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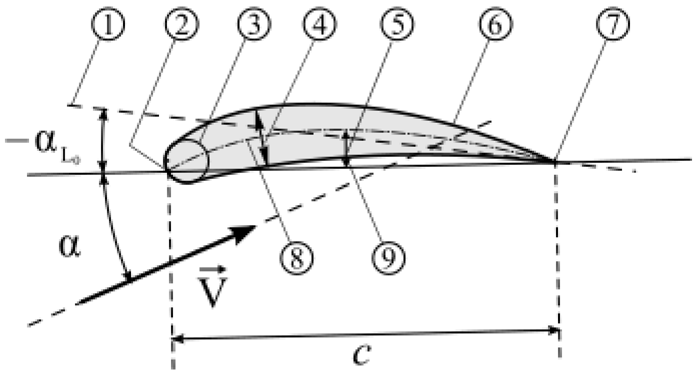
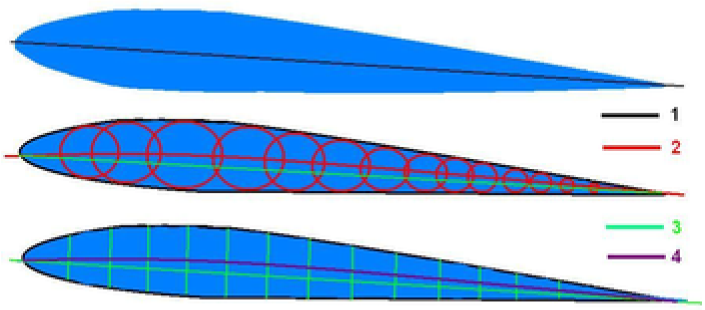
NACA airfoil

The **NACA airfoils** are airfoil shapes for aircraft wings developed by the National Advisory Committee for Aeronautics (NACA). The shape of the NACA airfoils is described using a series of digits following the word "NACA". The parameters in the numerical code can be entered into equations to precisely generate the cross-section of the airfoil and calculate its properties.



Profile geometry – 1: Zero-lift line; 2: Leading edge; 3: Nose circle; 4: Max. thickness; 5: Camber; 6: Upper surface; 7: Trailing edge; 8: Camber mean-line; 9: Lower surface



Profile lines – 1: Chord, 2: Camber, 3: Length, 4: Midline

Contents

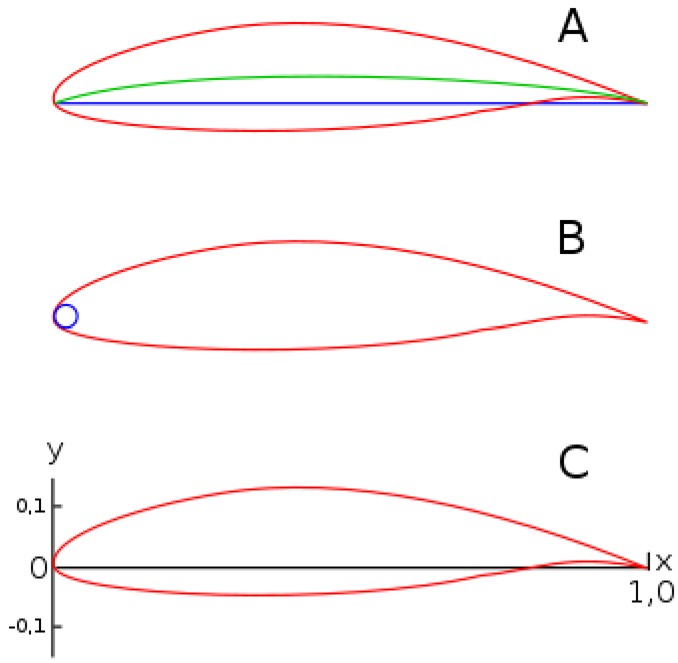
Four-digit series

The NACA four-digit wing sections define the profile by:[1]

1. First digit describing maximum camber as percentage of the chord.
2. Second digit describing the distance of maximum camber from the airfoil leading edge in tenths of the chord.
3. Last two digits describing maximum thickness of the airfoil as percent of the chord.[2]

For example, the NACA 2412 airfoil has a maximum camber of 2% located 40% (0.4 chords) from the leading edge with a maximum thickness of 12% of the chord.

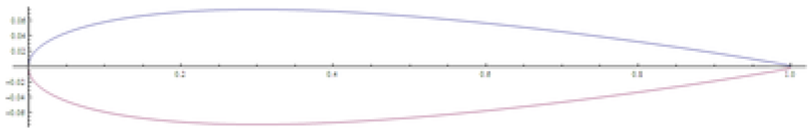
The NACA 0015 airfoil is symmetrical, the 00 indicating that it has no camber. The 15 indicates that the airfoil has a 15% thickness to chord length ratio: it is 15% as thick as it is long.



A: blue line = chord, green line = camber mean-line, B: leading-edge radius, C: *xy* coordinates for the profile geometry (chord = *x* axis; *y* axis line on that leading edge)

Equation for a symmetrical 4-digit NACA airfoil

The formula for the shape of a NACA 00xx foil, with "xx" being replaced by the percentage of thickness to chord, is[3]



Plot of a NACA 0015 foil generated from formula

$$y_t = 5t\left[0.2969\sqrt{x} - 0.1260x - 0.3516x^2 + 0.2843x^3 - 0.1015x^4\right],$$ [4][5]

where:

> *x* is the position along the chord from 0 to 1.00 (0 to 100%),
> y_t is the half thickness at a given value of *x* (centerline to surface),
> *t* is the maximum thickness as a fraction of the chord (so *t* gives the last two digits in the NACA 4-digit denomination divided by 100).

Note that in this equation, at $x/c = 1$ (the trailing edge of the airfoil), the thickness is not quite zero. If a zero-thickness trailing edge is required, for example for computational work, one of the coefficients should be modified such that they sum to zero. Modifying the last coefficient (i.e. to −0.1036) will result in the smallest change to the overall shape of the airfoil. The leading edge approximates a cylinder with a radius of

$$r = 1.1019\frac{t^2}{c}.$$ [6]

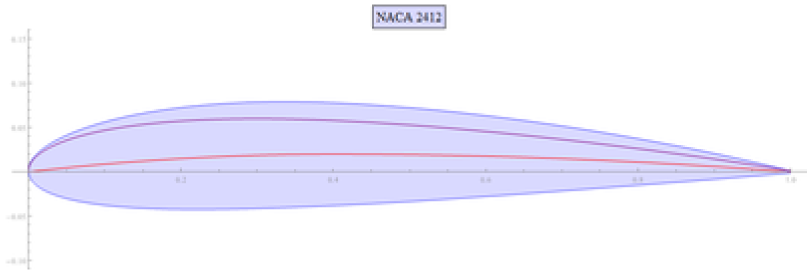
Now the coordinates (x_U, y_U) of the upper airfoil surface and (x_L, y_L) of the lower airfoil surface are

$$x_U = x_L = x, \quad y_U = +y_t, \quad y_L = -y_t.$$

Symmetrical 4-digit series airfoils by default have maximum thickness at 30% of the chord from the leading edge.

Equation for a cambered 4-digit NACA airfoil

The simplest asymmetric foils are the NACA 4-digit series foils, which use the same formula as that used to generate the 00xx symmetric foils, but with the line of mean camber bent. The formula used to calculate the mean camber line is[3]



Plot of a NACA 2412 foil. The camber line is shown in red, and the thickness – or the symmetrical airfoil 0012 – is shown in purple.

$$y_c = \begin{cases} \dfrac{m}{p^2}\left(2p\left(\dfrac{x}{c}\right) - \left(\dfrac{x}{c}\right)^2\right), & 0 \le x \le pc, \\ \dfrac{m}{(1-p)^2}\left((1-2p) + 2p\left(\dfrac{x}{c}\right) - \left(\dfrac{x}{c}\right)^2\right), & pc \le x \le c, \end{cases}$$

where

m is the maximum camber (100 *m* is the first of the four digits),
p is the location of maximum camber (10 *p* is the second digit in the NACA xxxx description).

For this cambered airfoil, because the thickness needs to be applied perpendicular to the camber line, the coordinates (x_U, y_U) and (x_L, y_L), of respectively the upper and lower airfoil surface, become[7]

$$\begin{aligned} x_U &= x - y_t\,\sin\theta, & y_U &= y_c + y_t\,\cos\theta, \\ x_L &= x + y_t\,\sin\theta, & y_L &= y_c - y_t\,\cos\theta, \end{aligned}$$

where

$$\theta = \arctan\frac{dy_c}{dx},$$

$$\frac{dy_c}{dx} = \begin{cases} \dfrac{2m}{p^2}\left(p - \dfrac{x}{c}\right), & 0 \le x \le pc, \\ \dfrac{2m}{(1-p)^2}\left(p - \dfrac{x}{c}\right), & pc \le x \le c. \end{cases}$$

Five-digit series

The NACA five-digit series describes more complex airfoil shapes.[8] Its format is LPSTT, where:

- L: a single digit representing the theoretical optimal lift coefficient at ideal angle of attack C_{LI} = 0.15 L (this is *not* the same as the lift coefficient C_L),
- P: a single digit for the *x* coordinate of the point of maximum camber (max. camber at *x* = 0.05 P),
- S: a single digit indicating whether the camber is simple (S = 0) or reflex (S = 1),
- TT: the maximum thickness in percent of chord, as in a four-digit NACA airfoil code.

For example, the NACA 23112 profile describes an airfoil with design lift coefficient of 0.3 (0.15 × 2), the point of maximum camber located at 15% chord (5 × 3), reflex camber (1), and maximum thickness of 12% of chord length (12).

The camber line is defined in two sections:[9]

$$y_c = \begin{cases} \frac{k_1}{6}\left(x^3 - 3mx^2 + m^2(3-m)x\right), & 0 < x < m, \\ \frac{k_1 m^3}{6}(1-x), & m < x < 1, \end{cases}$$

where the chordwise location x and the ordinate y have been normalized by the chord. The constant m is chosen so that the maximum camber occurs at $x = p$; for example, for the 230 camber line, $p = 0.3/2 = 0.15$ and $m = 0.2025$. Finally, constant k_1 is determined to give the desired lift coefficient. For a 230 camber-line profile (the first 3 numbers in the 5-digit series), $k_1 = 15.957$ is used.

Non-reflexed 3 digit camber lines

3-digit camber lines provide a very far forward location for the maximum camber.

The camber line is defined as[9]

$$y_c = \begin{cases} \frac{k}{6}\left(x^3 - 3rx^2 + r^2(3-r)x\right)^2, & 0 < x < r, \\ \frac{kr^3}{6}(1-x), & r < x < 1. \end{cases}$$

The following table presents the various camber-line profile coefficients:

Camber-line profile	p	m	k_1
210	0.05	0.0580	361.40
220	0.10	0.126	51.640
230	0.15	0.2025	15.957
240	0.20	0.290	6.643
250	0.25	0.391	3.230

Reflexed 3-digit camber lines

Camber lines such as 231 makes the negative trailing edge camber of the 230 series profile to be positively cambered. This results in a theoretical pitching moment of 0.

From $\frac{x}{c} \le r$

$$\frac{y}{c} = \frac{k_1}{6}\left[\left(\frac{x}{c} - r\right)^3 - \frac{k_2}{k_1}(1-r)^3\frac{x}{c} - r^3\frac{x}{c} + r^3\right].$$

From $r < \frac{x}{c} \le 1.0$

$$\frac{y}{c} = \frac{k_1}{6}\left[\frac{k_2}{k_1}\left(\frac{x}{c} - r\right)^3 - \frac{k_2}{k_1}(1-r)^3\frac{x}{c} - r^3\frac{x}{c} + r^3\right].$$

The following table presents the various camber-line profile coefficients:

Camber-line profile	p	m	k_1	k_2/k_1
221	0.10	0.130	51.990	0.000764
231	0.15	0.217	15.793	0.00677
241	0.20	0.318	6.520	0.0303
251	0.25	0.441	3.191	0.1355

Modifications

Four- and five-digit series airfoils can be modified with a two-digit code preceded by a hyphen in the following sequence:

1. One digit describing the roundness of the leading edge, with 0 being sharp, 6 being the same as the original airfoil, and larger values indicating a more rounded leading edge.
2. One digit describing the distance of maximum thickness from the leading edge in tenths of the chord.

For example, the NACA 1234-05 is a NACA 1234 airfoil with a sharp leading edge and maximum thickness 50% of the chord (0.5 chords) from the leading edge.

In addition, for a more precise description of the airfoil all numbers can be presented as decimals.

1-series

A new approach to airfoil design pioneered in the 1930s, in which the airfoil shape was mathematically derived from the desired lift characteristics. Prior to this, airfoil shapes were first created and then had their characteristics measured in a wind tunnel. The 1-series airfoils are described by five digits in the following sequence:

1. The number "1" indicating the series.
2. One digit describing the distance of the minimum-pressure area in tenths of chord.
3. A hyphen.
4. One digit describing the lift coefficient in tenths.
5. Two digits describing the maximum thickness in percent of chord.

For example, the NACA 16-123 airfoil has minimum pressure 60% of the chord back with a lift coefficient of 0.1 and maximum thickness of 23% of the chord.

6-series

An improvement over 1-series airfoils with emphasis on maximizing laminar flow. The airfoil is described using six digits in the following sequence:

1. The number "6" indicating the series.

2. One digit describing the distance of the minimum pressure area in tenths of the chord.
3. The subscript digit gives the range of lift coefficient in tenths above and below the design lift coefficient in which favorable pressure gradients exist on both surfaces.
4. A hyphen.
5. One digit describing the design lift coefficient in tenths.
6. Two digits describing the maximum thickness as percent of chord.

For example, the NACA 61_2-315 a=0.5 has the area of minimum pressure 10% of the chord back, maintains low drag 0.2 above and below the lift coefficient of 0.3, has a maximum thickness of 15% of the chord, and maintains laminar flow over 50% of the chord.

7-series

Further advancement in maximizing laminar flow achieved by separately identifying the low-pressure zones on upper and lower surfaces of the airfoil. The airfoil is described by seven digits in the following sequence:

1. The number "7" indicating the series.
2. One digit describing the distance of the minimum pressure area on the upper surface in tenths of the chord.
3. One digit describing the distance of the minimum pressure area on the lower surface in tenths of the chord.
4. One letter referring to a standard profile from the earlier NACA series.
5. One digit describing the lift coefficient in tenths.
6. Two digits describing the maximum thickness as percent of chord.
7. "a=" followed by a decimal number describing the fraction of chord over which laminar flow is maintained. a=1 is the default if no value is given.

For example, the NACA 712A315 has the area of minimum pressure 10% of the chord back on the upper surface and 20% of the chord back on the lower surface, uses the standard "A" profile, has a lift coefficient of 0.3, and has a maximum thickness of 15% of the chord.

8-series

Supercritical airfoils designed to independently maximize airflow above and below the wing. The numbering is identical to the 7-series airfoils except that the sequence begins with an "8" to identify the series.

See also

- NACA cowling
- NACA scoop

References

1. E. N. Jacobs, K. E. Ward, & R. M. Pinkerton. NACA Report No. 460, "The characteristics of 78 related airfoil sections from tests in the variable-density wind tunnel" (http://hdl.handle.net/2060/19930091108). NACA, 1933.
2. "Fundamentals of aerodynamics", John D. Anderson, Jr., third ed., ch. 4.

3. Moran, Jack (2003). *An introduction to theoretical and computational aerodynamics* (https://books.google.com/books?id=4eVP3yWZ1LgC). Dover. p. 7 (https://books.google.com/books?id=4eVP3yWZ1LgC&pg=PA7). ISBN 0-486-42879-6.
4. Aerospaceweb.org | Ask Us - NACA Airfoil Series (http://www.aerospaceweb.org/question/airfoils/q0041.shtml)
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6. Gordon J. Leishman. *Principles of Helicopter Aerodynamics*. p. 361.
7. Marzocca, Pier. "The NACA airfoil series" (http://people.clarkson.edu/~pmarzocc/AE429/The%20NACA%20airfoil%20series.pdf) (PDF). Clarkson University. Retrieved July 5, 2016.
8. E. N. Jacobs & R. M. Pinkerton 1936 Test in the variable-density wind tunnel of related airfoils having the maximum camber unusually far forward, NACA Report No. 537 (http://hdl.handle.net/2060/19930091610).
9. Abbott, Ira (1959). *Theory of Wing Sections: Including a Summary of Airfoil Data*. New York: Dover Publications. p. 115. ISBN 978-0486605869.

- Airfoils on Aerospaceweb.org (http://www.aerospaceweb.org/question/airfoils/q0041.shtml)

External links

2D Flow Aerofoil Sections (http://www.southampton.ac.uk/~jps7/Aircraft%20Design%20Resources/Sydney%20aerodynamics%20for%20students/) Source for NACA Java Java Applet Source Code for NACA 4 & 5-digit aerofoil generator (http://www.southampton.ac.uk/~jps7/Aircraft%20Design%20Resources/Sydney%20aerodynamics%20for%20students/naca45/naca/)

- David Lednicer's NACA airfoil coordinate generation program (http://www.ae.illinois.edu/m-selig/ads/naca.exe) Before running this Windows 95 executable, read this (http://www.ae.illinois.edu/m-selig/ads/nacaCrash.txt).
- John Dreese's NACA airfoil coordinate generation program (http://www.dreesecode.com/) Works on Windows XP, 7 and 8.
- NACA 4 & 5-digit Excel spreadsheets (https://groups.google.com/group/SketchUp3d/browse_thread/thread/68865aa2fc881e30/26a9ccea8c3c9af1#26a9ccea8c3c9af1)
- NACA Airfoil Series (http://www.aerospaceweb.org/question/airfoils/q0041.shtml)