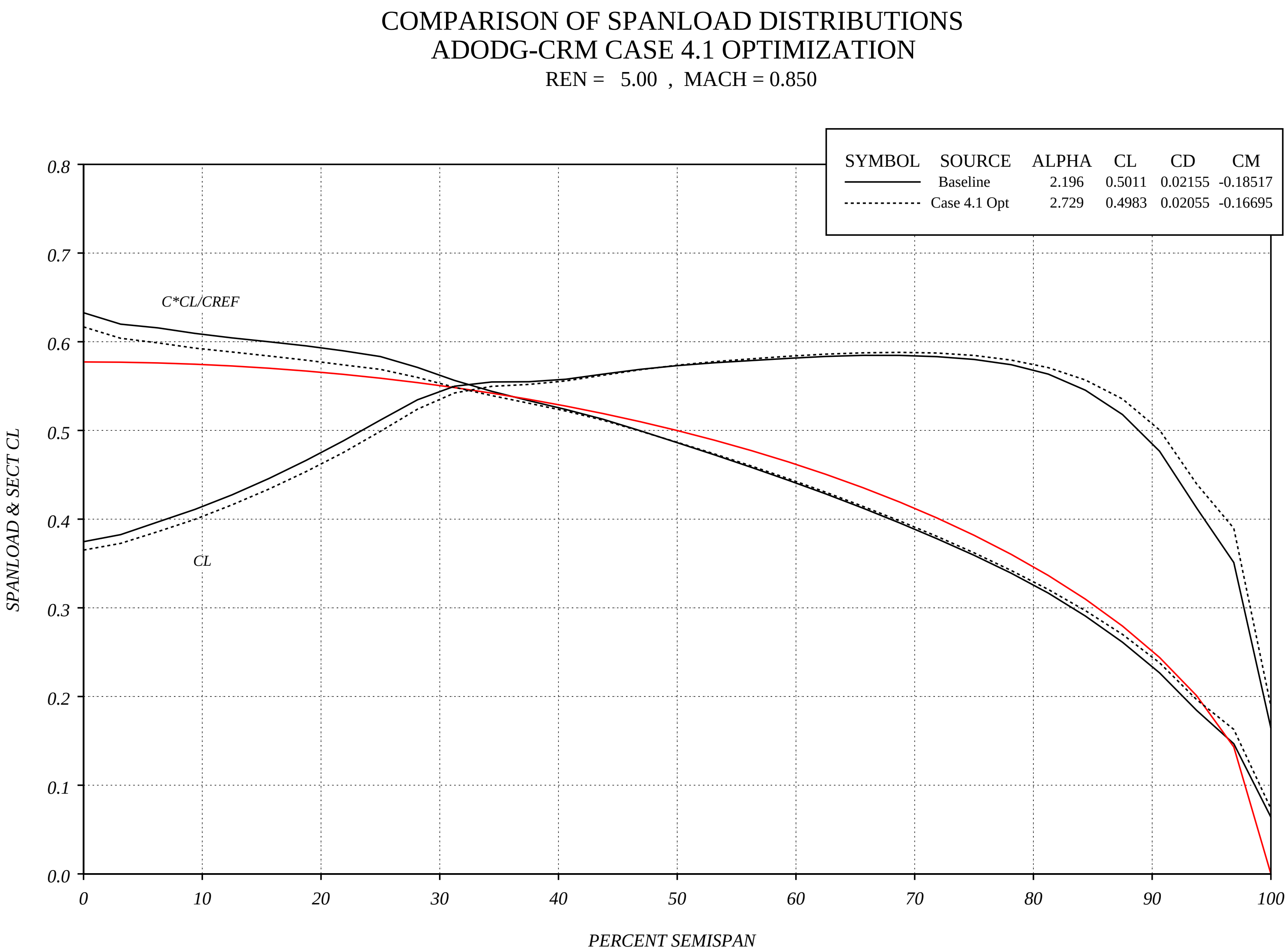
17:34 Sun
14 Dec 14

COMPARISON OF UPPER SURFACE CONTOURS
ADODG-CRM CASE 4.1 OPTIMIZATION
REN = 5.00 , MACH = 0.850
(Contours at 0.05 Cp)

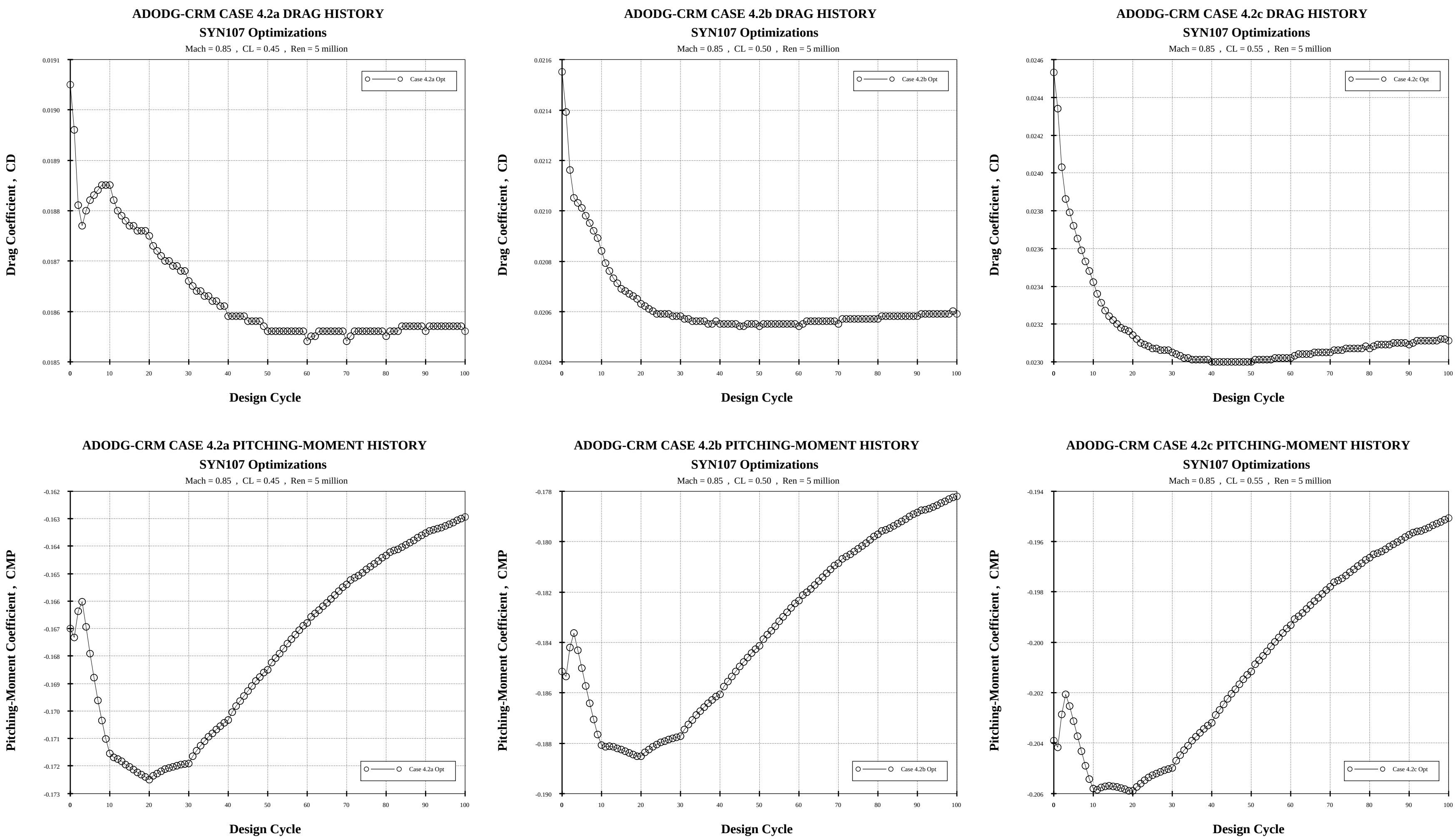




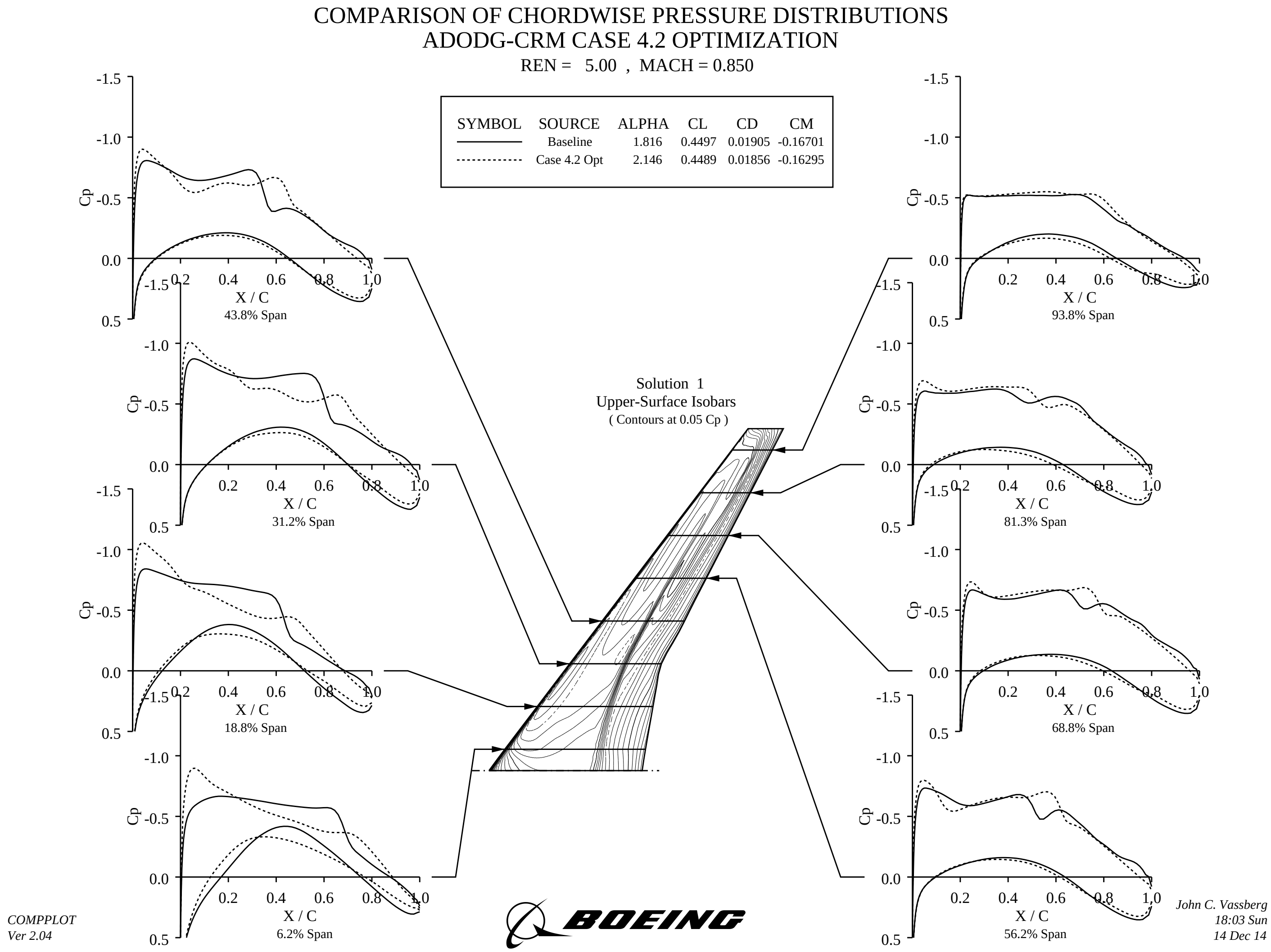

COMPARISON OF SPANLOAD DISTRIBUTIONS
ADODG-CRM CASE 4.1 OPTIMIZATION
REN = 5.00 , MACH = 0.850
SYMBOL SOURCE ALPHA CL CD CM
Baseline 2.196 0.5011 0.02155 -0.18517
Case 4.1 Opt 2.729 0.4983 0.02055 -0.16695
C*CL/CREF
CL
SPANLOAD & SECT CL
PERCENT SEMISPAN
0.0
0.1
0.2
0.3
0.4
0.5
0.6
0.7
0.8
0
10
20
30
40
50
60
70
80
90
100

ADO-DG CASE 4.2: Triple-Point C_L

- $M = 0.85$, $C_L = [0.45\ ,\ 0.50\ ,\ 0.55]$, $Re = 5.0 \times 10^6$
- **Design Cycle Histories, All 3 Conditions**
  - Drag & Pitching Moment
- **Baseline .vs. Final Design Comparisons**
  - Overlaid Pressure Distributions
  - Overlaid Spanload Distributions



ADO-DG Case 4.2 SYN107 Design-Cycle Histories.


COMPARISON OF CHORDWISE PRESSURE DISTRIBUTIONS
ADODG-CRM CASE 4.2 OPTIMIZATION
REN = 5.00 , MACH = 0.850
SYMBOL SOURCE ALPHA CL CD CM
Baseline 1.816 0.4497 0.01905 -0.16701
Case 4.2 Opt 2.146 0.4489 0.01856 -0.16295
Solution 1
Upper-Surface Isobars
(Contours at 0.05 Cp)
Cp
X / C
43.8% Span
31.2% Span
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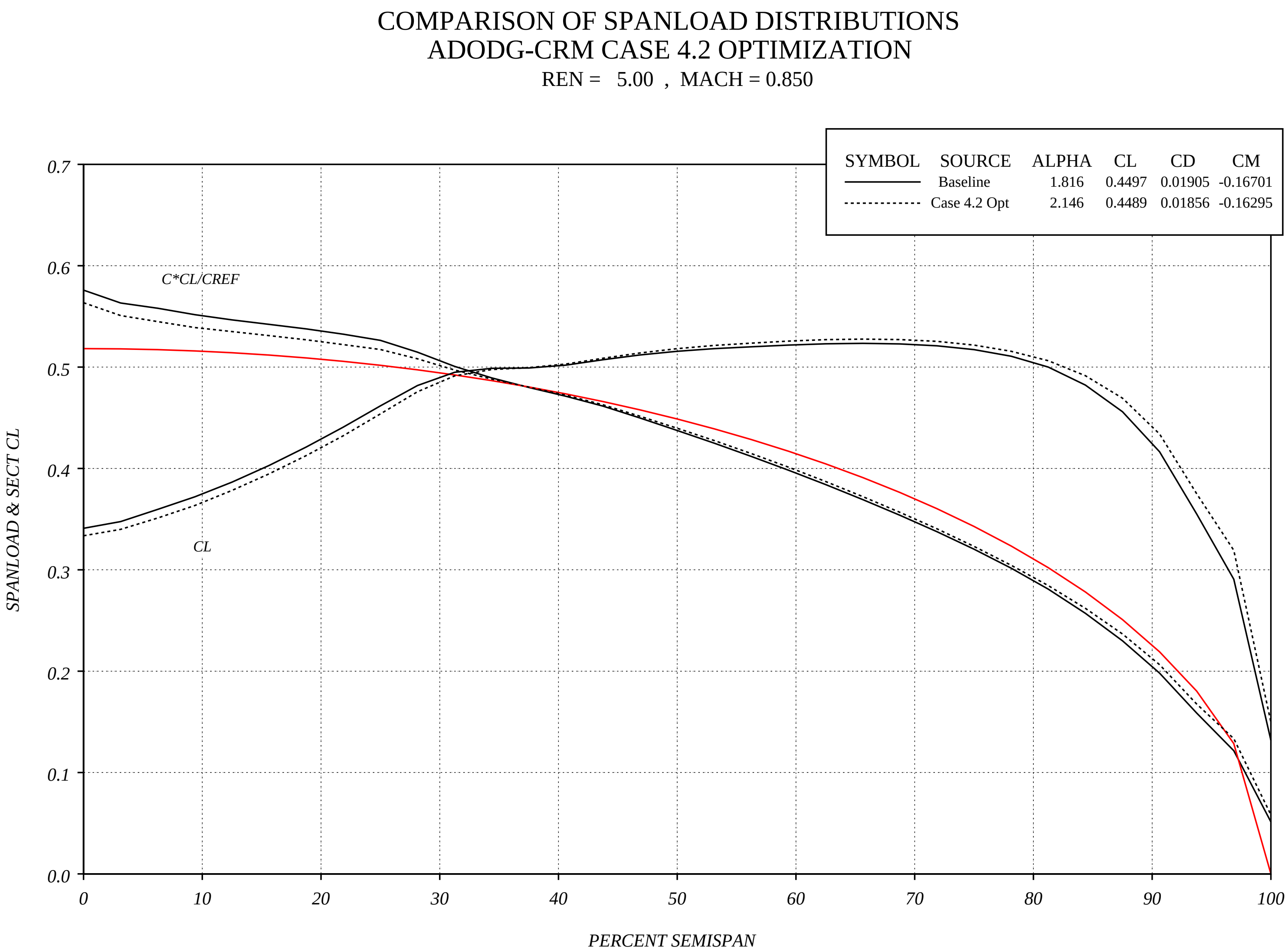
COMPARISON OF SPANLOAD DISTRIBUTIONS
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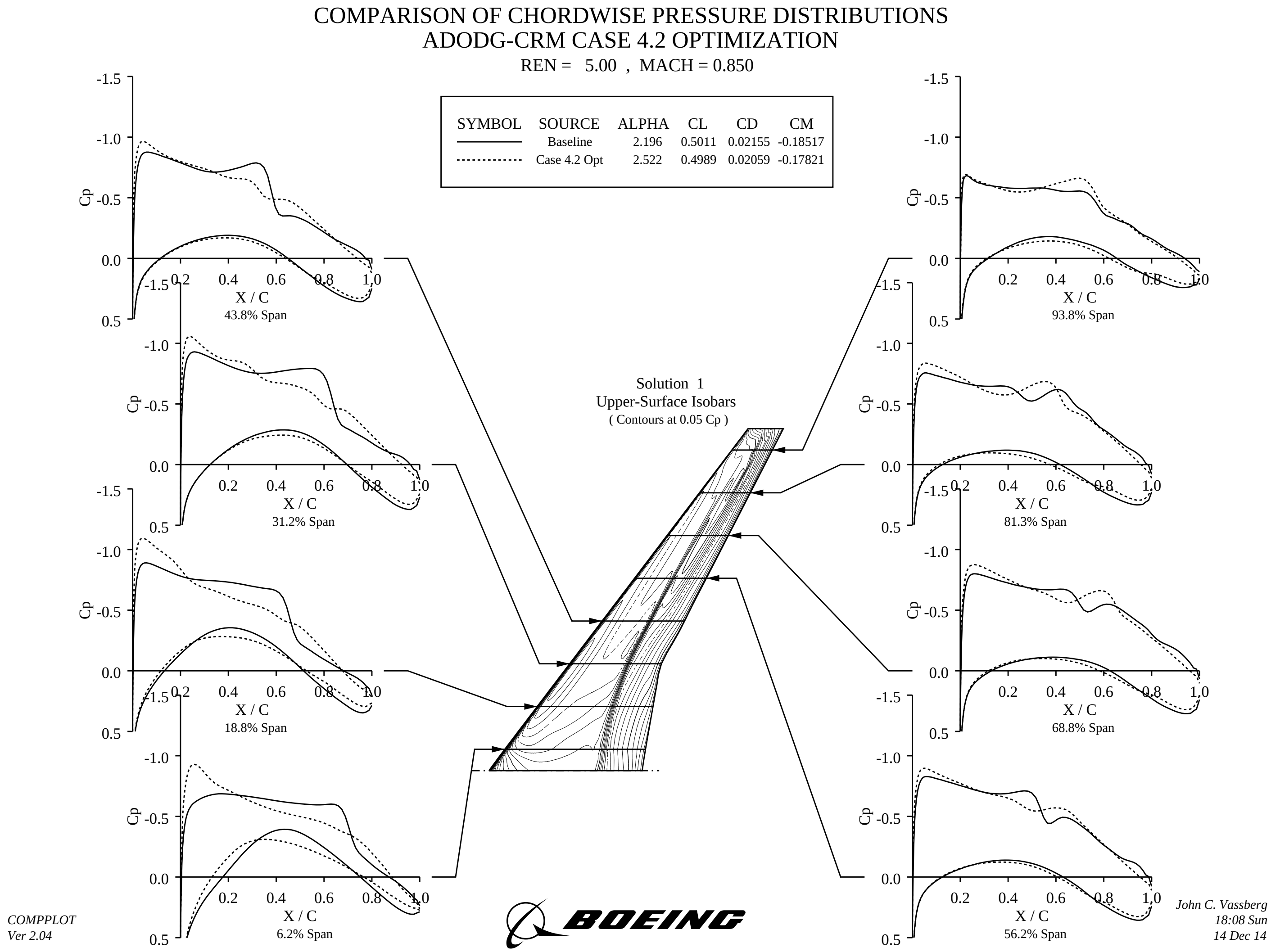
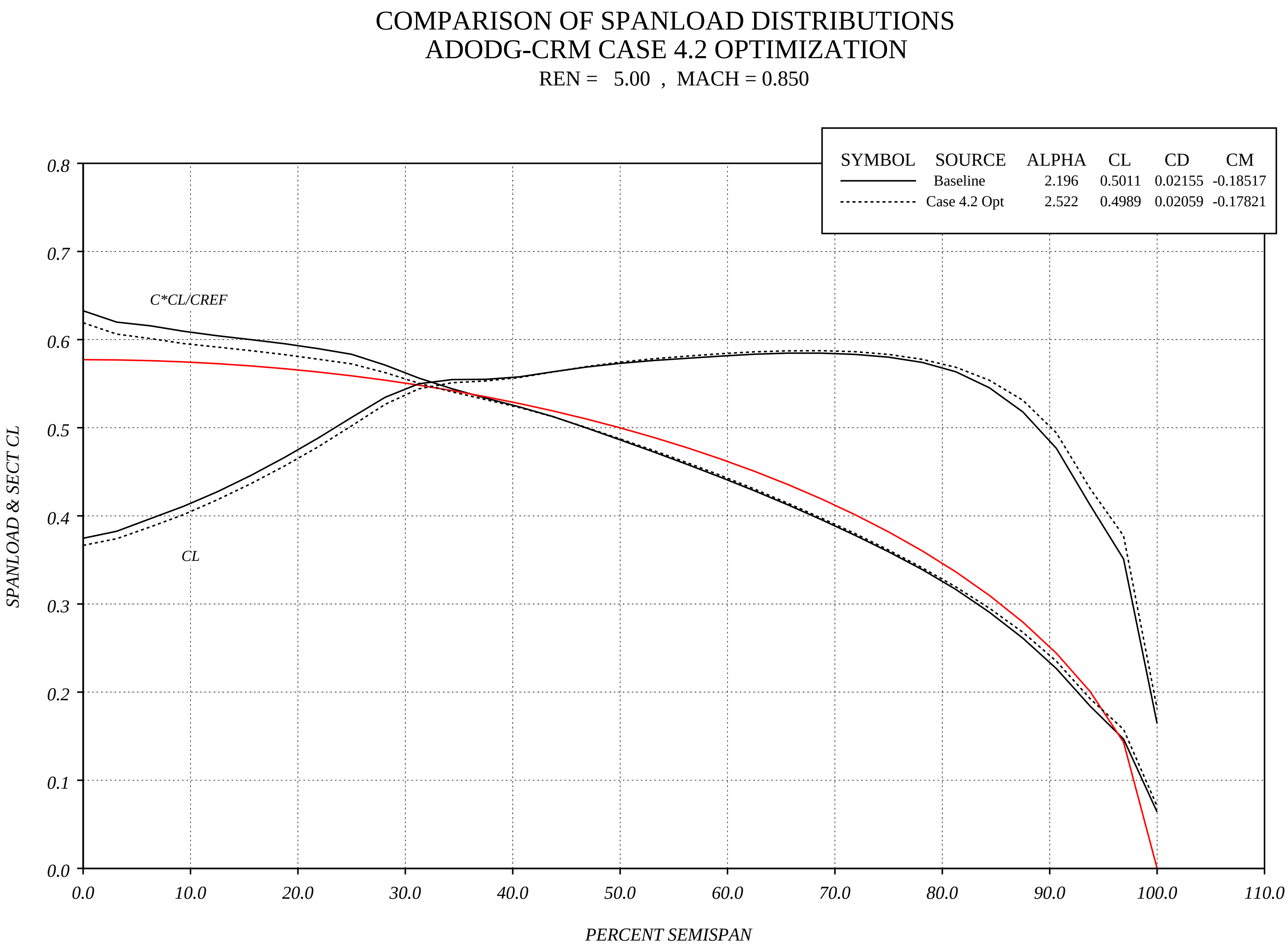
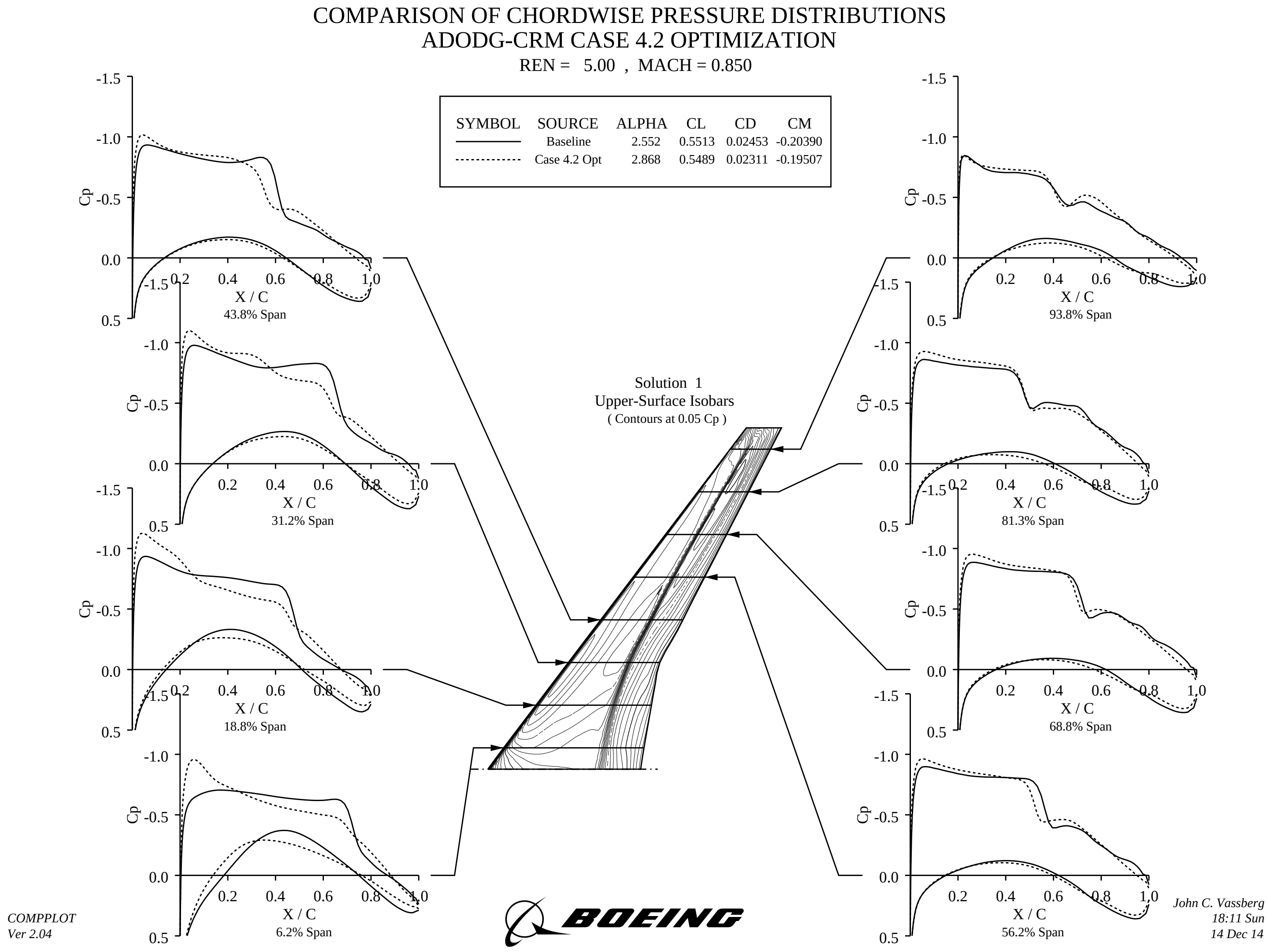
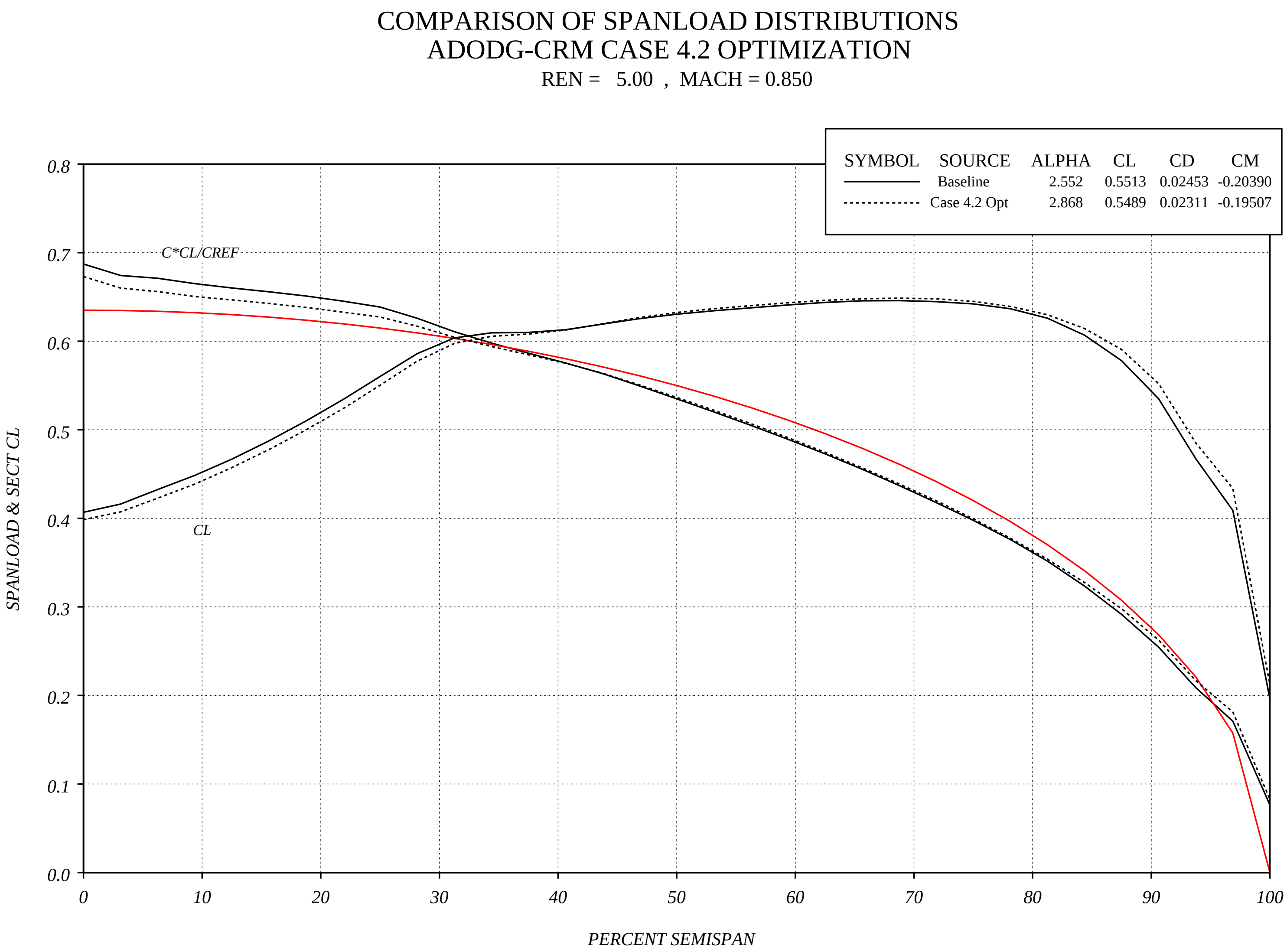
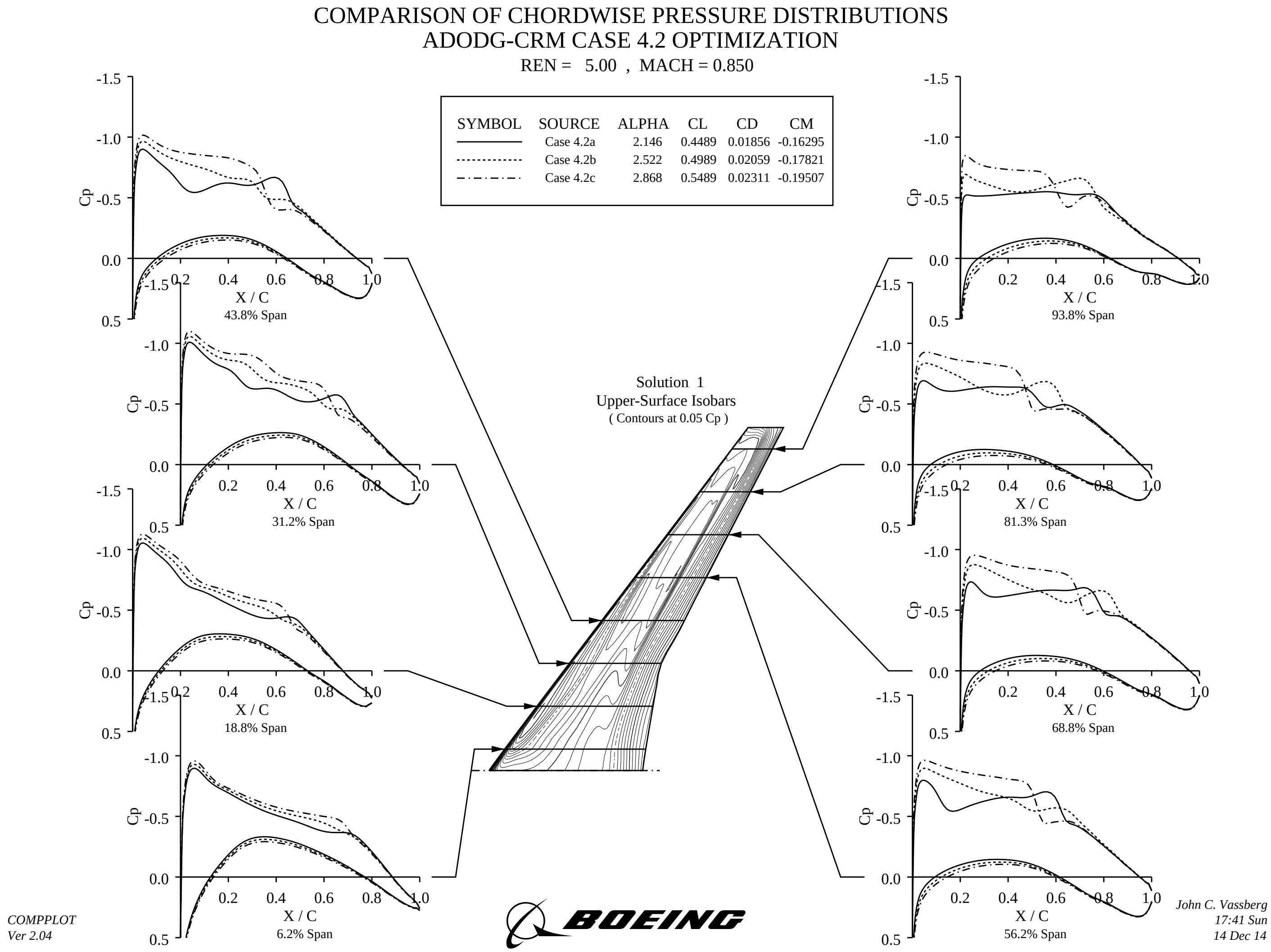
REN = 5.00 , MACH = 0.850
SYMBOL SOURCE ALPHA CL CD CM
Baseline 1.816 0.4497 0.01905 -0.16701
Case 4.2 Opt 2.146 0.4489 0.01856 -0.16295
C*CL/CREF
CL
SPANLOAD & SECT CL
PERCENT SEMISPAN
0.0
0.1
0.2
0.3
0.4
0.5
0.6
0.7
0
10
20
30
40
50
60
70
80
90
100


COMPARISON OF CHORDWISE PRESSURE DISTRIBUTIONS
ADODG-CRM CASE 4.2 OPTIMIZATION
REN = 5.00 , MACH = 0.850
SYMBOL SOURCE ALPHA CL CD CM
Baseline 2.196 0.5011 0.02155 -0.18517
Case 4.2 Opt 2.522 0.4989 0.02059 -0.17821
Solution 1
Upper-Surface Isobars
(Contours at 0.05 Cp)
43.8% Span
31.2% Span
18.8% Span
6.2% Span
93.8% Span
81.3% Span
68.8% Span
56.2% Span
X / C
Cp
COMPPLOT
Ver 2.04
BOEING
John C. Vassberg
18:08 Sun
14 Dec 14


COMPARISON OF SPANLOAD DISTRIBUTIONS
ADODG-CRM CASE 4.2 OPTIMIZATION
REN = 5.00 , MACH = 0.850
SYMBOL SOURCE ALPHA CL CD CM
Baseline 2.196 0.5011 0.02155 -0.18517
Case 4.2 Opt 2.522 0.4989 0.02059 -0.17821
C*CL/CREF
CL
SPANLOAD & SECT CL
PERCENT SEMISPAN
0.0
0.1
0.2
0.3
0.4
0.5
0.6
0.7
0.8
10.0
20.0
30.0
40.0
50.0
60.0
70.0
80.0
90.0
100.0
110.0

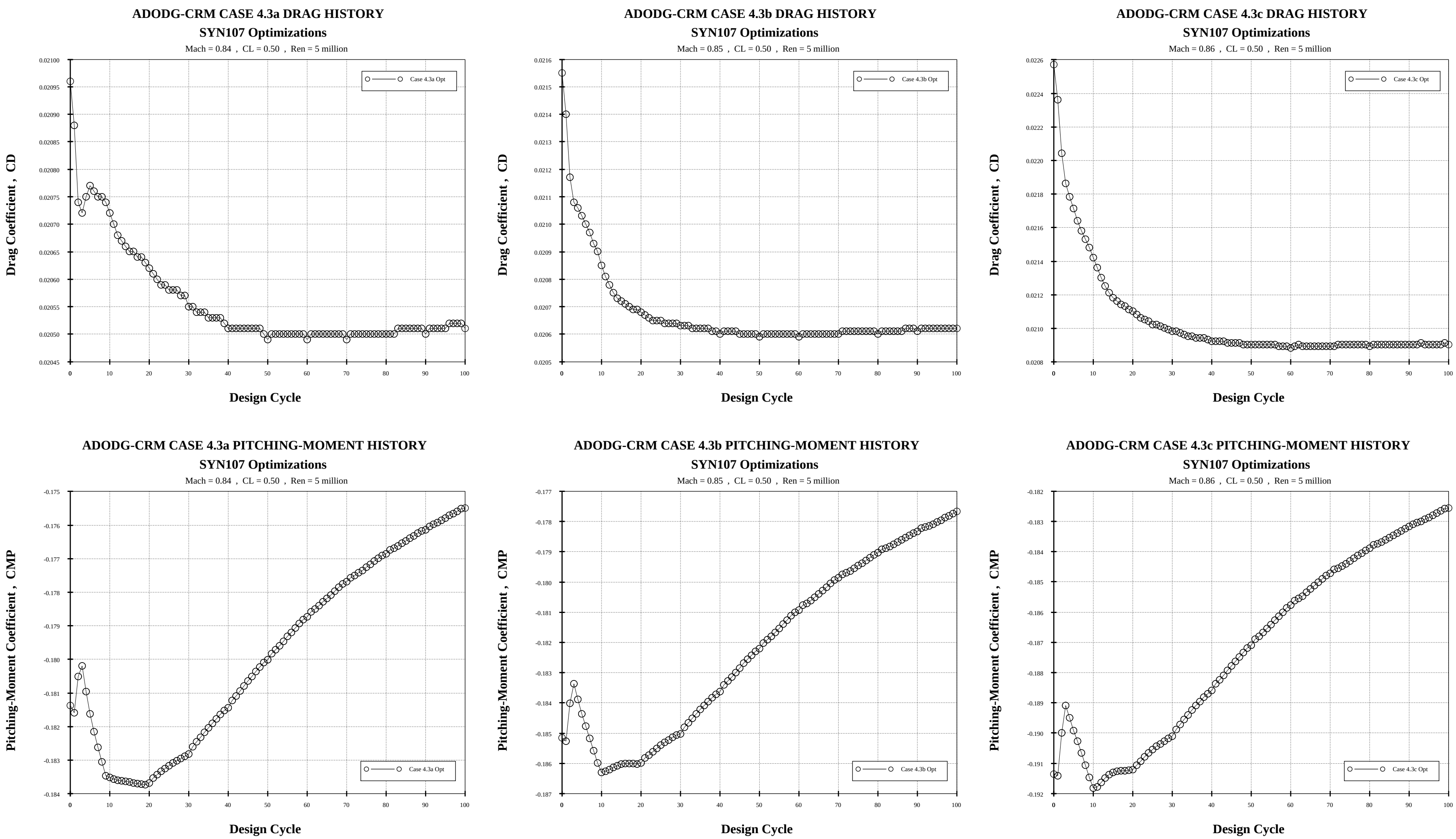

COMPARISON OF CHORDWISE PRESSURE DISTRIBUTIONS
ADODG-CRM CASE 4.2 OPTIMIZATION
REN = 5.00 , MACH = 0.850
SYMBOL SOURCE ALPHA CL CD CM
Baseline 2.552 0.5513 0.02453 -0.20390
Case 4.2 Opt 2.868 0.5489 0.02311 -0.19507
Solution 1
Upper-Surface Isobars
(Contours at 0.05 Cp)
43.8% Span
31.2% Span
18.8% Span
6.2% Span
93.8% Span
81.3% Span
68.8% Span
56.2% Span
X / C
Cp
COMPPLOT
Ver 2.04
BOEING
John C. Vassberg
18:11 Sun
14 Dec 14


COMPARISON OF SPANLOAD DISTRIBUTIONS
ADODG-CRM CASE 4.2 OPTIMIZATION
REN = 5.00 , MACH = 0.850
SYMBOL SOURCE ALPHA CL CD CM
Baseline 2.552 0.5513 0.02453 -0.20390
Case 4.2 Opt 2.868 0.5489 0.02311 -0.19507
C*CL/CREF
CL
SPANLOAD & SECT CL
PERCENT SEMISPAN
0.0
0.1
0.2
0.3
0.4
0.5
0.6
0.7
0.8
0
10
20
30
40
50
60
70
80
90
100

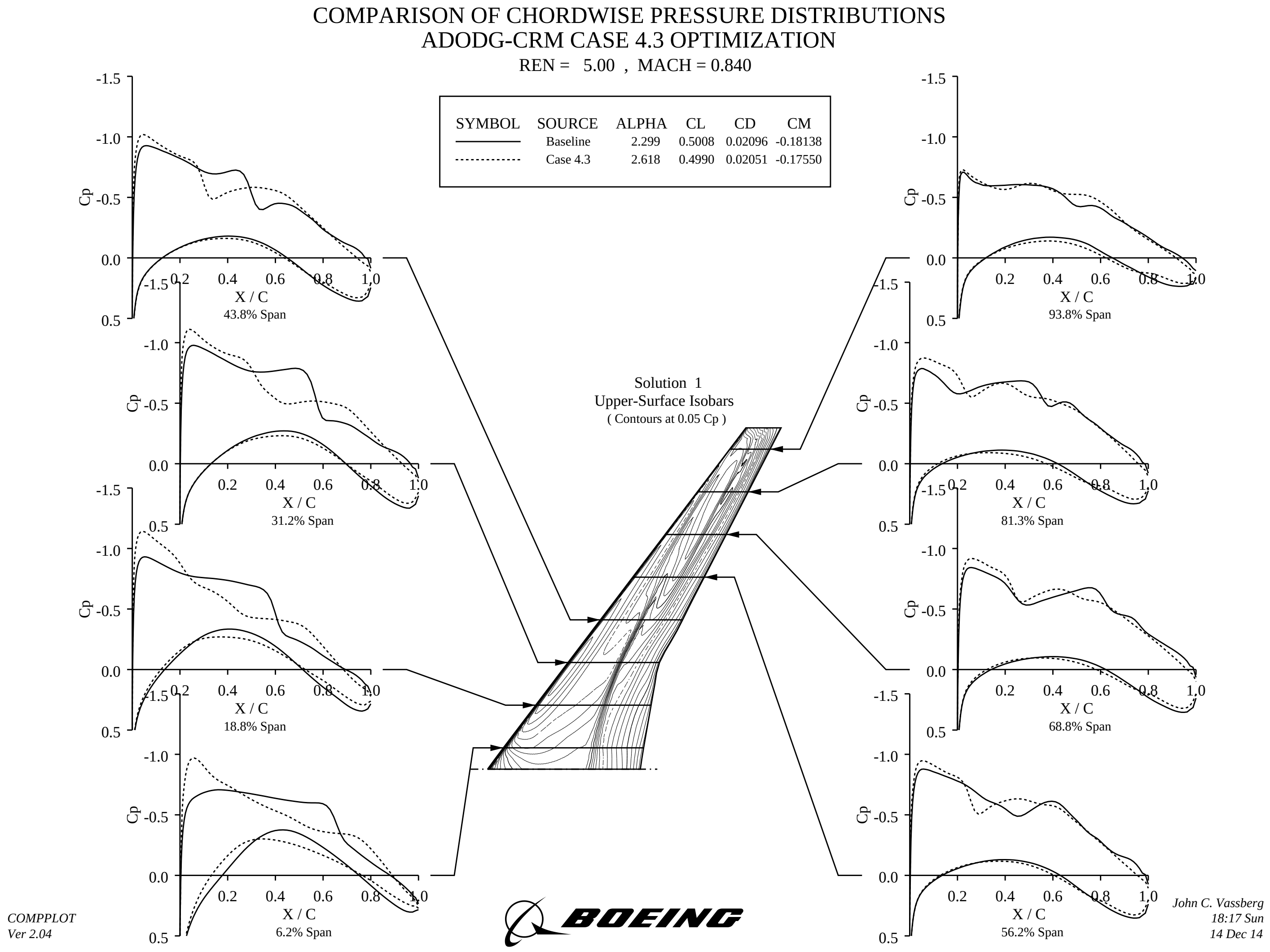

COMPARISON OF CHORDWISE PRESSURE DISTRIBUTIONS
ADODG-CRM CASE 4.2 OPTIMIZATION
REN = 5.00 , MACH = 0.850
SYMBOL SOURCE ALPHA CL CD CM
Case 4.2a 2.146 0.4489 0.01856 -0.16295
Case 4.2b 2.522 0.4989 0.02059 -0.17821
Case 4.2c 2.868 0.5489 0.02311 -0.19507
Solution 1
Upper-Surface Isobars
(Contours at 0.05 Cp)
Cp
X / C
43.8% Span
31.2% Span
18.8% Span
6.2% Span
93.8% Span
81.3% Span
68.8% Span
56.2% Span
COMPPLOT
Ver 2.04
BOEING
John C. Vassberg
17:41 Sun
14 Dec 14

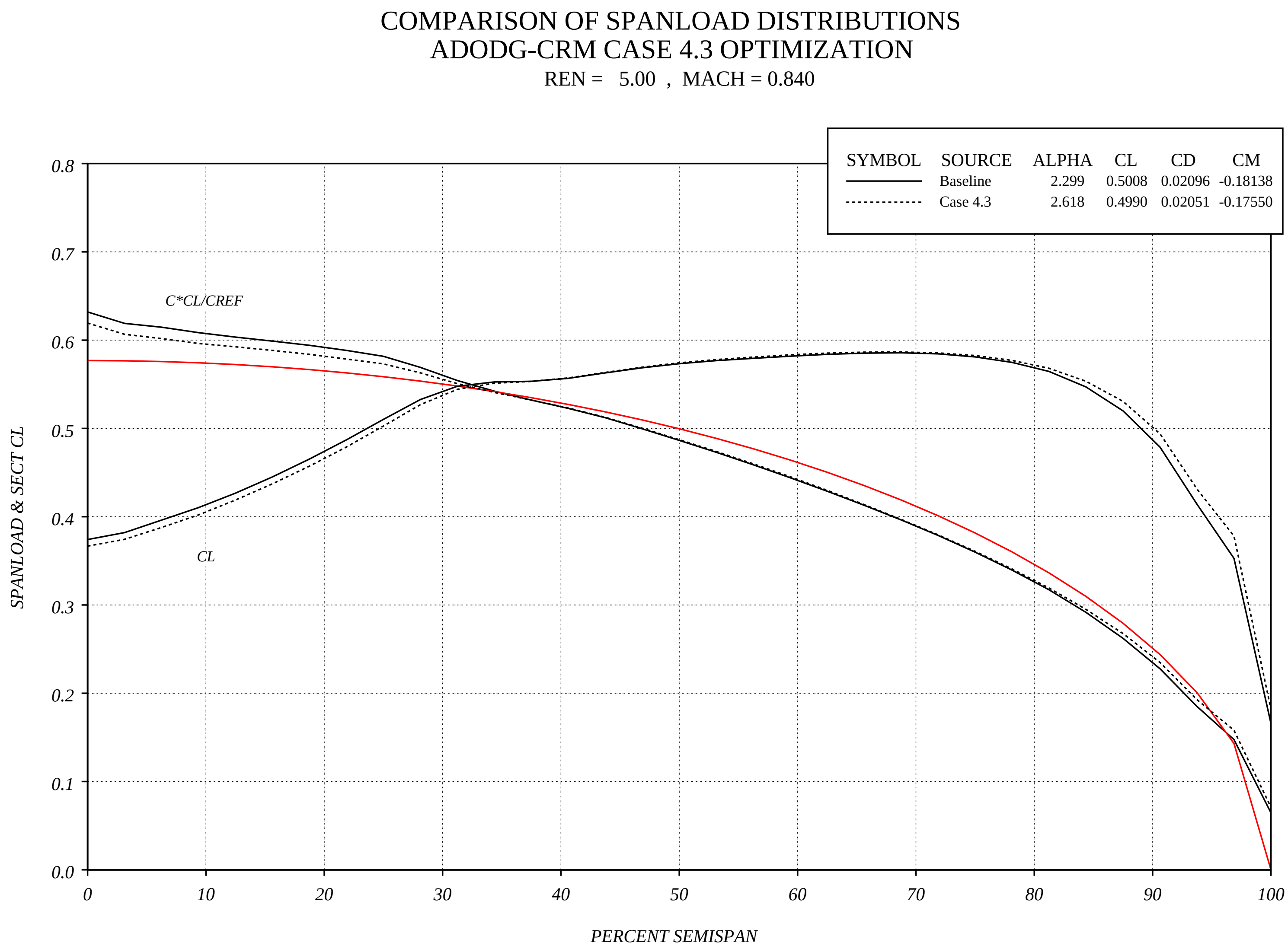
ADO-DG CASE 4.3: Triple-Point M

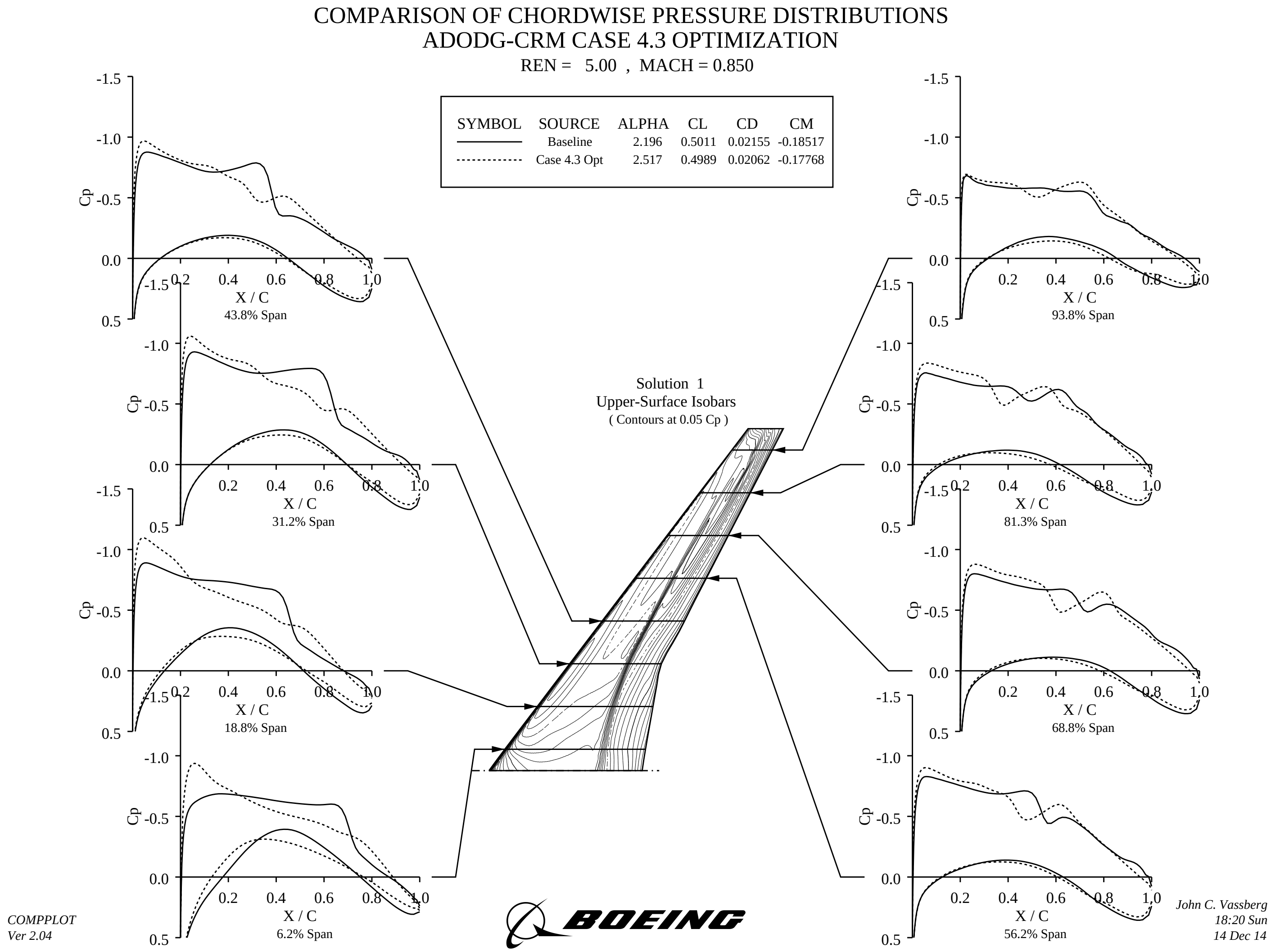
- $M = [0.84\ ,\ 0.85\ ,\ 0.86]$, $C_L = 0.50$, $Re = 5.0{\times}10^6$
- **Design Cycle Histories, All 3 Conditions**
  - Drag & Pitching Moment
- **Baseline .vs. Final Design Comparisons**
  - Overlaid Pressure Distributions
  - Overlaid Spanload Distributions



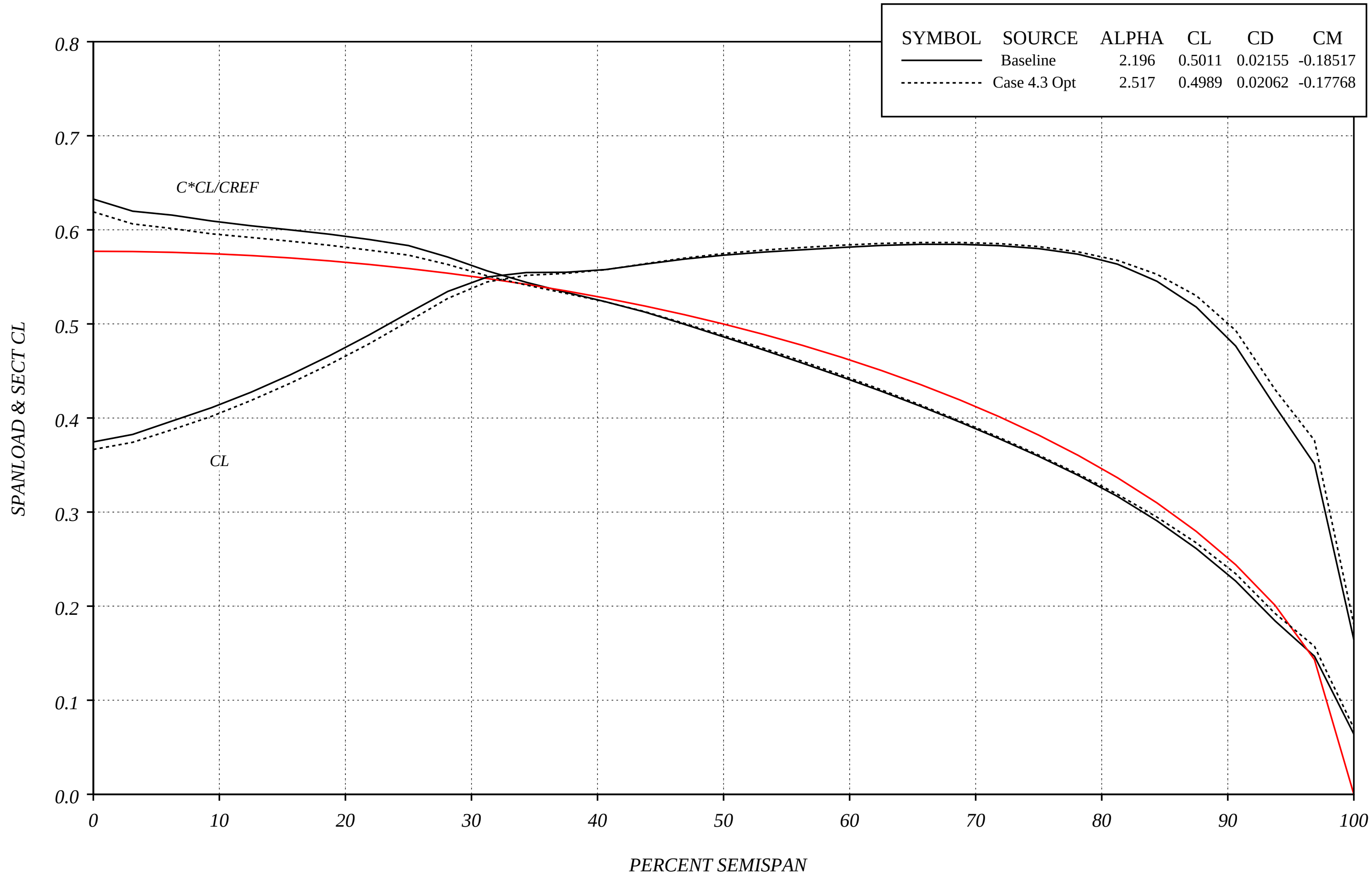
ADO-DG Case 4.3 SYN107 Design-Cycle Histories.

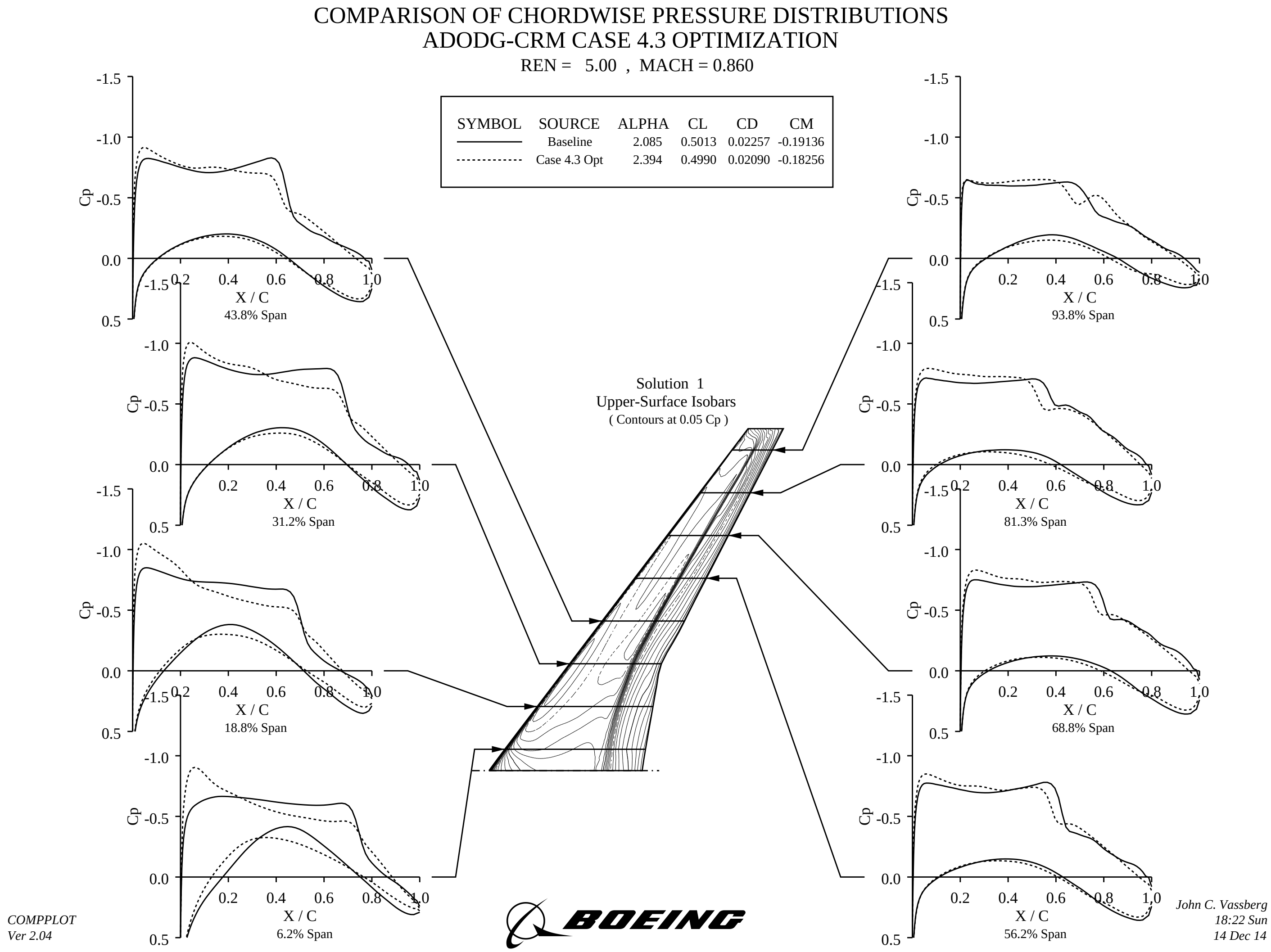

COMPARISON OF CHORDWISE PRESSURE DISTRIBUTIONS
ADODG-CRM CASE 4.3 OPTIMIZATION
REN = 5.00 , MACH = 0.840
SYMBOL SOURCE ALPHA CL CD CM
Baseline 2.299 0.5008 0.02096 -0.18138
Case 4.3 2.618 0.4990 0.02051 -0.17550
Solution 1
Upper-Surface Isobars
(Contours at 0.05 Cp)
Cp
X / C
43.8% Span
31.2% Span
18.8% Span
6.2% Span
93.8% Span
81.3% Span
68.8% Span
56.2% Span
BOEING
COMPPLOT
Ver 2.04
John C. Vassberg
18:17 Sun
14 Dec 14

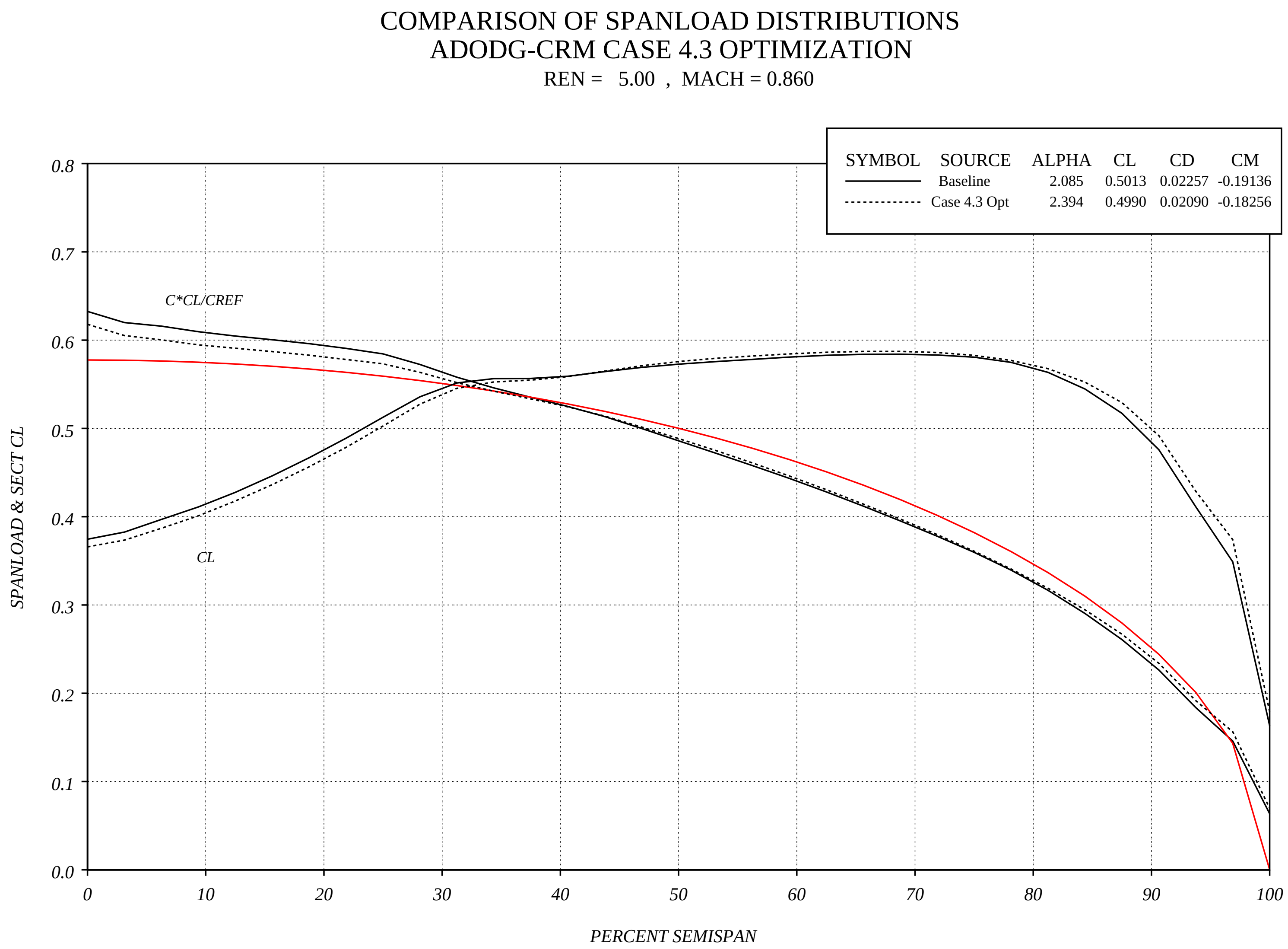

COMPARISON OF SPANLOAD DISTRIBUTIONS
ADODG-CRM CASE 4.3 OPTIMIZATION
REN = 5.00 , MACH = 0.840
SYMBOL SOURCE ALPHA CL CD CM
Baseline 2.299 0.5008 0.02096 -0.18138
Case 4.3 2.618 0.4990 0.02051 -0.17550
C*CL/CREF
CL
SPANLOAD & SECT CL
PERCENT SEMISPAN
0.0
0.1
0.2
0.3
0.4
0.5
0.6
0.7
0.8
0
10
20
30
40
50
60
70
80
90
100

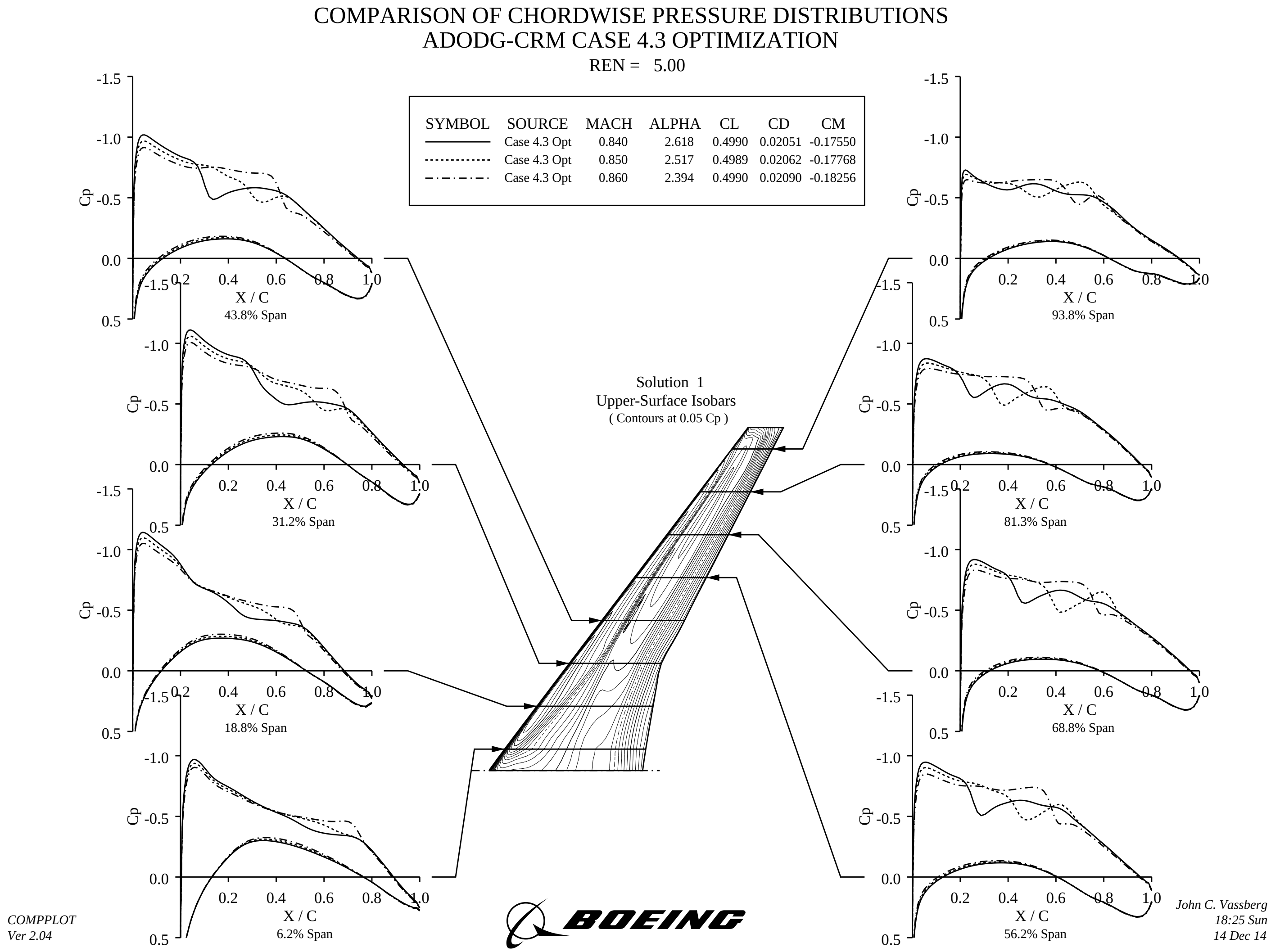

COMPARISON OF CHORDWISE PRESSURE DISTRIBUTIONS
ADODG-CRM CASE 4.3 OPTIMIZATION
REN = 5.00 , MACH = 0.850
SYMBOL SOURCE ALPHA CL CD CM
Baseline 2.196 0.5011 0.02155 -0.18517
Case 4.3 Opt 2.517 0.4989 0.02062 -0.17768
Solution 1
Upper-Surface Isobars
(Contours at 0.05 Cp)
43.8% Span
31.2% Span
18.8% Span
6.2% Span
93.8% Span
81.3% Span
68.8% Span
56.2% Span
Cp
X / C
COMPPLOT
Ver 2.04
BOEING
John C. Vassberg
18:20 Sun
14 Dec 14

COMPARISON OF SPANLOAD DISTRIBUTIONS
ADODG-CRM CASE 4.3 OPTIMIZATION
REN = 5.00 , MACH = 0.850




COMPARISON OF CHORDWISE PRESSURE DISTRIBUTIONS
ADODG-CRM CASE 4.3 OPTIMIZATION
REN = 5.00 , MACH = 0.860
SYMBOL SOURCE ALPHA CL CD CM
Baseline 2.085 0.5013 0.02257 -0.19136
Case 4.3 Opt 2.394 0.4990 0.02090 -0.18256
Solution 1
Upper-Surface Isobars
(Contours at 0.05 Cp)
Cp
X / C
43.8% Span
31.2% Span
18.8% Span
6.2% Span
93.8% Span
81.3% Span
68.8% Span
56.2% Span
COMPPLOT
Ver 2.04
BOEING
John C. Vassberg
18:22 Sun
14 Dec 14


COMPARISON OF SPANLOAD DISTRIBUTIONS
ADODG-CRM CASE 4.3 OPTIMIZATION
REN = 5.00 , MACH = 0.860
SYMBOL SOURCE ALPHA CL CD CM
Baseline 2.085 0.5013 0.02257 -0.19136
Case 4.3 Opt 2.394 0.4990 0.02090 -0.18256
C*CL/CREF
CL
SPANLOAD & SECT CL
PERCENT SEMISPAN
0.0
0.1
0.2
0.3
0.4
0.5
0.6
0.7
0.8
0
10
20
30
40
50
60
70
80
90
100


COMPARISON OF CHORDWISE PRESSURE DISTRIBUTIONS
ADODG-CRM CASE 4.3 OPTIMIZATION
REN = 5.00
SYMBOL SOURCE MACH ALPHA CL CD CM
Case 4.3 Opt 0.840 2.618 0.4990 0.02051 -0.17550
Case 4.3 Opt 0.850 2.517 0.4989 0.02062 -0.17768
Case 4.3 Opt 0.860 2.394 0.4990 0.02090 -0.18256
Solution 1
Upper-Surface Isobars
(Contours at 0.05 Cp)
Cp
X / C
43.8% Span
31.2% Span
18.8% Span
6.2% Span
93.8% Span
81.3% Span
68.8% Span
56.2% Span
COMPPLOT
Ver 2.04
BOEING
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14 Dec 14

ADO-DG CASE 4 RESULTS

- $M = 0.85$, $C_L = 0.50$, $Re = 5.0 \times 10^6$
- **Comparison of Baseline & Final Designs**
  - Tabulated Data
  - Overlaid Pressure Distributions
- **Drag Polars; Alpha Sweeps**
- **Drag Rises; Mach Sweeps**

ADO-DG CASE 4 RESULTS

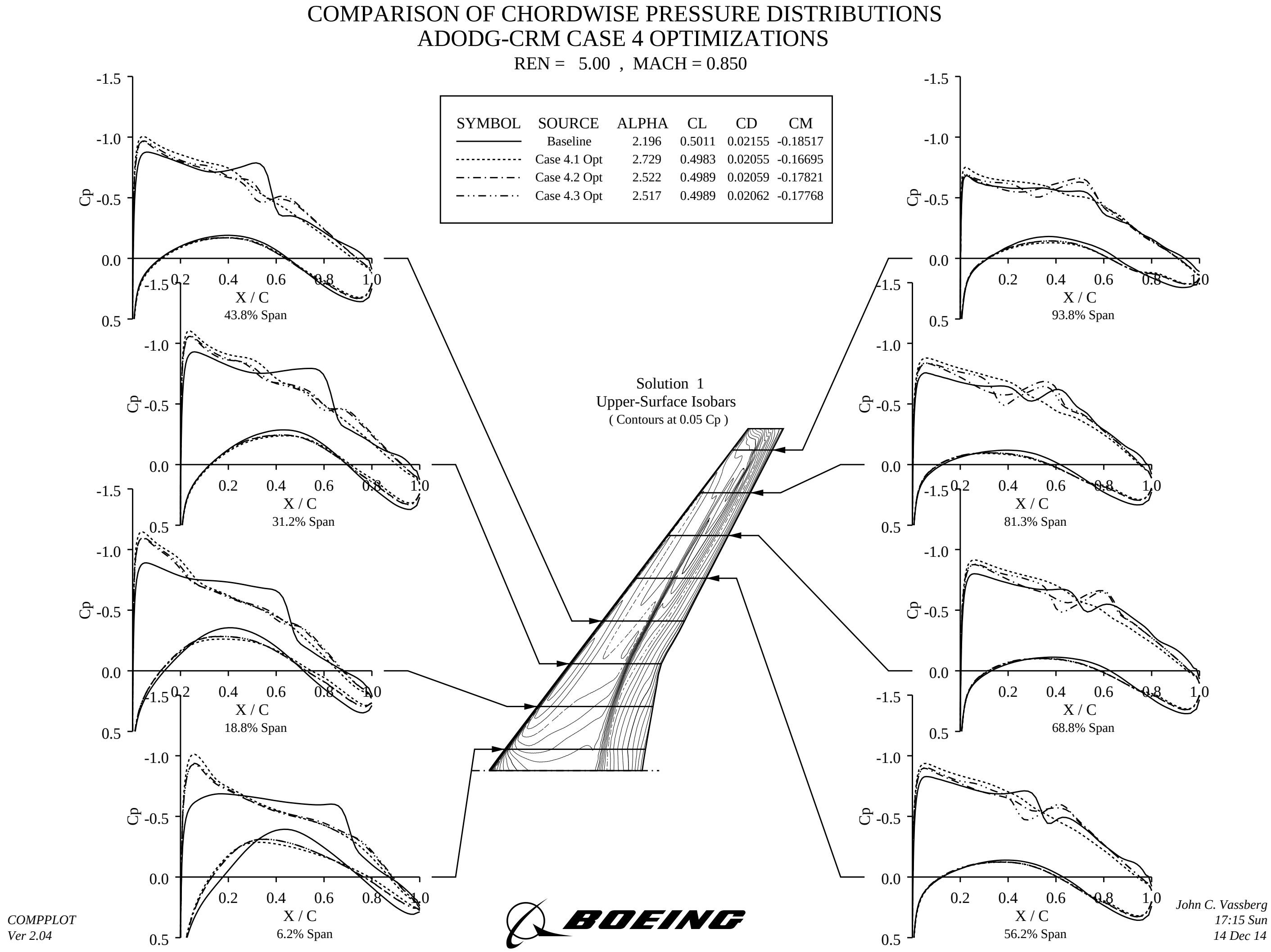
Baseline ADO-CRM-Wing.			
M	C_L	$C_D cor$	$C_M cor$
0.84	0.50	0.02092	-0.18114
0.85	0.45	0.01906	-0.16710
0.85	0.50	0.02149	-0.18482
0.85	0.55	0.02443	-0.20340
0.86	0.50	0.02249	-0.19087

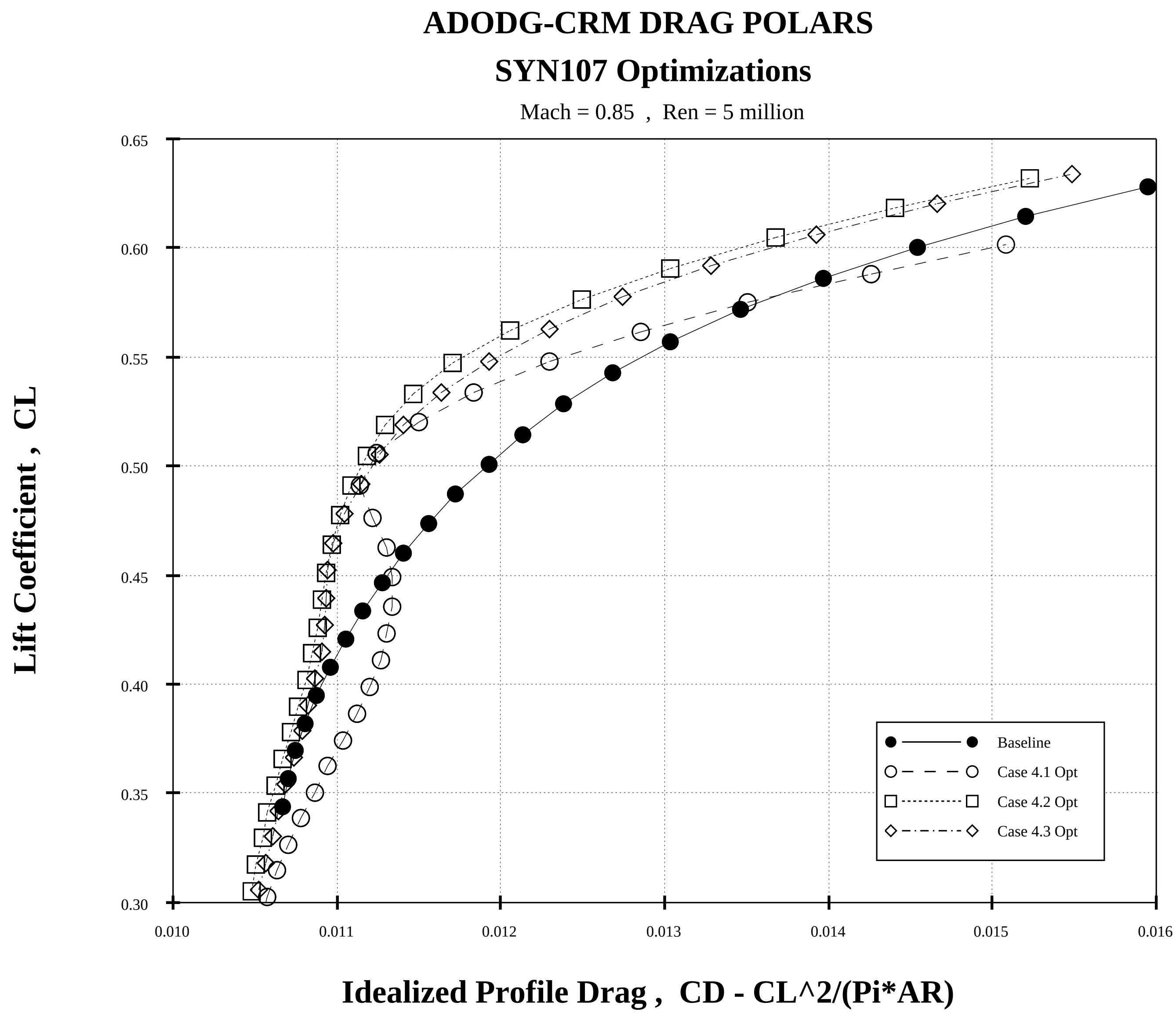
Case 4.1 Optimum Wing.				
M	C_L	$C_D cor$	ΔC_D	$C_M cor$
0.85	0.50	0.02062	-0.00087	-0.16750

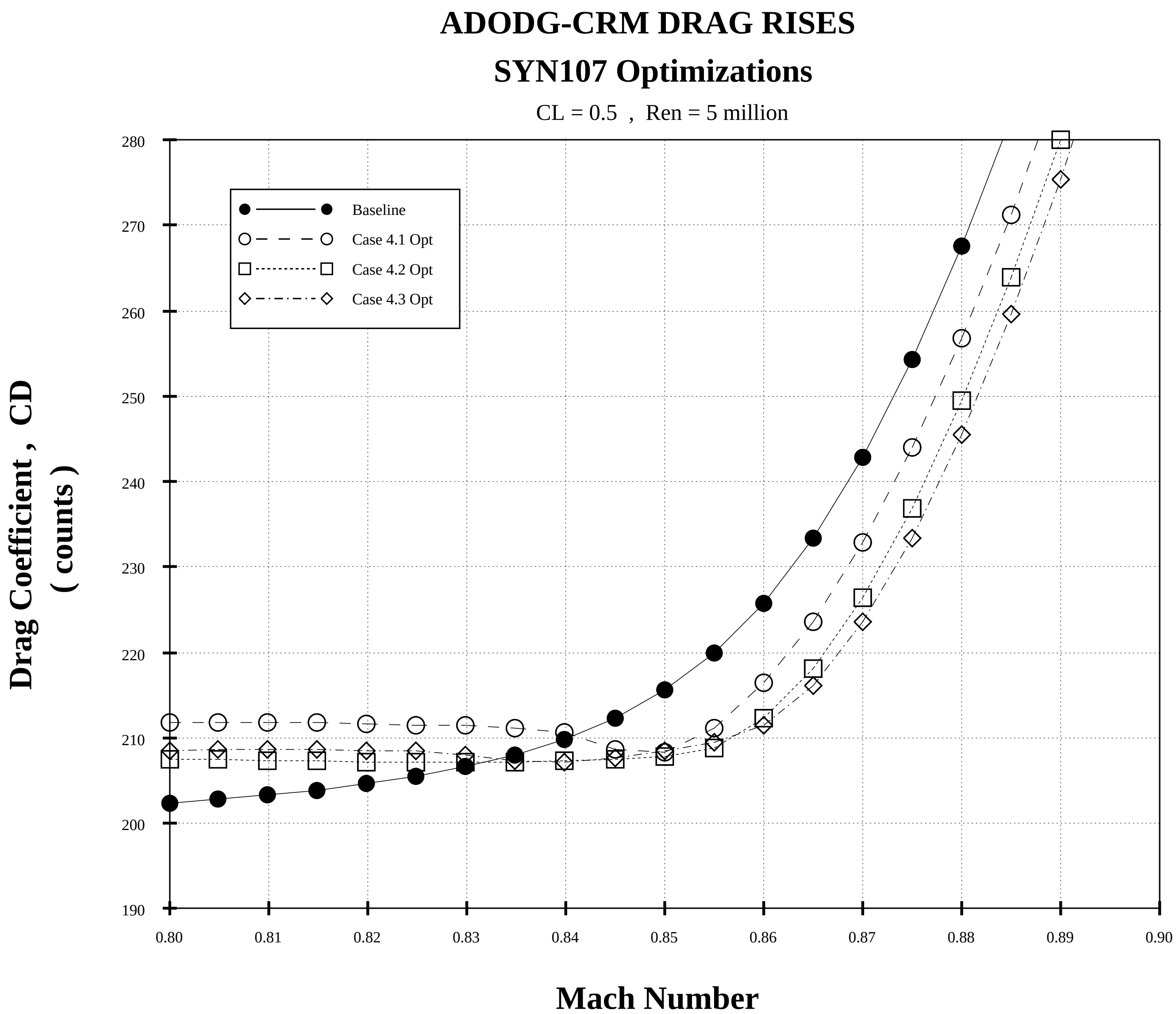
ADO-DG CASE 4 RESULTS

Case 4.2 Optimum Wing.				
M	C_L	$C_D cor$	ΔC_D	$C_M cor$
0.85	0.45	0.01861	-0.00045	-0.16330
0.85	0.50	0.02064	-0.00085	-0.17855
0.85	0.55	0.02318	-0.00125	-0.19548

Case 4.3 Optimum Wing.				
M	C_L	$C_D cor$	ΔC_D	$C_M cor$
0.84	0.50	0.02056	-0.00036	-0.17578
0.85	0.50	0.02066	-0.00083	-0.17801
0.86	0.50	0.02096	-0.00153	-0.18294


COMPARISON OF CHORDWISE PRESSURE DISTRIBUTIONS
ADODG-CRM CASE 4 OPTIMIZATIONS
REN = 5.00 , MACH = 0.850
SYMBOL SOURCE ALPHA CL CD CM
Baseline 2.196 0.5011 0.02155 -0.18517
Case 4.1 Opt 2.729 0.4983 0.02055 -0.16695
Case 4.2 Opt 2.522 0.4989 0.02059 -0.17821
Case 4.3 Opt 2.517 0.4989 0.02062 -0.17768
Solution 1
Upper-Surface Isobars
(Contours at 0.05 Cp)
Cp
X / C
43.8% Span
31.2% Span
18.8% Span
6.2% Span
93.8% Span
81.3% Span
68.8% Span
56.2% Span
BOEING
COMPPLOT
Ver 2.04
John C. Vassberg
17:15 Sun
14 Dec 14


ADODG-CRM DRAG POLARS
SYN107 Optimizations
Mach = 0.85 , Ren = 5 million
Lift Coefficient , CL
Idealized Profile Drag , CD - CL^2/(Pi*AR)
0.65
0.60
0.55
0.50
0.45
0.40
0.35
0.30
0.010
0.011
0.012
0.013
0.014
0.015
0.016
Baseline
Case 4.1 Opt
Case 4.2 Opt
Case 4.3 Opt


ADODG-CRM DRAG RISES
SYN107 Optimizations
CL = 0.5 , Ren = 5 million
Baseline
Case 4.1 Opt
Case 4.2 Opt
Case 4.3 Opt
Drag Coefficient , CD (counts)
Mach Number
190
200
210
220
230
240
250
260
270
280
0.80
0.81
0.82
0.83
0.84
0.85
0.86
0.87
0.88
0.89
0.90

ADO-DG CASE 4 SUMMARY

- Optimizations Yield Designs with Near Theoretical Minimum Drag
  - Essentially No Shock Drag
  - Near Minimum Induced Drag
  - Near Minimum Profile Drag per Thickness Distribution

- Case 4.1 Single-Point Design
  - Exhibits Poor Off-Design Performance

- Case 4.2-4.3 Triple-Point Designs
  - Relinquished < 0.5 counts at Cruise Condition

- Multi-Point Optimizations Clearly Improve Off-Design Characteristics

Select ADO-DG Case Studies in Aerodynamic Design Optimization

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John C. Vassberg

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Invited Presentation

AIAA Sci-Tech Conference
Kissimmee, FL
5 January, 2015