

LAMDES User's Manual

This is a modified version of John Lamar's design program (Ref. 1). It is based on the vortex lattice program **VLMpc**, but converted to do design and optimization.

Basic capabilities include:

- For a single lifting surface the code will find the twist and camber required for minimum induced drag. For a planar surface this is of course an elliptic spanload.
- When two surfaces are included the code will do an optimization finding the minimum trimmed induced drag and required spanload subject to the input *cg* location. The required twist distribution and camber shape for subsonic flow are also found.
- A wing root bending moment constraint can also be specified, and the minimum induced drag satisfying the constraint will be found together with the associated required spanload and twist and camber.
- Conversely, the spanloads can be specified and the span *e* will be computed.
- The airfoil section drag can be input and the program can find the minimum trimmed drag of the sum of the induced and profile drag. If the airfoil polar is parabolic, the problem still results in the solution of a linear system of equations. If the airfoil polars are not parabolic, an iteration is used to find the minimum of the sum of induced and profile drag. See Ref. 2 for the details and examples for this extension. Note that this code was used to understand how to develop the Grumman X-29, where the limits of the airfoil performance had to be included in the design. Please note that for normal cruise conditions the profile drag should be very small and not needed to minimize drag. This capability is for maneuvering conditions.
- The moment contribution from the fuselage can be included by specifying a value for the fuselage moment.

Notes:

1. The computed camber and twist distributions are only valid for subsonic flow. However, although the camber lines are absolutely invalid for transonic flow, the computed twist distribution has been found to be a reasonable starting point for transonic wing design.
2. The "twist" distribution is actually an incidence relative to the freestream velocity. The user can decide what's the angle of attack and what's the "twist."
3. Profile drag of an airfoil includes skin friction, pressure drag and shock wave drag. Drag definitions can be somewhat arbitrary. Pressure drag is sometimes known as form drag. In addition, some form of simple sweep theory needs to be used to convert the 2D polar to the streamwise polar. The code uses strip theory to find the total profile drag.

References:

1. J.E. Lamar, "A Vortex Lattice Method for the Mean Camber Shapes of Trimmed Non-Coplanar Planforms with Minimum Vortex Drag," NASA TN D-8090, June, 1976.
2. W.H. Mason, "Wing-Canard Aerodynamics at Transonic Speeds - Fundamental Considerations on Minimum Drag Spanloads," AIAA Paper No. 82-0097, January 1982.

The following example shows how the code is used. The geometry used for the study is given in Fig. 1. It is the test case geometry in Lamar's NASA TN. You find out how to balance a two-surface configuration for minimum trimmed drag. The static margin is found using VLMpc.

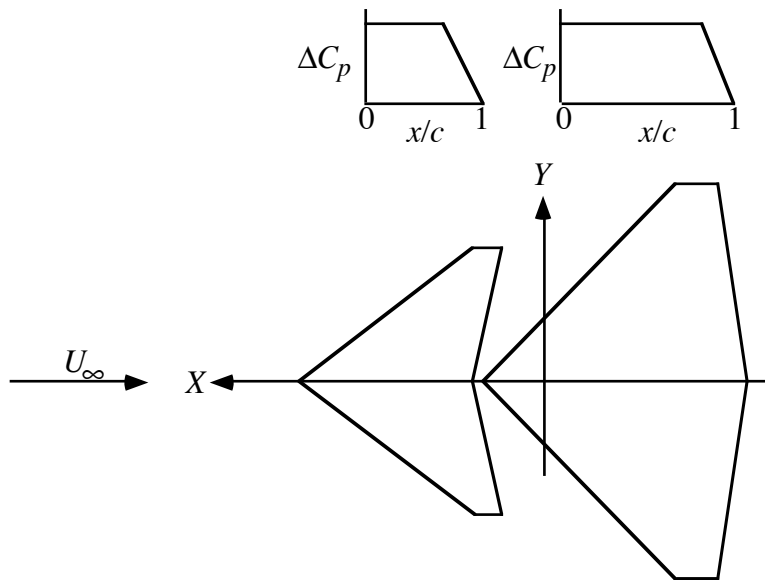


Figure 1. Example geometry to study trimmed drag.

Figure 2 shows the results obtained for the minimum trimmed drag over a range of cg locations for three different vertical separations between the canard and the wing. In this case the minimum trimmed drag is found for a cg that results in an unstable design. This is why the concept of relaxed static stability and canards were of interest in the 1970s. It also shows why vertical separation can reduce the drag. Although a canard configuration can't have a large vertical separation, this is why Joe Lippert used the maximum possible separation on the Grumman Ag-Cat biplane.

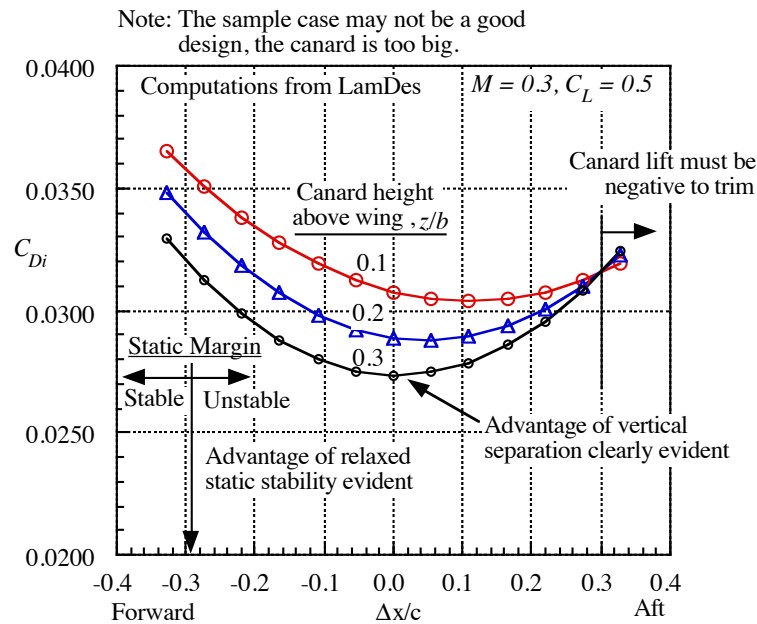


Figure 2. Trimmed drag for various cg 's and vertical separations.

To run the program

The program prompts the user for the input file name. A sample input and a portion of the corresponding output are included here. The user should study the sample input carefully.

Input Instructions:

The program assumes the load distribution on each surface is constant chordwise until a designated chordwise location (XCFW on the first surface and XCFT on the second surface). The loading then decreases linearly to the trailing edge. This corresponds to 6 & 6A series camber line distributions (the value for the 6A series is usually 0.8). If airfoil polars are used to model the effects of viscosity in the airfoil section, the polars are input in a streamwise coordinate system. The user is responsible for adjusting them from 2D to 3D.

This program uses an input file that is very similar to, *but not the same as*, the VLMpc code. It is based on the same geometry and coordinate system ideas. The VLMpc manual should be consulted for a discussion of the geometry system. The code has its heritage in the days of cards, and it is important that the input data be located in the correct columns (with no tabs!).

In this program you should not try to model the fuselage as you would using VLMpc. Just extend the lifting surfaces to the centerline.

Input Format

<u>Card #</u>	<u>Format</u>	<u>Field</u>	<u>Name</u>	<u>Remarks</u>
1	Literal		DATA	Title card for the data set
2	8F10.6	1	PLAN	Number of lifting surfaces for the configuration; use 1.0 or 2.0.
		2	XMREF	<i>c.g.</i> shift from origin of the input planform coordinate system (the program originally trimmed the configuration about the input planform origin). + is a <i>c.g.</i> shift forward - is a <i>c.g.</i> shift aft
		3	CREF	reference chord of the configuration, used only to nondimensionalize the pitching moment coefficients.
		4	SREF	reference area of the configuration
		5	TDKLUE	minimization clue = 0.0 - minimize induced drag only = 1.0 - minimize induced plus profile drag
		6	CASE	options for the drag polar = 0.0, model polar, same <i>a</i> , CLmin, CD0 for each surface(see note 3 below). = 1.0, model polar, each surface has its own <i>a</i> , CLmin, CD0 = 2.0, one general polar for entire config. = 3.0, one general polar for each surface
		7	SPNKLU	spanload clue = 0.0 spanload is internally computed

using the minimization
 = 1.0, no minimization is done, the spanload is read in, and the span e and pressure drag are computed.

Geometric/Planform Data - see the VLMpc manual for a complete description

<u>Card #</u>	<u>Format</u>	<u>Field</u>	<u>Name</u>	<u>Remarks</u>
1-P	8F10.6	1	AAN(IT)	# of straight lines defining this surface
		2	XS(IT)	= 0.0 (not used in this code)
		3	YS(IT)	= 0.0 (not used in this code)
		4	RTCDHT(IT)	root chord height ("-" is "higher") Note: for two surfaces they can't both be in the same plane.
		5	PDRG1(IT)	C_{lmin} - see Note 3 for definitions
		6	PDRG2(IT)	" af "
		7	PDRG3(IT)	C_{d0}
2-P	8F10.6	1	XREG	X point of line segment (positive is forward)
		2	YREG	Y point of line segment (positive is forward)
		3	DIH	dihedral angle of line
		4	AMCD	sweep wing move code, set = 1.0 for this program

- Note:
1. Card 2-P is read in AAN + 1 times. The surface description starts at the forward centerline and works outboard and around, returning to the aft centerline of the surface.
 2. Cards 1-P and 2-P are read in as a set for each lifting surface
 3. If airfoil section drag is included using the model polar it is given by:

$$C_d = C_{d0} + af(C_l - C_{lmin})^2$$

Control Data (corresponding to "Group Two" data in Lamar's nomenclature)

<u>Card #</u>	<u>Format</u>	<u>Field</u>	<u>Name</u>	<u>Remarks</u>
1-C	6F5.3, 2F10.6	1	CONFIG	arbitrary configuration number or ID (may include up to four digits)
		2	SCW	Number of chordwise horseshoe vortices to be used to represent the wing; a maximum of 20 may be used, do not set to zero.
		3	VIC	nominal number of spanwise rows at which chordwise horseshoe vortices may be located; a maximum of 50 may be used. The product of SCW and SSW cannot exceed 400.

		4	XMCH	Mach number, used to apply the Prandtl-Glauert compressibility correction factor.
		5	CLDES	design lift coefficient
		6	XITMAX	Maximum number of iterations allowed in finding the solution for minimum + profile drag with arbitrary polars input. Must be less than 50. 20 is sufficient for most cases.
		7	EPSMAX	The convergence criteria for the general polar case. A value of .0005 appears to be reasonable.
2-C	6F10.4	1	XCFW	The chord fraction at which the chord load shape changes from rooftop to a linear decrease to zero at the trailing edge on the first planform. See the introduction to this section for more discussion.
		2	XCFT	Same as XCFW, except applies to the second planform.
		3	FKON	Clue for constraints = 0.0 body moment constraint = 1.0 no constraints = 2.0 root bending moment constraint = 3.0 both moment and root bending moment constraints.
		4	CMB	The design wing CM when FKON = 0.0
		5	FICAM	Camber computation clue. = 0.0, no cambers computed = 1.0, wing cambers computed Note: To get the twist table the cambers must be computed, FICAM = 1.0
		6	PUNCH	clue to write cambers to a file = 0.0 - no camber file created = 1.0 - cambers output (unit 7)
		7	CRBMT	Design root bending moment for FKON = 2.0.
3-C	8F10.6	1	RELAX	The under-relaxation factor for the general polar solution. RELAX = 0.03 to 0.3 is satisfactory for most applications.
		2	FIOUTW	Output clue. = 0.0 - debug output = 1.0 - only final results are output
		3	CD0	Basic drag coefficient that will be added to the drag computed by summing the

induced drag and the profile drag contained in the input polars.

Arbitrary Polar Input (the following cards are read only if CASE \geq 2.0)

<u>Card #</u>	<u>Format</u>	<u>Field</u>	<u>Name</u>	<u>Remarks</u>
1-D	Literal		TITLE	The identifying title for the input drag polar for this surface.
2-D	8F10.5	1	FNCLCD	The number of CL,CD pairs used to define the input polar.
3-D	8F10.5	1	FQCL	The value of streamwise lift coefficient for this point on the drag polar.
		2	FQCD	The value of streamwise drag coefficient for the given lift coefficient.

- Note: 1. Card 3-D is read FNCLCD times
 2. Cards 1-D, 2-D and 3-D are read for each planform if CASE = 3.0.

Spanload Input (the following cards are read only if SPNKLU = 1.0)

<u>Card #</u>	<u>Format</u>	<u>Field</u>	<u>Name</u>	<u>Remarks</u>
1-S	Literal		TITLE	This is the title card for the input spanloads.
2-S	7F10.5	1	FSPNPT	Number of points on the spanload to be read in for this planform.
3-S	7F10.5	1	YSPNPT	Span location in physical coordinates at which ccl/ca is input (y is positive here!)
		2	CLSPNP	The spanload at YSPNPT

- Note: 1. Card 3-S is read FSPNPT times
 2. Cards 2-S and 3-S are read for each planform as a set.

A general comment: this program is extremely valuable for an experienced aerodynamicist. Before trying to use this program on a specific design problem, the user must run and examine the sample cases and then should experiment with the various options to get to understand the program. Although the TN has a winglet example, the planform is divided based on the projection, not the running length, so the winglet may a very small number of stations. Be careful.

Output Description:

The program output was extensively re-organized from the NASA TN code for typical computer screen use rather than as a “printout”.

The initial output includes an echo of the input to facilitate debugging. Initially these are the various control and reference conditions. The next section provides the geometry input. These should be checked to ensure that the correct planform is being used.

The next section includes more of the program control options. They should be examined to make sure that the program is making the desired computation.

Next, if the general airfoil polar is used in the minimization, the iteration history is output. At each iteration the induced and pressure drag are printed. When the iteration is converged the values should not be changing. The iteration history is then summarized. Generally the initial c_{di} is lowest, and the c_{dp} is highest. As the sum of the two is reduced the induced drag will rise as the spanload changes to reduce the pressure drag.

Finally, the results are output. First, the computed induced drag and pressure drag are output, together with more computed results, such as the true area and AR , etc. Results are presented for each planform. Note that span e is output. In this program the elliptic spanload is typically found to be 1.01, indicating the level of discretization error since the correct value is 1.0.

Then the resulting spanload, section c_l and c_d distributions are output.

The next section provides the computed camber slopes then the mean camber lines for each spanwise station. They are given both as relative to the freestream and adjusted so that the camber is zero at both the leading and the trailing edge.

The final section of output summarizes the twist distribution found from the lifting surface camber slopes, integrated previously to get the camber shapers. The “twist” is given as incidence relative to the freestream, in degrees. Note that they start at the tip and move inboard. A negative twist is washout, and a positive value is washin.

Sample Cases: Three sample cases are provided with the code. The first, included here in the manual, is the NASA TN case, LamDesTNsamp.inp, that was used to generate Figure 2. The second is a single planform of aspect ratio 10, sweep of 30° and taper ratio of 0.5, ar1030.inp, that includes the twist and camber required to generate the elliptic spanload. The third sample case, WCfoildes.inp, illustrates the use of a general airfoil polar for a wing – canard configuration.

Sample Input: LamDesTNsamp.inp (note: it is important to put data in proper columns!)

```
Lamar program sample input - TN D-8990 canard example
 2.000   -0.00   9.18   160.0   0.0   0.0   0.0
 3.000    0.0    0.0   -1.00   0.0   0.0
14.57    0.0    0.0    1.0
 5.73   -6.73    0.0    1.0
 4.29   -6.73
 5.77    0.00
 3.0     0.0    0.0    0.0    0.0    0.0
 5.29    0.0    0.0    1.0
-4.45   -10.0   0.0    1.0
-6.61   -10.0   0.0    1.0
-8.12    0.0
2.0 16.0 15. 0.3 0.20 0.0 0.000
0.6   0.80 0.0   0.00 0.0
0.0   1.0 0.0
```

Sample Output: this output only includes the drag, the cambers and twist calculations are lengthy and included on the web site.

williams-mbp:sourcecodefolder whmason\$./LamDes

enter name of input file:

LamDestNsamp.inp

Lamar Design Code mods by W.H. Mason

Lamar program sample input - TN D-8990 canard example

plan = 2.0 xmref = -0.0000 cref = 9.1800
tdklue = 0.0 case = 0.0 spnklu = 0.0
sref = 160.0000

1st REFERENCE PLANFORM HAS 3 CURVES
ROOT CHORD HEIGHT = -1.0000

POINT	X	Y	SWEEP	DIHEDRAL
	REF	REF	ANGLE	ANGLE
1	14.5700	0.0000	52.71754	0.00000
2	5.7300	-6.7300	90.00000	0.00000
3	4.2900	-6.7300	12.40255	0.00000
4	5.7700	0.0000		

2nd REFERENCE PLANFORM HAS 3 CURVES
ROOT CHORD HEIGHT = 0.0000

POINT	X	Y	SWEEP	DIHEDRAL
	REF	REF	ANGLE	ANGLE
1	5.2900	0.0000	44.24539	0.00000
2	-4.4500	-10.0000	90.00000	0.00000
3	-6.6100	-10.0000	-8.58679	0.00000
4	-8.1200	0.0000		

scw = 16.0 vic = 15.0
xitmax = 0.0 epsmax = 0.00000

CONFIGURATION NO. 2.
delta ord shift for moment = -0.0000

CURVE 1 IS SWEPT 52.7175 DEGREES ON PLANFORM 1
CURVE 1 IS SWEPT 44.2454 DEGREES ON PLANFORM 2

BREAK POINTS FOR THIS CONFIGURATION

POINT	X	Y	Z	SWEEP	DIHEDRAL
				ANGLE	ANGLE
1	14.5700	0.0000	-1.0000	52.7175	0.0000
2	5.7300	-6.7300	-1.0000	90.0000	0.0000
3	4.2900	-6.7300	-1.0000	12.4025	0.0000
4	5.7700	0.0000	-1.0000		
SECOND PLANFORM BREAK POINTS					
1	5.2900	0.0000	0.0000	44.2454	0.0000
2	-1.2650	-6.7300	0.0000	44.2454	0.0000
3	-4.4500	-10.0000	0.0000	90.0000	0.0000
4	-6.6100	-10.0000	0.0000	-8.5868	0.0000
5	-8.1200	0.0000	0.0000		

a = 0.000 clmin = 0.000 cd0 = 0.0000

400 HORSESHOE VORTICES USED
 PLANFORM TOTAL SPANWISE
 1 160 10
 2 240 15

16. HORSESHOE VORTICES IN EACH CHORDWISE ROW

xcfw = 0.60 xcft = 0.80 fkon = 0.00
 ficam = 0.00 punch = 0.00 crbmnt = 0.000
 cmb = 0.00 iflag = 1

 relax = 0.00 fioutw = 1.00 cd0 = 0.0000
 firbm = 0.00 yrbm = 0.0000 zrbm = 0.0000

induced drag cd = 0.00401 pressure drag cdpt = 0.00000

induced drag alone was minimized on this run

ref. chord = 9.180 c average = 8.0000 true area = 224.615
 ref. area = 160.000 b/2 = 10.0000 ref ar = 2.5000
 true ar = 1.7808 Mach number = 0.3000

first planform cl = 0.04917 cm = 0.04617 cb = -0.00681
 second planform cl = 0.15083 cm = -0.04617 cb = -0.03411

1st planform CL = 0.0494 CDP = 0.0000 CM = 0.0463 CB = -0.0069
 2nd planform CL = 0.1510 CDP = 0.0000 CM = -0.0463 CB = 0.0000

no root bending moment constraint

CL DES = 0.20000 CL COMPUTED = 0.2004 CM = -0.0000
 CD I = 0.00401 E = 1.2742
 CDPRESS = 0.00000 CDTOTAL = 0.00401

first planform

Y	CL*C/CAVE	C/CAVE	CL	CD
-6.3967	0.02291	0.22557	0.10157	0.00000
-5.7300	0.04442	0.31670	0.14027	0.00000
-5.0633	0.05847	0.40784	0.14337	0.00000
-4.3967	0.06925	0.49897	0.13878	0.00000
-3.7300	0.07774	0.59010	0.13174	0.00000
-3.0633	0.08435	0.68124	0.12382	0.00000
-2.3967	0.08937	0.77237	0.11571	0.00000
-1.7300	0.09300	0.86351	0.10770	0.00000
-1.0633	0.09534	0.95464	0.09987	0.00000
-0.3650	0.09652	1.05011	0.09192	0.00000

second planform

-9.6667	0.06198	0.31687	0.19559	0.00000
-9.0000	0.10043	0.41063	0.24457	0.00000
-8.3333	0.12546	0.50438	0.24874	0.00000
-7.6667	0.14335	0.59813	0.23967	0.00000
-7.0317	0.15469	0.68742	0.22503	0.00000
-6.3967	0.16077	0.77672	0.20698	0.00000
-5.7300	0.16350	0.87047	0.18783	0.00000
-5.0633	0.16515	0.96422	0.17128	0.00000

-4.3967	0.16662	1.05797	0.15749	0.00000
-3.7300	0.16798	1.15172	0.14585	0.00000
-3.0633	0.16919	1.24547	0.13584	0.00000
-2.3967	0.17020	1.33922	0.12709	0.00000
-1.7300	0.17099	1.43297	0.11933	0.00000
-1.0633	0.17152	1.52672	0.11235	0.00000
-0.3650	0.17179	1.62492	0.10572	0.00000

calculation complete