WORLD WAR II FIGHTER *AERODYNAMICS*

BY DAVID LEDNICER
EAA 135815

Previously, we have explored the aerodynamics of modern homebuilt aircraft. Here, we will instead look at a different class of aircraft - World War II fighters. As time progresses, many of the valuable lessons learned in the original design of vintage aircraft are being lost. It is the purpose of this study to use modern aerodynamic analysis tools to recover some of this lost knowledge.

Great strides in aircraft design were made in the era of 1935-1945, and this is most evident in the design of fighter aircraft of this period. For this reason, an evaluation of three prominent fighter aircraft of this era, the North American P-51 Mustang, the Supermarine Spitfire and the Focke Wulf Fw 190 is presented here. As so much misinformation has appeared on these aircraft, references will be cited to support the data discussed here.

Wing Geometry

In a sense, these three aircraft types represent three stages within a single generation of fighter development. This can be most easily seen in the wing airfoils used on the aircraft. The Spitfire, designed in the mid 1930s, used the NACA 2200 series of airfoils, which was new at the time. The wing root airfoil is a NACA 2213, transitioning to a NACA 2209.4 at the tip rib. The Fw 190, which was designed at the end of the 1930s, used the NACA 23000 series of airfoils. The wing root airfoil is a NACA 23015.3 and the tip airfoil a NACA 23009. The P-51's wing, designed in the early 1940s, uses an early laminar flow airfoil which is a NACA/NAA hybrid called the 45-100. The wing root airfoil (of the basic trapezoidal wing, excluding the inboard leading edge extension) is 16% thick, while the airfoil at the tip rib is 11.4% thick. With the inboard leading edge extension, the wing root airfoil on the P-51B is 15.2% thick and on the P-51D 13.8% thick. The later model P-51H used a NACA 66,2-(1.8)15.5 a=.6 at the wing root and a NACA 66,2-(1.8)12 a=.6 at the tip and has no inboard leading edge extension.

It is interesting to note that approximately 2 degrees of washout was used on all three aircraft. However, the distribution of twist

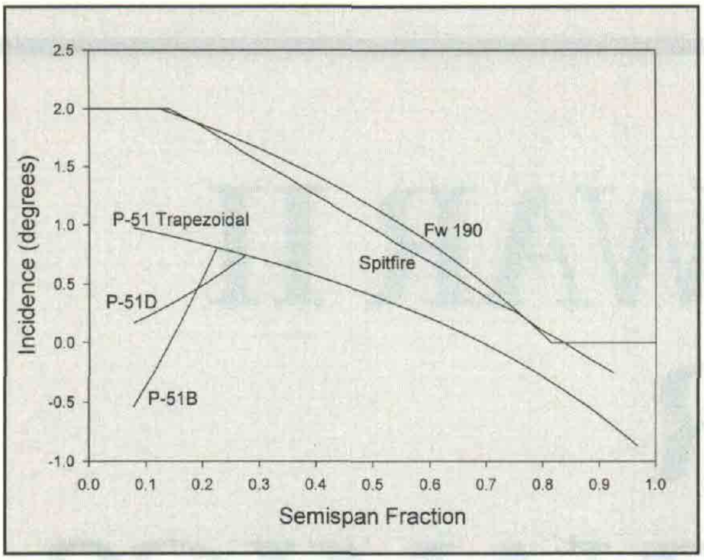


Figure 1- Wing twist distributions for the P-51B, P-51D, Spitfire and Fw 190.

varied for each aircraft. The Spitfire wing has a constant incidence (2 degrees) to the dihedral break, where the twist starts. This aircraft actually has 2.25 degrees of washout, distributed linearly (Fig. 1). The Fw 190 wing is unusual in that 2 degrees of washout exists between the root and a point at 81.5% semispan. Outboard of this location there is no more washout, the incidence holding fixed at zero degrees. The basic trapezoidal wing of the P-51B and P-51D has 2 degrees of washout, with the tip rib at -.85 degrees of incidence. However, addition of the drooped inboard leading edge extension modifies the appearance of the twist distribution. Lift distributions for the three aircraft show the results of these twist distributions (Fig. 2). These lift distributions were calculated, using VSAERO, with the aircraft trimmed at 360 kts and 15,000 feet of altitude to representative Gross Weights and CG locations.

The Spitfire wing is famous for having an elliptic planform. Indeed, the chord distribution is elliptical. An examination of the resulting circulation distribution for a trimmed condition mentioned above, shows that the loading distribution is not elliptical, though it is probably the most optimum of the three aircraft from the induced drag standpoint. The reason for deviation from elliptical is the 2 degrees of washout that have been added to the elliptical planform, which shifts the loading inboard. The elliptical wing planform appears to have been chosen primarily to provide greater wing depth in the inboard portion of the wing, while keeping the airfoil thickness-to-chord ratios low. This depth was necessary to house the outward retracting landing gear and wing gun ammunition boxes.

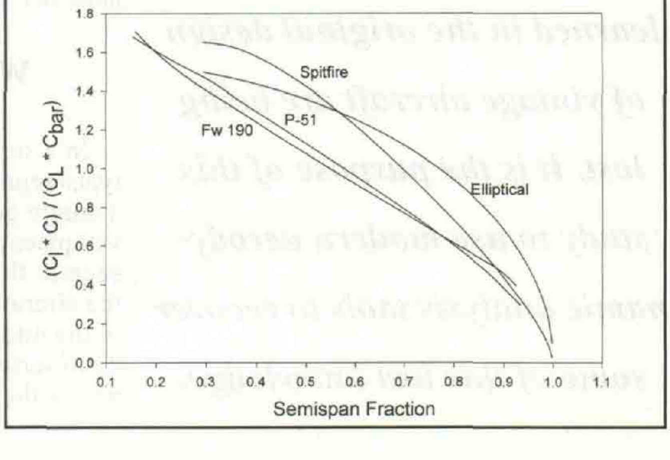
P-51 Mustang Analysis

The original North American Aviation drawing set for the Mustang are available from the National Air and Space Museum. A friend of mine living in England, Arthur Bentley, had obtained the set and was kind enough to sort through it for the drawings that were of relevance to my endeavor. It was found that models of the P-51B/C and P-51D/K were relatively easy to prepare, as the North American Aviation drawings contained surface coordinates, in a familiar Fuselage Station/Buttline/Waterline system. However, the NASM drawing set did not appear to contain the wing definition. After quite a bit of searching, I was put in touch with the Ed Horkey, who had been the Chief Aerodynamicist on the P-51 at North American. Ed was kind enough to supply the wing definition drawings for both the P-51B and P-51D.

The pressure distributions calculated by VSAERO on the P-51B and P-51D are shown in Fig. 3 and 4. Particularly noteworthy is the region of strong suction on the P-51D bubble canopy. This region is not present on the less bulged P-51B canopy. On both aircraft the suction region on the wing upper surface extends fairly far back on the wingís chord. This indicates that the wing should be capable of supporting a fairly large amount of laminar flow. The P-51 Mustang is renowned for being one the first aircraft to make use of airfoils designed to be capable of having extensive runs of laminar flow. Both the Spitfire and Fw 190 use airfoils that do not support substantial amounts of laminar flow. A two-dimensional cut through the wing pressure and skin friction distributions calculated by VSAERO on the Mustang (Fig. 5) show that, at a representative cruise condition, the wing was capable of sustaining long laminar boundary layer runs, with transition occurring at roughly 47% of chord. However, this calculation is for an ideal case, for a wing without fasteners, gaps, misalignments or surface waviness. During World War II, a Mustang was flight tested by NACA with a wake rake behind the wing at roughly 66.7% semispan (Ref. 1). The results of this test show that, in service the aircraft was unlikely to have a substantial laminar flow on the wing and transition occurred in the first 15% of the chord. Testing in an as-manufactured condition showed slightly lower drag and further, when the wing was refined to remove waviness and surface imperfections, a drag level was measured indicative of a substantial region of laminar flow. Wartime windtunnel tests of the Mustang's wing airfoil in Germany gave similar results (Ref. 2).

Early models of the P-51 experienced boundary layer separation in the radiator inlet duct. Pilots reported a rumbling noise emanating from the

Figure 2 - Calculated wing loading comparison with the aircraft trimmed at 360 kt and 15,000 feet altitude to representative gross weights and CG locations.



ductwork behind and beneath the cockpit on early model Mustangs. To investigate this phenomena, a complete Mustang fuselage was installed in a wind tunnel at the newly opened NACA Ames Research Center. It was found that the rumble was the result of the separated flow in the cooling inlet duct striking the radiator (Ref. 3). Changes, both in duct shape and the addition of a deep boundary layer splitter on the inlet eliminated the rumble and improved the aircraft's cooling. The results of these changes can be seen in the VSAERO boundary layer calculation, which shows that boundary layer on the upper surface of the cooling system does not separate until far back in the duct (Fig. 6). The boundary layer on the lower surface of the duct, starting fresh behind the oil cooler makes it to within inches of the water radiator and intercooler before separating. The losses in this system are much lower than that of the Spitfire. This efficient cooling system arrangement is credited with much of the Mustang's superior performance over the Spitfire.

Figure 3 - Pressure distribution calculated on the P-51B Mustang.

The Mustang has long had a reputation for being longitudinally unstable at aft CG locations resulting from the addition of a long-range fuel tank added behind the pilotís seat. Results of a wind tunnel test of a P-51B (Ref. 18) place the aircraftís power-off stick-fixed Neutral Point at 39.11% MAC, which agrees quite well with the VSAERO results, which places this point at 38.97% MAC. P-51Bs could be flown at CGs as far aft as 31.55% MAC (Ref. 4). Stick-fixed to stick-free effects and power effects account for roughly 7.5% MAC difference.

Supermarine Spitfire Analysis

Arthur Bentley also was able to supply me with the original Supermarine drawings for the Spitfire. The Spitfire drawing set contained definition for various models, ranging from the Spitfire I to the Seafire 47. It was decided to build the panel model to represent a Spitfire IX, which could be fully defined from the drawings. Coordinates were present on the drawings, but preparation of the fuselage proved to be difficult as a global coordinate system was not used. For instance, bulkheads could only be located by accumulating distances from a known reference, in a system more akin to that used in the design of ships.

The surface pressure distribution calculated for the Spitfire IX is shown in Fig. 7. Unlike the Mustang, the chordwise extent of suction on the wing upper surface can be seen to be relatively small, limiting the amount of laminar flow the wing can support. It is interesting that the greatest suction on the entire aircraft appears on the bulged canopy. Other strong suctions appear at the corners of the windshield, which was made up of panels of flat armor glass and had sharp corners.

One of the first things to come to light in the VSAERO analysis of the Spitfire is a region of separated flow at the base of the windscreen. The computation indicates that the boundary layer separates approximately 6 inches in front of the windscreen, due to the increasing pressure in this region (Fig. 8). The boundary layer traces that stop at separation have been restarted on the windshield at the point where the static pressure is the same as that at separation. Such a separation is not present on either of the other two aircraft reviewed here. However, this is a feature quite common on automobiles and is related to the slope of the windscreen. The Spitfire's windscreen is at a 35-degree angle to the forward deck, while the Fw 190's is at a 22-degree angle and the P-51's is at a 31-degree angle. Ev-

Figure 4 - Pressure distribution calculated on the P-51D Mustang.

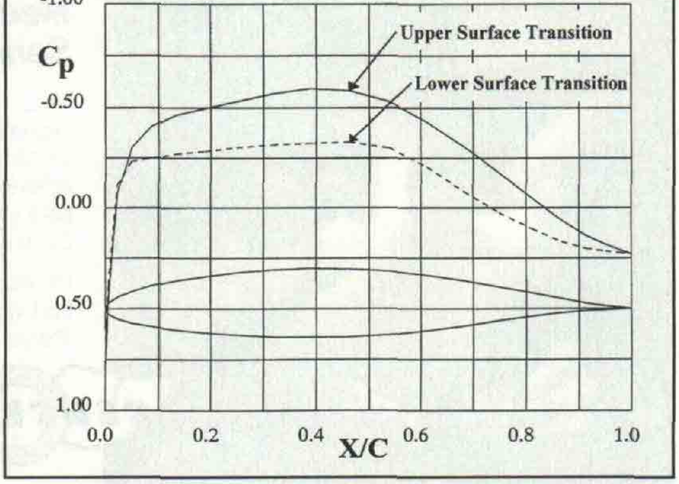


Figure 5 - Calculated Mustang wing airfoil pressure distribution and boundary layer transition locations in cruise, for ideal surface conditions.

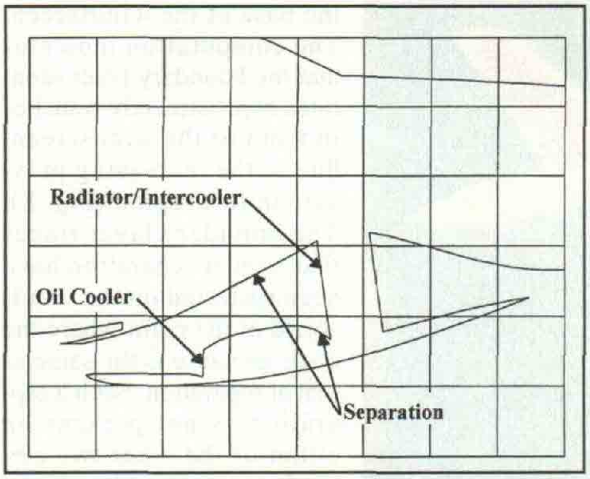


Figure 6 - Calculated boundary layer separation in the Mustang cooling system

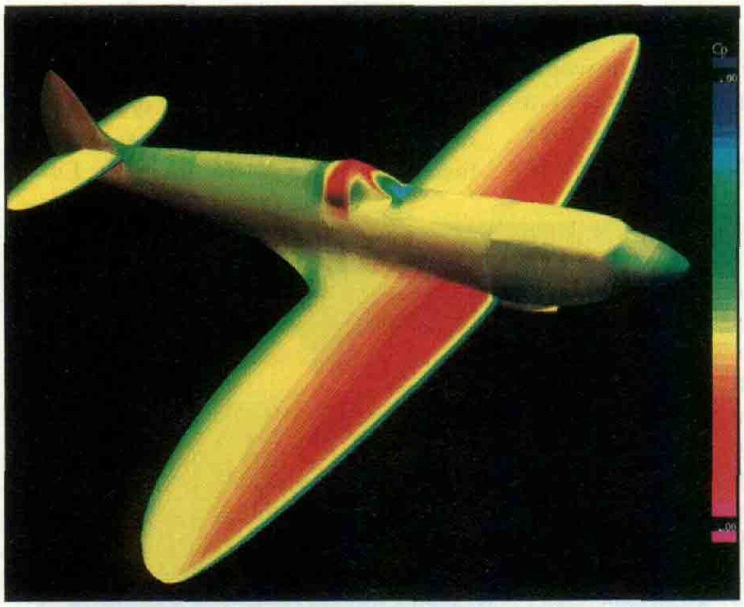


Figure 7 - Pressure distribution calculated on the Spitfire IX.

idently, the Spitfire's windscreen is too steep. An experimental windscreen, rounded and of shallower slope, was fitted to a Spitfire IX in 1943 produced a speed increase of 12 mph, at a Mach number of .79 (Ref. 5). A similar windscreen introduced on the Seafire XVII, is credited with a speed gain of 7 mph, at 400 mph (Ref. 6).

Supermarine is often regarded as being one of the first companies to make use of the breakthroughs made by Meredith at RAE Farnborough in the design of ducts for cooling systems (Ref. 7). In fact, the Spitfire's radiator ducts were designed using these guidelines. However, the VSAERO calculation indicates the boundary layer on the lower surface of the wing is ingested by the cooling system inlet. Running into the severe adverse (increasing) pressure gradient ahead of the radiator, the boundary layer separates shortly after entering the duct, resulting in a large drag penalty (Fig. 9). Experimentally, it was determined that the Spitfire cooling system drag, expressed as the ratio of equivalent cooling-drag power to total engine power, was considerably higher than that of other aircraft tested by the RAE. This was attributed to "the presence of a boundary layer ahead of the duct tends to precipitate separation and makes the ducting problem more difficult" (Ref. 8). Similar problems are present on the early model Messerschmitt Bf 109, up through the E model. A complete redesign of the cooling system, during development of the Bf 109F, resulted in the use of a boundary layer bypass duct, which significantly improved the pressure recovery at the radiator face (Ref. 9).

The Spitfire has long had a reputation of being longitudinally neutrally stable. Results of wartime flight tests of a Spitfire VA by NACA (Ref. 10) confirm that the aircraft was indeed longitudinally neutrally stable at a typical CG location. The NACA report mentions that no change in elevator position was necessary to maintain longitudinal trim when changing airspeed, implying that the CG was positioned at the location of the stick-fixed longitudinal Neutral Point. The CG location in this test was at 31.3% MAC. VSAERO analysis of the Spitfire places the power-off stick-fixed Neutral Point at 36.66%

MAC. Standard estimates of power effects show that the Neutral Point will shift forward 4-5% due to these effects, which accounts for the difference between the VSAERO and flight test results. The NACA testing also found there was a stable gradient of stick force with increasing airspeed. This means that the Spitfire was stick-free longitudinally stable. Bobweights in the elevator control circuit helped turn the stick-fixed neutrally stable airplane into an airplane with a small degree of stick-free stability. As the pilot mostly is aware of stick-free stability and low margins of stability are associated with high maneuverability, this was a satisfactory situation.

Focke Wulf Fw 190 Analysis

Arthur Bentley was once again the source of my geometrical information. In this case, several years ago he had prepared a set of Fw 190 drawings for a modeling magazine, working from the original Focke Wulf drawings. Initially, I first modeled a radial engined Fw 190 A-8, but I later modified this model to represent an inline engined Fw 190 D-9, in this case using actual Focke Wulf drawings. Despite sparse fuselage cross section information, this model was constructed with relative ease.

The pressure distribution calculated on the Fw 190 A-8 and Fw 190 D-9 are shown in Fig. 10 and 11. Here, like on the Spitfire, the chordwise extent of suction on the wing is limited by the choice of airfoils and not much laminar flow is supported. Also, as on the Spitfire, the bulged canopy of the Fw 190 D-9 has a region of strong suction, not present on the Fw 190 A-8.

At the time that the Fw 190 first appeared in combat, in 1941, it was superior to the contemporary fighters on nearly every count. When the RAF captured the first flyable Fw 190 in 1942, a thorough evaluation revealed the Achilles Heal to be a harsh stalling characteristic, which limited its maneuver margins. Captain Eric Brown states (Ref. 11):

The stalling speed of the Fw 190A-4 in clean configuration was 127 mph (204 km/h) and the stall came suddenly and virtually without warning, the port wing dropping so violently that the aircraft almost inverted itself. In fact, if the German fighter was pulled into a g stall in a tight turn, it would flick out into the opposite bank and an incipient spin was the inevitable outcome if the pilot did not have his wits about him. The stall in landing configuration was quite different, there being intense pre-stall buffeting before the starboard wing dropped comparatively gently at 102 mph (164 km/h).

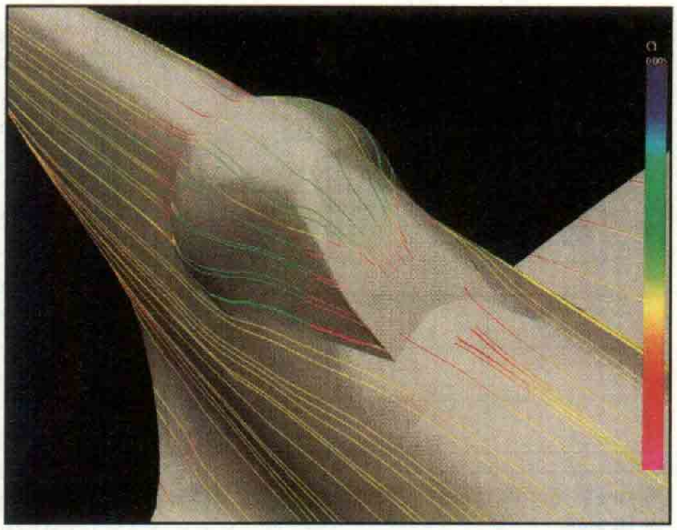


Figure 8 - Calculated Spitfire windshield boundary layer separation. Separation is calculated to take place at the base of the windshield where the streamline traces end. The location where the separated flow is estimated to reattach higher up the windshield is shown by where the streamline traces resume.

The results of an USAAF evaluation of the Fw 190 (Ref. 12 and 13) report the aircraft to have a gentle stall. However, these reports admit that the Fw 190 stalled abruptly when maneuvering. The reason for this reported difference in non-maneuvering stall behavior is unknown. A comparison of the local wing lift coefficients, calculated by VSAERO, at stall with the estimated stalling lift coefficients of the airfoils two-dimensionally (Fig. 12) shows that approximately the inner 40% of the wing reaches C_{lmax} at the same aircraft angle of attack. A wartime Focke Wulf report (Ref. 14) indicates that at higher loading conditions (i.e., when pulling more gs) elastic deformation of the Fw 190 outer wing shifts the load distribution outboard. This would cause even more of the wing to reach its stalling lift coefficient simultaneously. Combined with the sharp stalling features of the NACA 230XX airfoils, this would produce the harsh stall found in by Capt. Brown. A gentle stall would be evidenced by a more gradual progression of the 2D stall spanwise.

Initial VSAERO calculations were made on a model of the Fw 190 A-8. This version of the aircraft was powered by a BMW 801D radial. Naturally, the question arose as to how the aerodynamics of this aircraft differed from the later, Junkers Jumo 213A powered Fw 190 D-9. The Jumo engine, an inline, is much longer than the BMW engine, giving the D-9 a elongated nose, which was counter balanced with a 500mm plug added to the aft fuselage. The VSAERO model was modified to represent a D-9 by making these changes and by adding the bulged canopy found on Fw 190 D-9s. It was found from the VSAERO results that the fuselage stretch designed by the Focke Wulf engineers resulted in a slight increase in stick fixed stability, with the Neutral Point moving from 35.8% MAC on the A-8 to 40.4% MAC on the D-9. It should be noted these results do not contain propeller effects, which were not modeled. Flight testing of an early model Fw 190A indicated that the aircraft was "just statically stable; stick fixed and free, engine off; and statically unstable to a slight degree, engine on" (Ref. 11). During the continued development of the Fw 190 series, the aircraft's CG moved rearward as fuel tanks and other equipment was added to the aft fuselage (Ref. 15). This Neutral Point shift during development of the Fw 190D model would have been quite valuable in maintaining the continued growth of the design.

Drag Comparison

There are many conflicting claims as to the equivalent flat plate drag area (f) of these fighter aircraft. Based upon my research, what I believe are

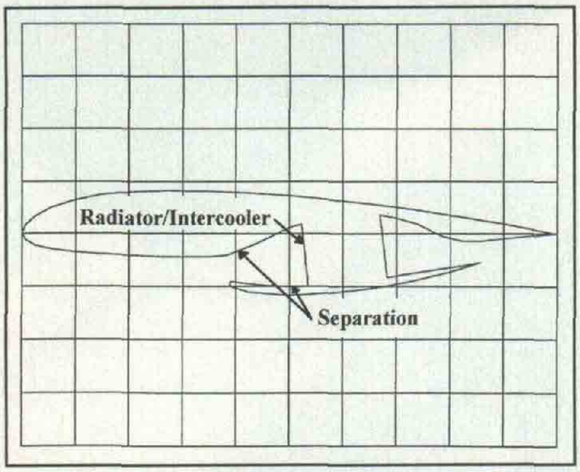


Figure 9 - Fw 190 calculated lift coefficient distribution at 1g stall.

Figure 10 - Pressure distribution calculated on the Fw 190 A-8.

the most accurate values are shown in Table 1.

The wetted areas of the aircraft are calculated by VSAERO, and exclude the ducts for cooling systems.

Notable is that the Mustang has the largest wetted area of this group of aircraft, but has the lowest drag. Evidence of this is that with the same version of the Rolls-Royce Merlin and propeller installed, the Mustang X was measured to be 23 mph faster than the Spitfire IX (Ref. 16). The Mustang X was an Allison powered Mustang reengined by Rolls-Royce with a Merlin 65. The P-51B, with an improved cooling system configuration is even faster than the Spitfire IX. The difference in performance between the Mustang and the Spitfire is attributed to several factors. These include the superior configuration of the Mustang's cooling system and the Spitfire's relatively high level of excrescence drag, generated by open wheel wells, a nonretractable tail wheel and other design details (Ref. 17-19).

One popular piece of aerodynamic folklore is the low CDswet value achieved with the Mustang. Various sources quote this value as ranging from .0038 to .0043. A review of available wind tunnel and flight test drag data for the Mustang demonstrates the need for having all details of the aircraft present if the drag is to be accurately measured. Subscale wind tunnel tests of the P-51A and P-51B resulted in values of C_{Dswet}, at a representative cruise lift coefficient, in the range of .0046-.0047 (Ref. 20-22). However, these tests usually were of models lacking exhaust stacks, surface discontinuities, etc. Measurements made in full-scale wind tunnel tests of the P-51B (Ref. 23) and flight tests of the P-51A (Ref. 24) and P-51B (Ref. 21) resulted in a value of C_{Dswet} of approximately .0053.

Conclusion

Important design features of three prominent World War II fighter aircraft have been examined by the use of a modern Computational Fluid Dynamics method. It is hoped that the results presented here will help demonstrate some of the valuable lessons learned from an important era in fighter aircraft design. This information, while historical, still has relevance in today's world of aircraft design. Important lessons to be learned are:

• Airfoil choice and surface quality are important in achieving the advantages of laminar flow.

• Cooling system duct design for liquid cooled engines must be conducted carefully to avoid losses.

• Attention to aerodynamic detail, such as windshield slope, can overcome the disadvantage of excess wetted area.

• An abrupt stall can be avoided if attention is paid to airfoil selection and wing twist.

• As seen with all three of these airplanes, longitudinal stability and control problems are common, but can be avoided by the resourceful designer.

Author's Note

This article is dedicated to Edward Horkey and Jeffery Ethell, who both contributed information vital to this work. Ed died as a result of injuries sustained in traffic accident in July 1996. Jeff was killed in the crash of a Lockheed P-38 Lightning in May 1997.

Far too young to have participated in World War II, I have long been fascinated about finding out how the famous aircraft of this war were designed. The deeper I have gotten into this pursuit, the more information I have uncovered that has proven to be valuable in my daily work as an aerodynamicist. I have become convinced that a study of ihistorical aerodynamicsî is an important part of an aerodynamicistís ongoing edu-

TABLE 1

Aircraft	f	Wetted Area	C_{Dswet}	Ref.
Spitfire IX	5.40 ft^2	831.2 ft2	.0065	16
P-51B Mustang	4.61 ft^2	874.0 ft2	.0053	21
P-51D Mustang	4.65 ft^2	882.2 ft2	.0053	27
Fw 190 A-8	5.22 ft^2	735.0 ft2	.0071	26
Fw 190 D-9	4.77 ft^2	761.6 ft2	.0063	26

cation. To this end, one of my goals has become to try and disseminate the knowledge I have unearthed, this article being an effort towards this end. For those seeking further information in this regard, I recommend taking a look at my iIncomplete Guide to Airfoil Usageî at: **http://amber.aae.uiuc.edu/~m selig/ads/aircraft.html.**

As mentioned in previous articles, I am an aeronautical engineer, specializing in applied computational fluid dynamics. Based in Redmond, Washington, I work for Analytical Methods, Inc. My aerodynamic (and hydrodynamic) consulting projects at AMI have included submarines, surface vessels, automobiles, trains, helicopters, aircraft and space launch vehicles. I can be reached at: dave@amiwest.com or:

Analytical Methods, Inc., 2133 152nd Ave NE, Redmond, WA 98052

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Figure 11 Pressure distribution calculated on the Fw 190 D-9.

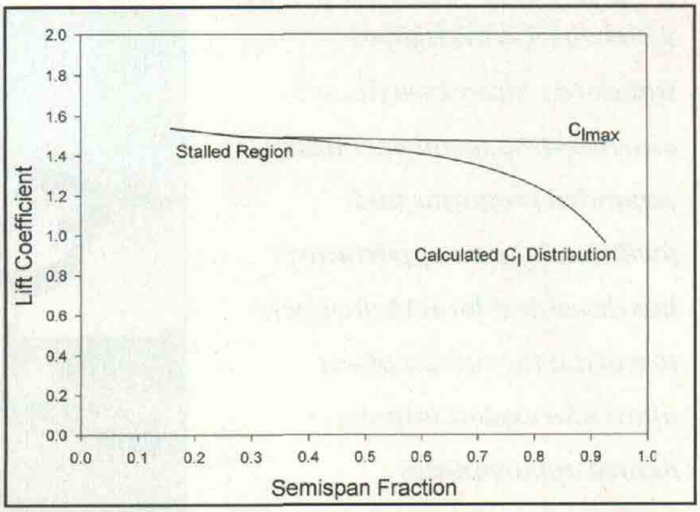

Figure 12 - Boundary layer separation calculated in the Spitfire cooling system.