The Aerodynamics of the Spitfire

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Abstract

This paper is a sequel to earlier publications in this Journal which suggested a possible origin for the Spitfire's wing planform. Here, new material provided by Collar's drag comparison between the Spitfire and the Hurricane is described and rather more details are given on the Spitfire's high subsonic Mach number performance. The paper also attempts to bring together other existing material so as to provide a more extensive picture of the Spitfire's aerodynamics.

1. Introduction

To celebrate the Spitfire's eightieth year, the Royal Aeronautical Society's Historical Group organised the Spitfire Seminar held at Hamilton Place, London, on 19th September 2016. This celebration offered the author the opportunity to talk about the aerodynamics of the Spitfire. Part of that lecture was based on material already published [1, 2] in this Journal which suggested that the semi-elliptical wing planform adopted could have come from earlier publications by Prandtl. Section 2 offers further thoughts on this planform choice, in particular how this interlinked with the structural decision to place the single wing spar at the quarter-chord point and the selection of the aerofoil section. Since the publication of References 1 and 2, new material has come to light in the form of an investigation by A. R. Collar in 1940 which compares the drags experienced by the Spitfire and the Hurricane. This investigation, described in Section 3, indicates that a significant part of the Spitfire's lower drag can be attributed to the better boundary-layer behaviour produced by its wing's thinner aerofoil section. Section 4 deals with such matters as surface finish, including the well-known split-pea flight tests, propulsion effects on aerodynamic performance and further data on the turning radius problem. The paper closes with more detailed information on the Spitfire's performance at high subsonic Mach numbers. This shows an aerodynamically cleaner version to be slightly superior in this respect to the aircraft which replaced it, the Mk IV Gloster Meteor. Suggestions are offered as to why this was so.

As a result of the Spitfire Seminar, the author has received two contributions containing information which has here been incorporated at appropriate points. The contributors are Ralph Pegram, the author of two books on Supermarine aircraft [3, 4], and Michael Salisbury, from 1947 an aerodynamicist at Supermarine, later Chief Aerodynamicist there and subsequently on the TSR2 project.

2. Structural and Other Considerations Influencing the Spitfire Wing's Aerodynamic Design Evolution

The explanation of aerofoil behaviour began to emerge in the early years of the twentieth century. In particular, Zhukovskii's aerofoil theory [5] of 1910, based on shapes having blunt noses and streamlined tails, was found to produce good agreement when subjected to, as Galileo put it, the ordeal of experiment. Wind-tunnel tests by Betz [6] in 1915, for example, confirmed this both for the pressure variation around a Zhukovskii aerofoil and for its lift force's steady increase with incidence. A further correct feature of this theory is its prediction of an aerofoil's centre-of-pressure behaviour with change of incidence. For symmetric aerofoils this point at which the lift force acts is found throughout the incidence range below stall to be fixed at the quarter-chord point aft of the leading edge. However, for cambered aerofoils the centre of pressure moves steadily forward from the aerofoil's rear as incidence increases, the lift force increasing as it does so. Yet this movement occurs in such a manner that the moment of the lift force about the quarter-chord point does not change with incidence. As argued in Reference 1, this lift loading can be replaced by one in which the lift is fixed at the quarter-chord point, the lift increasing with incidence, and a pure torque which, in contrast, does not change with incidence. The latter behaviour provides an attractive feature for structural design.

These characteristics of the Zhukovskii aerofoil are fairly common to many of the aerofoil sections designed in the first thirty or so years of powered flight. A further feature is that many of these aerofoils have their maximum thicknesses around one-third chord aft of the leading edge.

During the biplane era, the wing structure often favoured involved the use of two spars. Their locations can be detected by studying the placement of the interplane struts. Occasionally, however, a single spar, in effect, was used. One early example is provided by the Fokker Dr. 1 Triplane of 1917. In this case, the single spar within each of the three wings can again be detected by the placement of the outboard interplane struts running through the wings. In side view, for the two lower wings this placement is seen to lie at one-third chord aft of the leading edge. Presumably this location was chosen because the aerofoil section is thickest at this point and therefore provides maximum spar depth. However, in his presentation to the Spitfire Seminar, this author mistakenly claimed the quarter-chord point as the spar's position. This error arose from a concentration on the side view of the upper wing alone, in which the aileron projects aft of the wing's trailing edge so as to make the chord appear greater than it is. For this error the author apologizes. As to the Triplane's spar itself, this is a hollow wooden rectangular box centred around one-third chord. It is partly the spar's twin webs but mainly the top and bottom flanges – booms in aeronautical parlance - which resist the lift load's direct bending effect whilst the complete box resists the twisting moment created by that load, the moment in this case varying with incidence.

So Mitchell, in his Spitfire design, was not the first to adopt the structural weight-saving measure of a single wing spar. As to resisting the moment created by the lift load, Mitchell accomplished this through the combination of the wing's nose skin and the spar web to form a D-nosed torsion box. In his contribution to Reference 7, Alan Clifton, Supermarine's Head of the Technical Office in the 1930s, points out that Mitchell had introduced this combination of a single spar and nose torsion box in his design for a six-engine flying boat in early 1930. Pegram [(8)] adds that a patent later that year covered the torsion box's use as the steam condenser for evaporatively cooled engines. For such an engine the water coolant emerged from the cylinder block as steam to be cooled back to the water phase as indicated, thus avoiding the need for a drag-producing external radiator. Although the flying boat itself was cancelled before completion, a one-twenty-fourth scale wing was built for testing in the Duplex wind tunnel at the Aerodynamics Department of the National Physical Laboratory (NPL). Due to the aircraft's cancellation, the testing programme itself was curtailed but the early results, for the wing's pressure distributions over a range of incidences, were deemed to be of general interest and were published in 1934 by Cowley and McMillan [(9)]. One aim of these tests was to assess the effect of aileron deflection on the wing's pressure distribution and thus calculate the wing torsional loads created.

A single spar combined with the nose torsion box also appeared in Mitchell's fighter design, the disappointing Supermarine Type 224 of 1934. This was powered by a Rolls-Royce Goshawk engine which employed evaporative cooling. The fighter was built to the Air Ministry's Specification F7/30 and became the starting point for the evolution of the Spitfire design. In the former case, the single-web spar is located at one-third chord from the leading edge [(10)], presumably on the grounds here again that this location provides maximum spar depth. The single spar and torsion-box combination was carried over to the Spitfire design although, as will emerge, the single spar's location moved to the quarter-chord point presumably so as to benefit from the torsional simplification mentioned earlier. It should be added that the use of metal skinning enabled the skin itself to carry some of the bending and torsional loads. The skin thickness adds to the spar's top and bottom boom areas in resisting the direct bending stresses and provides a further torsion box aft of the spar which aids in resisting torsion; hence the term 'stressed-skin construction'.

Aside from the Type 224's problems associated with its evaporative cooling system, which created difficulties when the aircraft was flown inverted, the other main problem lay in the aircraft's high drag. Obvious culprits were the fixed, trousered undercarriage and open cockpit. These were dealt with, together with the removal of the wing's anhedral inboard portion, in the re-design which began in the late spring of 1934. The result is shown in Figure 1 taken from Clifton's contribution to Reference 7. Moreover, two further members of Mitchell's design team, Ernest Mansbridge (see Reference 10) and Beverley Shenstone (see References 11 and 12), are both on record as suspecting the aircraft's thick aerofoil sections as significant contributors to this high drag. As Mansbridge [(10)] put it,

> "We were a bit over-cautious with the wing and made it thicker than it need have been. We were still very concerned about possible flutter, having encountered that with the S.4 seaplane."

The Type 224's aerofoil section for the inboard anhedral portion was a symmetric section of 18% thickness/chord (t/c) ratio (NACA 0018) [(10)]. For the dihedral outboard portion an RAF 34 section (12.5% t/c ratio) was modified to a t/c ratio of 18% to match the anhedral NACA 0018 inboard section and tapering linearly to 9.6% at the tip (Pegram [(8)]). Thus the Type 224's t/c distribution was similar to that of the Hurricane (see Table 1 below). A twist, or washout, of

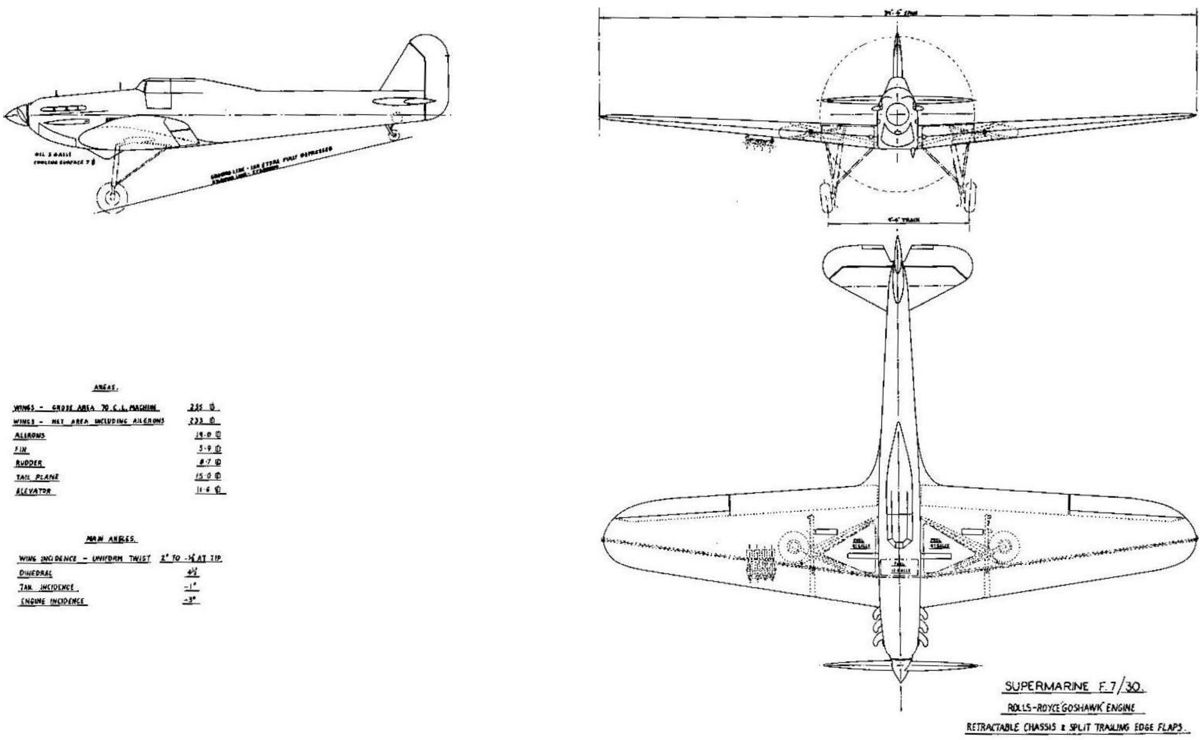

Figure 1 Supermarine F7/30, Drawing No. 30000 Sheet 2. Clifton's contribution to Reference 7

2.5°to improve stalling characteristics was used and this was carried over to the Spitfire design [8, 10]. Previously, the Supermarine team had used thinner sections for the Schneider Trophy floatplanes listed in Table 1. Indeed, for the Stranraer biplane flying boat of 1934 the aerofoil used was a similarly thin NACA section [8]; the Supermarine team were here remarkably quick off the mark in adopting these new aerofoil sections. However, in all these cases the wings were externally-braced. In the case of the monoplanes, as Mansbridge's above comment indicates, this measure was adopted following the accident to the unbraced S.4 Schneider Trophy contender of 1925. The Type 224's thicker wing, in contrast, required no external bracing.

Table 1 Thickness/Chord Ratios (%)

Supermarine S. 5 (RAF 30 symmetric section) [8]	12.6
Supermarine S. 6 (RAF 27 symmetric section) [8]	9.8
Supermarine Stranraer (NACA 2409 cambered section) [8]	9.0
Hawker Hurricane (Clark YH cambered section)	18 (root) – 12 (tip)
Bf 109 (Resembles NACA 23 series cambered section) [13]	14.8 (root) – 10.5 (tip)
Spitfire (NACA 22 series cambered section) [10]	13 (root) – 6 (tip)

So during the second half of 1934, one aim of the Supermarine team became the reduction of the Type 224's t/c values towards those used on their earlier aircraft. Roger Dickson's documents in the Davies archive (14), Pergram (8) reports, show that this work was initiated on 1st May 1934 in an instruction issued by Clifton to Dickson. Amongst the questions asked was "How much can the wing area be reduced using a higher lift section, preferably one we know, say NACA 2412?" Reference 10 quotes a report by Shenstone, dated May 1934 and after his visit to the United States, in which he comments favourably on the use there of NACA four-digit sections from the 22 and 24 series. The data from the NACA's aerofoil testing programme at high Reynolds numbers in its variable-density wind tunnel had begun to emerge in 1930. The data for the 24 series appeared in a technical note (15) dated January 1932 and by December of that year a comprehensive NACA Report (16) brought together the data for a large selection of such four-digit sections. The aerofoil sections finally selected for the Spitfire were drawn from these NACA designs, in this case the NACA 22 series, 2213 at the root, 2206 at the tip: 2% camber located at 20% chord, t/c 13% at the root, 6% at the tip (10). In his contribution to References 11 and 12, Shenstone merely states that the NACA 22 series "was just right". Perhaps the preference for the NACA 22 series lay in the fact that the aerodynamic moment about the quarter-chord point of that series is roughly half that of the NACA 24 series (16). The point here, as revealed by the original NACA data of Reference 16, is that it is the camber value and the shape of the camber line which largely determine the value of the section's aerodynamic moment. Crudely speaking, moving the point of maximum camber forward from 40% chord to 20% chord whilst retaining the same camber value reduces the length of the upper surface suction peak immediately aft of the leading edge, thereby reducing the moment. This reduction would be beneficial for the torque on the D-nosed torsion box.

As to planform shape, by the early autumn of 1934 straight taper was still being used, as shown in the Supermarine drawing presented by Joseph Smith, Mitchell's eventual successor as Chief Designer, in his lecture (17) to this Society in 1947. This was reproduced in Reference 1 and is here repeated as Figure 2 for ease of reference. The single spar is clearly

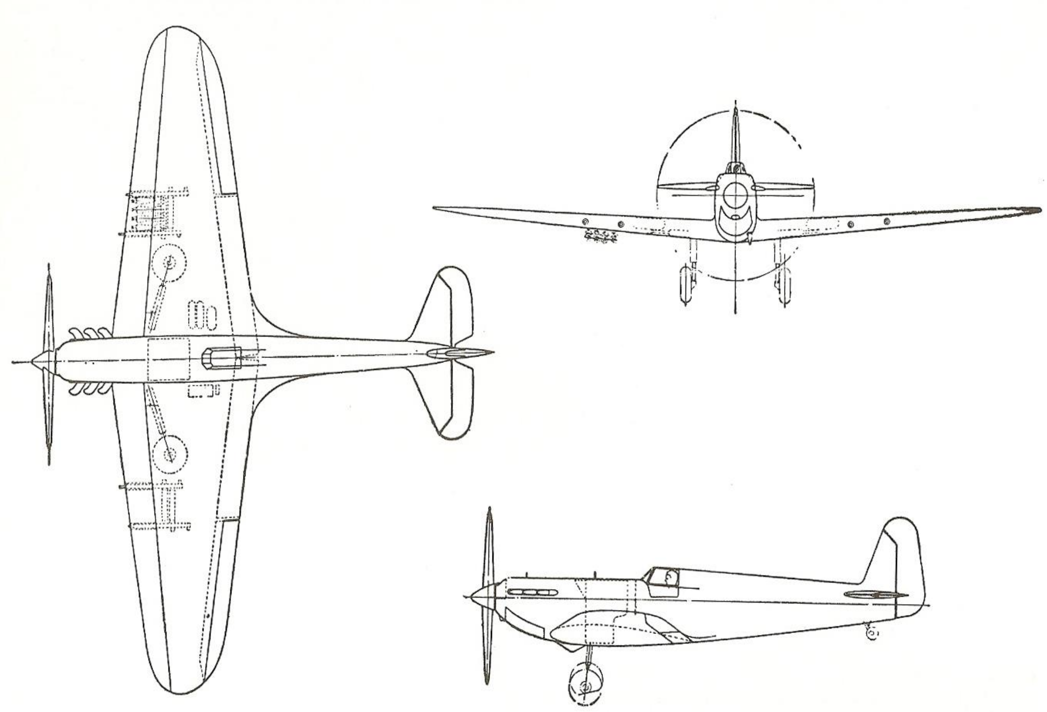

Figure 2 Supermarine F7/30 development, autumn 1934. Smith (17)

seen to be now at the quarter-chord point, although in plan view the spar is, like that of Figure 1, still angled slightly backwards. Yet the wing has now acquired a broader chord at the root, suggesting that the t/c value is lower than that of Type 224. Straight taper is also used for the tail surfaces. However, the fuselage is now closer to that of the Spitfire, the Type 224's short fairing aft of the cockpit, still evident in Figure 1, having been discarded in favour of a perhaps less drag-prone continuation of the cockpit's upper contour to the tail in Figure 2. The engine is still the steam-cooled Goshawk.

Planform shape changed dramatically around the close of 1934 with the adoption of the semi-elliptical wing. This particular combination of two semi-ellipses wedded at a common major axis resulted in the spar's quarter-chord location now lying in plan view perpendicular to the aircraft's fore-and-aft axis. This produced a number of structural and other benefits. In the layouts of Figures 1 and 2 the overall lift load on the spar would act to the rear of the spar's fuselage joint, creating a twisting moment there which would require a strengthened and heavier joint; this was eliminated in the straight-spar layout. Reference 10 recounts the comments of E. J. Davis concerning the difficulties of manufacturing the spar-rib arrangement had these components not been at right angles to each other; these too were eliminated in the new layout. A further factor, as Price [11] points out, is that around the time of the change to the semi-elliptical wing the decision was taken to switch from the Goshawk engine to the new, more powerful Rolls-Royce PV12 (Merlin) engine, as yet, like the Goshawk, still steam-cooled. However, the PV12 engine was heavier, forcing a forward movement of the aircraft's centre of gravity. A compensatory forward shift of the centre of lift was accomplished by straightening the spar.

Other practical benefits followed from the adoption of the semi-elliptical wing planform. Its broad root chord, continued outboard with initially only slight reduction, provided a wing thick enough to accommodate the necessary spar depth, the undercarriage wells, the radiator (once the switch to water/glycol cooling was adopted) and the outboard placement of the guns whilst keeping the latter in line with the aircraft's centre of gravity. Yet this broad chord, whilst allowing all this, also provided a wing thinner, in terms of the t/c ratios of Table 1, than any of its contemporaries. As will be seen in Section 3 below, this provided the significant aerodynamic benefit of reduced boundary-layer drag.

Also this wing planform offered the further aerodynamic advantage of minimum, or near-minimum, induced drag. However, to reiterate the point made in Reference 1, one should not become too transfixed by this feature since the advantage is often slight. The induced drag coefficient, C_{Di}, is given by [18]

$$C_{Di} = k\, C_L^{2} /(\pi A), \tag{1}$$

where C_L is the lift coefficient and A the wing aspect ratio. The factor k has its minimum value of unity for elliptic wings alone (excluding the fuselage and tail) of constant aerofoil section camber and no twist. However, Glauert [18] shows that for wings alone having straight tapers of the ratios under discussion the value of k increases to at most 1.03. His calculations for the effect of twist, when adjusted to the 2.5° value of the Spitfire, suggest a value of k at most about 1.05. Far more significant are the effects of the fuselage and tail in the determination of k for the complete aircraft. Both Anderson [19] and Loftin [20] provide suggested ranges for k for complete aircraft, although they use an aerodynamic efficiency factor which is the reciprocal of k. Thus their suggested k values range from 1.05 – 1.18 (Anderson [19]) and 1.33 – 1.4 (Loftin [20]) although in the latter case older, less aerodynamically efficient aircraft are often of main interest. However, Salisbury [21] recalls that when he joined Supermarine in 1947 the value of k for the Spitfire's successor, the Spiteful, was taken to be 1.15, the increase above near-unity being largely due to the fuselage. This value might have been obtained from experience with the Spitfire but he has no certain knowledge of this.

It is interesting to note that, together with the adoption of the semi-elliptical wing, the Spitfire's tail surfaces also changed from the straight taper of Figures 1 and 2 to elliptic shapes. These surfaces, when experiencing lifting forces (or side forces for the fin), will create trailing vortices and hence produce additional induced drag contributions which add to the value of k. These are likely to have a small effect so that it is surprising that the Supermarine team appear to have gone to the trouble of trying to minimize them. On the other hand, the Spitfire design is a prime example of close attention being paid to fine detail; small percentage improvements add up. A further example of this is contained in Jeffrey Quill's contribution to Reference 7 in which he states that the Spitfire "had an unusually thin tailplane of 9% t/c ratio". To this Pegram [8] adds the interesting point that the section is so thin here, and for the wing tip, that Mitchell patented a method for attaching the skin to the ribs as it had proved impossible to insert conventional riveting tools into the narrow interior. This illustrates Mitchell's keenness to achieve small t/c ratios.

Yet the fact – some might call it this author's obsession – remains that the Spitfire's semi-elliptical wing planform is geometrically identical to the planform published by Prandtl as early as 1918. Reference 1 provided a ***suggestion*** as to how that might have come about, a suggestion backed by no hard evidence but merely by what might be called the circumstantial evidence given in that reference and in Reference 2. Against this can be set the recollections [10] of R. J. Fenner of the design office:

> "When we came to the final version with the Merlin engine RJM (Mitchell) had fixed the smaller wing area, lower thickness/chord ratio and the optimum single spar position and I had the job of producing the lay-out to meet his proposals….. The planform of the wing was, originally, perfectly elliptical and then bent forward along its major axis until the optimum spar position was straight. I remember clearly making several drawings of alternative planform and RJM, in his rounds of the drawing boards, selecting the scheme as described except for minor changes to the wing tips."

That being so, it is nonetheless remarkable that the final shape is so precisely that of Prandtl's design, not only in the 1-to-2 ratio of the ellipses' semi-minor axes but also in the 4.1 value for the ratio of the span (excluding the fuselage) to the root chord. Perhaps both approaches were used, one confirming the other to create the desired optimum spar position along a straightened quarter-chord line.

In Reference 1 it was suggested that Shenstone played a significant role in a number of these key decisions made during the Spitfire's aerodynamic design evolution. In commenting on

Shenstone's involvement, Pegram [8] suggests the apt term for him as Mitchell's 'truffle-hound', seeking out the new ideas in aerodynamics which could be fruitfully incorporated in that evolution. For example, Shenstone's contributions to References 11 and 12 imply that he had tried to interest Mitchell in some form of elliptic planform. David Faddy, in an article [22] on his father, Alfred Faddy, another key member of Mitchell's team, suggests that it was Alfred Faddy who had persuaded Mitchell to take seriously Shenstone's suggestion in this respect. As to the choice of the NACA 22 series aerofoil section, Shenstone again had a hand in this as described above. He also pressed for the smooth surface finish [11, 12] discussed in Section 4.1 below. The appearance of wing root fillets in the Spitfire's design development during the spring of 1935 would also, as suggested in Reference1, have been favoured by Shenstone after his visit to the United States. There these features had been adopted by, for example, the Northrop and Douglas companies. However, it appears that it was not he who instigated this. At the time of the Spitfire's design, the published data on fillets came from Muttray [23] in Germany and Klein [24] working with von Kármán in California. Both favoured fairly extensive fillets which begin at the wing leading edge. In contrast, the Bellanca 28-70 racing monoplane of 1934 used a reduced fillet which begins near the wing's maximum thickness and thereafter flares towards the trailing edge. Reference 10 recounts that in October 1934 Mitchell had been made aware of the Bellanca's fillet and had been shown associated wind-tunnel data. Flight tests of the Type 224 with such fillets added occurred in February 1935, wool tufts being used to determine flow patterns at the fillets. Shenstone's photograph of it, plus wool tufts, is reproduced in References 10 and 12. Similar fillets were used on the Spitfire prototype, K5054, and flight tests to investigate them occurred in late March 1936 following its first flight near the beginning of that month. In this case oil was used to indicate flow direction and thereafter no shape changes seem to have been deemed necessary.

As to the need for wing root fillets, this arises from the fact that a wing root's pressure distribution does not match that around the fuselage. This mismatch can create local boundary-layer separation which results in increased drag and an irregular turbulent wake which can bombard the tailplane and elevator to detrimental effect. According to Reference 10, it was this tailplane buffeting on the Type 224 which Mitchell discussed with the Bellanca's pilot, visiting Supermarine at the time, and it was this conversation which led to the testing of similar fillets on the Type 224.

Whatever the background to the selection of the Spitfire's wing planform may have been, it is interesting to note that variants of this planform were adopted for later fighter aircraft. Examples are the Hawker Tempest and Fury, the Fairey Firefly and the Republic P-47 Thunderbolt, although straighter or even straight leading edges were used presumably for ease of manufacture. However, when it came to Supermarine's replacement for the Spitfire, the Spiteful of 1944 (Specification F1/43), this planform was discarded in favour of a straight-tapered wing incorporating a so-called laminar flow aerofoil section. Here the emphasis was on reducing the boundary-layer skin friction drag by maintaining larger regions of laminar boundary-layer flow over the wing surfaces. The efficacy of this approach to drag reduction depended on high manufacturing accuracy and the avoidance of the smallest surface irregularities. However, since much of the inboard wing flow would be contaminated by turbulence from the propeller wash, it is doubtful that this choice would have provided much benefit. Quill [25], for example, comments on its disappointing performance improvements compared with the later marks of Spitfire. However, the Spiteful's wing was adopted for the jet-powered Supermarine Attacker on which, of course, propeller wash was absent.

3. Collar's Investigation

In 1940, at a meeting of the Aeronautical Research Committee (ARC), the question was asked: why is the Spitfire 40 mph faster than the Hurricane when both are powered by the same Merlin engine? The person delegated to answer this was Arthur Roderick Collar (1908 – 1986), then at the Aerodynamics Department of the NPL. His report [26], dated June 1940, to the ARC's Aerodynamics Sub-Committee had remained an internal ARC document hitherto unremarked in the debate on the Spitfire's aerodynamics.

Collar's investigation benefitted from three things. Firstly, he had accurate data on the performance of both aeroplanes: maximum speeds of 325 mph (Hurricane) and 365 mph (Spitfire) at 18,000 ft, the Merlin's power at that altitude being 1,020 hp, both propeller efficiencies 80%. At these conditions, he states, the C_L values are about 0.12 for the Spitfire, 0.15 for the Hurricane. From these data he is able to calculate both the overall drag and the induced drag for both aircraft at the much lower speed of 100 ft/s; the reason for this choice of speed will shortly become apparent.

Secondly, a complete Hurricane had been tested in the 24 ft open-jet wind tunnel at the Royal Aircraft Establishment (RAE), Farnborough, so that data were available on excrescence drags (tailwheel, aerial post and such) which could also be attributed to the Spitfire, which as yet had not been tested. Farnborough's wind-tunnel tests had been conducted at a speed of 100 ft/s so, by suitably scaling his maximum speed data, Collar is able to draw up drag 'balance sheets' for both aircraft at that lower speed (see Table 2). The basis of these calculations is given in Appendix 1, the results being that the Spitfire's total drag is 59 lb whereas the Hurricane's is 82 lb. Thus the aim of Collar's 'balance sheets' is to provide reasoned values for individual drag items leading to these totals.

Thirdly, Collar had available methods of proven accuracy for estimating the boundary-layer drag contributions for both aircraft, contributions which turned out to be major parts of the totals (see Table 2). One method [27], for aerofoil sections and dating from 1937, had been devised by two Farnborough aerodynamicists, Herbert Brian Squire (1909 – 1961) and Alec David Young (1913 - 2005). For a given aerofoil section shape, the method used the boundary-layer integral momentum equation together with a momentum balance between the aerofoil's trailing edge flow and its subsequent wake so as to calculate both the skin friction and pressure drags created by the boundary layer. The boundary layer was taken to be turbulent with its internal velocity variations following von Kármán's logarithmic law [28, 29]. Young [30] subsequently extended the method to bodies of revolution. Thus the boundary-layer drags, or profile drags, of both wings and fuselages could be estimated and are the first items listed in Table 2.

Table 2 Drag Contributions in lb at 100 ft/s

	Spitfire	**Hurricane**
Profile drag (wings, fuselage, and tail)	32.2	40.5
Roughness (including rivets and joints)	2.0	3.3
Induced drag (including washout)	3.0	4.0
Cooling drag (including oil cooler)	7.0	8.0
Air intake	1.0	1.0
Tail wheel	2.0	2.0
Tailplane protection	0.3	---
Gun holes, aerial post	0.8	0.8
Windscreen	1.2	1.5
Leaks	5.0	11.0
Wing-body interference	4.5	9.9
Total	**59.0**	**82.0**

As to the choice of turbulent boundary-layer methods for the estimation of profile drag, Collar [26] makes the important point that, due to turbulence in the propeller wash, the boundary layers on the fuselages and the inboard parts of the wings are likely to be entirely turbulent. Moreover, the gun holes outboard of the propeller washes, four per wing in both cases, are also likely to incite turbulence there.

Although Table 2 shows that roughly half of the drag, the profile drag, is created by the boundary layer in both cases, the Spitfire's profile drag is significantly less than that of the Hurricane. From his calculations, Collar attributes much of this to the Spitfire's thinner wing. Contributions from air intake, tail wheel, gun hole and aerial post drags had been obtained from the wind-tunnel tests on the Hurricane and are assumed to have the same values for the Spitfire. The induced drags are calculated using equation (1) above on the assumption that the C_L values in both cases are those small quantities at the maximum speed conditions quoted above. Table 2 shows that the induced drags are about 5% of the total in both cases. However, at higher incidence/C_L cases, such as the turning problem discussed in Section 4.3 and its Appendix 2 below, the induced drag contributions will be found to be far higher. As to Collar's estimates for the Spitfire's other drag contributions, these are based on reasoned assessments drawn from close inspections of both aircraft. He judges the Spitfire's better finish to be the main factor in his slightly lower-valued assessments for roughness, cooling, windscreen, leaks (control gaps etc.) and wing-body interference (the Spitfire's wing root fillet receives particular approval); here again, small percentage reductions add up. There remains the small-valued item labelled 'tailplane protection', present on the Spitfire but absent on the Hurricane, and on this Collar offers no explanation so that this item is something of a mystery.

As to the importance of t/c ratio, Young [31], in reviewing Reference 27's and similar methods, points out that a good approximation to their results is

$$(\text{Boundary-layer pressure drag})/(\text{Total boundary-layer drag}) \approx t/c.$$

The total boundary-layer drag is the sum of the skin friction and pressure drags. The pressure drag arises from the fact that the slower moving boundary layer slightly displaces the flow field exterior to it. Thus that flow's pressure at the aerofoil's trailing edge does not return to the value it would have achieved had the boundary layer been entirely absent, thereby creating a fore-and-aft pressure imbalance. Crudely speaking, the greater the t/c ratio, the greater is the flow deceleration aft of the aerofoil's maximum thickness, the consequence being that the boundary layer's growth rate increases and so does its displacement effect.

The above underlines the wisdom of choosing as small a value as possible for the t/c ratio in the Spitfire's design. In the discussion following his lecture recorded in Reference 7, Alan Clifton remarks that this choice went against the advice of the NPL, advice to the effect that "there was no advantage in going below a thickness chord ratio of 15%". It appears that similar advice was given to Sydney Camm at Hawker during the design of the Hurricane. Roy Chaplin, a senior member of Camm's team at that time, recalls in his contribution to Reference 32 that wind tunnel tests at the NPL on a Hurricane model produced the advice that "no improvement in drag would be obtained by reducing the thickness-chord ratio of the wing below 20%." As mentioned in Reference 1, it appears that this advice arose from measurements obtained in the NPL's Compressed Air Tunnel (CAT), running since 1932, which, it was later realised, could produce misleading results.

Two ARC reports [33, 34] published roughly two years apart shed some light on this issue. Both reports carry the same title, have the same authors and present aerodynamic data for the same six aerofoils (RAF 28, 34, 38, 48, Clark YH, Göttingen 387) over a similar range of Reynolds numbers; the influence of the Reynolds number will be discussed later. The aerofoils have t/c ratios ranging from around 9% to 15%. However, the second report, dated April 1936, stresses that it replaces the first report of September 1934. The 1934 report [33] argues that, whilst the turbulence level in the CAT is still under investigation, it is felt that this is significantly less than that in the NACA variable-density tunnel (from which the data of Reference 16 had been obtained) and is therefore more capable of achieving conditions representative of those at full-scale in free flight. As to a possible drag dependence on t/c ratio, the authors merely note that, apart from the higher results for the Göttingen 387 section (one of the thickest), the others lie fairly close together and include the RAF 48 section of similar t/c ratio to the Göttingen section but of reduced camber. Their graph for the drag results shows a rather scattered band in which no clear trend is evident. However, the second report [34] of 1936 states that corrections to the previous results are necessary due to parasitic drag in the aerofoils' wire suspension system and slight deviation of the flow from the horizontal, the errors for which are additive and affect, in particular, the earlier drag results. After re-testing with better aerofoil supports and correcting for flow deviation, the authors [34] conclude that "the whole series suggests that for normal aerofoils between 9 and 12 per cent thick there is very little variation of profile drag, but that above 12 per cent thickness the drag begins to rise appreciably." Since the ARC continued to assert that Reference 34 replaced Reference 33, it appears that the former paper remained the final word on the matter.

It seems that Sydney Camm had followed the earlier flawed evidence for the Hurricane, the latter emerging with the significantly higher t/c ratios shown in Table 1 and the consequent higher drag. Collar (26), in commenting on this, can foresee some improvements to the Hurricane's performance if leaks and such are attended to but predicts that the increase in maximum speed might be only 15 mph. Otherwise, a major re-design with particular emphasis on reducing the wing's t/c ratio would be necessary in order to produce any marked improvement.

Corroboration of Collar's Spitfire results is provided by Price (11) in his description of the Speed Spitfire of 1938, an aerodynamically cleaner version intended for an attempt at the world landplane speed record. Price (11) states that, whereas the standard Spitfire Mk I was estimated to have a drag of 60.2 lb at 100 ft/s (Collar's estimate above is 59 lb ±3 lb), the Speed Spitfire's drag was 53.3 lb. This was accomplished through a wing-span reduction, a streamlined skid replacing the tailwheel, a longer, more streamlined windscreen and flat canopy, the filling of all cracks and a finely polished finish.

Collar's report (26) undertakes a further task, which is to resolve what appeared to be inconsistencies between the drags measured in Spitfire and Hurricane model tests in the CAT and those measured on the full-scale aircraft. Collar (26) notes that the Hurricane model lacked such features as the air intake, tail wheel and aerial post whereas all these items were present on the Spitfire model. Moreover, both models were leak-free and some further adjustments to his drag totals are allowed in the assessment of model roughness in comparison with that of the full-scale aircraft. Taking all these matters into consideration, Collar (26) reduces his drag totals of Table 2 to 52.6 lb for the Spitfire and 59.8 lb for the Hurricane. These values are then taken to be the drags at 100 ft/s on full-scale aircraft re-modelled to be close to the configurations of the CAT wind-tunnel models. The drag coefficients, C_D, calculated by Collar (26) on this basis are

$$\text{Spitfire } C_D = 0.0186 \quad \text{Hurricane } C_D = 0.0190.$$

It is seen that the Hurricane's drag has been whittled down rather more than that for the Spitfire. Since the resulting drags are, by chance, almost in the ratio of the wing areas (238 ft^2 for the Spitfire, 264 ft^2 for the Hurricane), the C_D values turn out to be almost the same. It must be stressed, however, that these C_D values are not those for the actual aircraft; this point will be dealt with later.

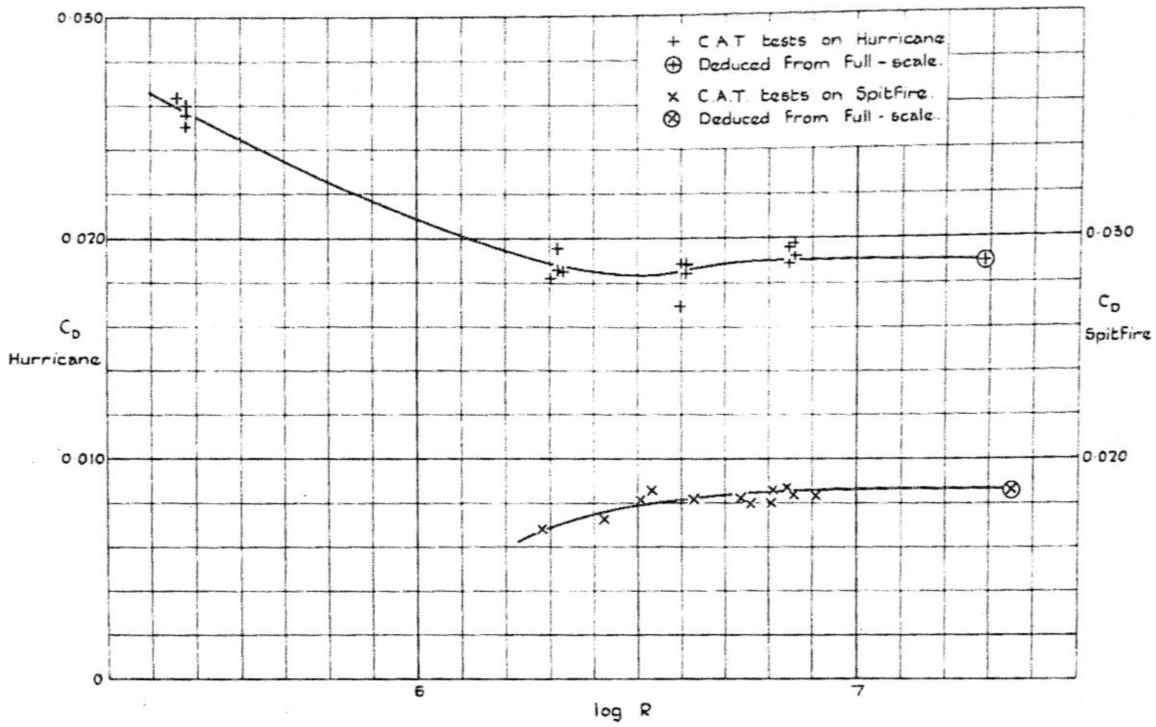

Figure 3 Drag coefficient, C_D, variations with Reynolds number, R, for Spitfire & Hurricane models. Collar [26]

The C_D values quoted above are shown as circled crosses in the graphs of Figure 3 in which the C_D values obtained from the CAT model tests, marked as plain crosses, are plotted against the logarithm (to the base of ten) of the test Reynolds number, R. Care is needed in reading these graphs since two C_D ordinates are used, one for each aircraft model, and are displaced relative to each other to avoid collision between the graphs. As Collar [26] argues, whilst the test models' Reynolds numbers do not reach the full-scale values of over 10^7, the CAT's C_D model values extrapolate well to the above full-scale C_D values. In this respect, then, there is no inconsistency between the CAT model and the full-scale results.

Figure 3 illustrates the long-running problem of wind-tunnel testing, which is an inability to achieve the Reynolds numbers of full-scale aeroplanes in flight. Readers might find it helpful here to study Appendices 6 and 7 of Reference 35 in order to understand this problem and to see the pitfalls of not achieving full-scale Reynolds number conditions. The essential point is that, to be truly representative of full-scale conditions, to be precisely dynamically equivalent, the model test Reynolds number and that for the actual full-scale aircraft must have exactly the same value. The Reynolds number, R, represents the ratio of inertia to viscous forces in a flow field and is given by

$$R = \rho V l/\mu,$$

where ρ is the air density, V is the airspeed, l is a representative length scale (the wing's root chord, say) and μ is the air's viscosity coefficient. Clearly, small models tested at the relatively low speeds achievable in many wind tunnels fall far short of achieving full-scale Reynolds number conditions. Even Farnborough's 24 ft open-jet tunnel mentioned above, whilst accommodating full-scale aircraft, ran at a flow speed of 100 ft/s, roughly 68 mph, which is far less than the maximum speeds of most aircraft. One method adopted to reduce this large difference in Reynolds number was to increase the air density by pressurisation. This was the measure adopted for the CAT which could be pressurised to 25 atmospheres, raising the air density in that ratio. The NACA's variable-density tunnel used a similar method to obtain the data for the then-new NACA aerofoil sections [16] mentioned in Section 2. Even so, as Figure 3 illustrates, the test Reynolds numbers achieved in the CAT were roughly ten times less than those at full scale. The question then was, would extrapolation of such wind-tunnel results provide realistic full-scale values?

Whilst Collar's extrapolations of the CAT's results provided a welcome affirmative answer, at least in this respect, as noted earlier it must be emphasised that his C_D values quoted immediately above are not those of the actual full-scale aircraft. The latter can be obtained directly from the totals of Table 2 and according to this author's calculations are

$$\text{Spitfire } C_D = 0.021 \qquad \text{Hurricane } C_D = 0.026.$$

Deducting the induced drag contributions of Table 2, the drag coefficients at zero lift, C_{D0}, are

$$\text{Spitfire } C_{D0} = 0.020 \quad \text{Hurricane } C_{D0} = 0.025.$$

It is intended to include these results in a later paper which will review aerodynamic data from this and earlier eras.

4. Other Aerodynamic Matters

4.1 Surface Finish

In his article included in References 11 and 12, Shenstone refutes accusations that Supermarine had simply copied the Heinkel 70's elliptic wing. As he rightly points out, the Heinkel's wing planform is not the semi-elliptic shape chosen for the Spitfire and the Heinkel's thicker aerofoil section differs significantly from the Spitfire's thin NACA section. He admits, however, that the one thing he wished to copy, in a sense, was the Heinkel's smooth surface finish. Impressed by this when he had seen it at the Paris Aero Show in 1934, he recommended that its flush-rivet construction be adopted for the Spitfire. Thus the Spitfire prototype, K5054, emerged in March 1936 with this smooth finish. However, this riveting process was more time-consuming and expensive than the conventional method using dome-headed rivets. To check on the efficacy of this smooth finish, split peas were glued to the flush rivet heads using Seccotine, a popular glue at the time. Reference 10 includes Shenstone's two photographs of K5054 in this condition, its surface appearing to have acquired a regularly arrayed chickenpox infection. In January 1937 flight tests were conducted by Quill in which the split peas were removed in progressive stages from nose to tail [10]. Ernest Mansbridge retained the results which are included in Price [11] and these are listed in Table 3.

Table 3
The loss of speed due to fixing split peas all over the aircraft is 22 mph

This is made up as follows:	
Leading edge aft to spar at ¼ chord	**8 mph**
Transverse rows on wing, fuselage & tail*	**4.5 mph**
Fore & aft rows on top surface of wing*	**5.5 mph**
Fore & aft rows on under surface of wing*	**3 mph**
Fore & aft on fuselage	**1 mph**
***Aft of spar on wing**	

Clearly, the choice of flush-riveting was vindicated by this exercise. At first sight, the detrimental effect of conventional riveting seems surprising, given Collar's contention (see Section 3) that the Spitfire's boundary layer would be largely turbulent. However, since the early 1920s it had been known that surface roughness has a detrimental effect on turbulent pipe flow. Moreover, in 1934 the investigation by Prandtl and Schlichting (36), in which various forms of roughness were assessed, established that roughness at the surfaces of flat plates experiencing turbulent boundary layers significantly increases the boundary-layer drag.

4.2 Propulsion Effects on Aerodynamic Performance

Early flight tests with the Spitfire prototype, K5054, established its maximum speed at around 335 mph (11, 25). This disappointing result, it was felt, might be due to compressibility effects on its coarse-pitch twin-bladed wooden propeller. Supermarine already had experience in this area as a result of its Schneider Trophy work; see Clifton's comments at the close of Section 5 below. Re-design of the propeller using a smaller t/c ratio and finer incidence at the tip region improved propeller efficiency, thus raising the maximum speed to 348 mph (11, 25). During subsequent tests with K5054, a number of propeller types were tried in an attempt to increase further the aircraft's maximum speed. These were mainly fixed-pitch twin, three and even four-bladed types although a two-pitch De Havilland three-bladed type was tried. No significant improvements seem to have emerged from this (11).

However, a distinct improvement emerged from work by Rolls-Royce which indicated that directing the engine's exhaust gases directly rearwards would augment thrust. As Price (11) relates, this thrust augmentation could amount to 70 lb, equivalent to a 70 hp increase at 300 mph. Alternatively, an aeroplane powered by about 1,000 hp flying at 300 mph experiences around 1,000 lb of thrust. Thus an additional thrust of 70 lb represents a 7% thrust increase, or nearly 9% when current propeller efficiencies around 80% are taken into account. In September 1937 ejector exhausts producing this effect were fitted to K5054 and its maximum speed rose to around 360 mph (11).

Despite the Type 224's Goshawk cooling system having the advantage of requiring no external radiator, its disadvantages have been mentioned in Section 2. The switch from the steam-cooling system of the Goshawk and initial PV 12 engines to one based on ethylene glycol as the coolant now introduced the requirement for a radiator. Since glycol boils at a little under 200°C at normal pressure, its use as a coolant was seen as attractive. However, glycol was expensive and very prone to leakage. Eventually, a mixture of 70% water and 30% glycol was selected [(10)].

As to the radiator, "cometh the hour, cometh the man", in this case F. W. Meredith of the RAE [(37)] with his proposal published in 1935 for a ducted radiator system which would greatly reduce radiator drag. Meredith's proposal rested on two interlinked ideas. The first reduces the drag of the radiator matrix itself, this varying as the square of the local air speed, by reducing the latter at the radiator. This is accomplished by capturing the cooling air in a duct which increases in area up to the radiator, the flow speed dropping progressively according to the continuity principle (see Appendix 1 of Reference 35) as the duct area increases. The second idea is to utilise the heat energy released by the radiator. The combination of the continuity principle and the energy equation indicates that some of the heat energy can be translated into the kinetic energy of a thrust-producing high-speed jet by passing the radiator efflux through a convergent duct. This would counteract the residual drag of the enclosed radiator. The scheme was introduced to the Spitfire's design after wind-tunnel tests on a model duct and wing were conducted at the RAE. These were the only wind-tunnel tests undertaken for the Spitfire's design, and indicated satisfactory performance [(10)]. If Collar's estimates of Section 3 are to be relied upon, the combined thrust at 100 ft/s provided by the ejector exhausts and the radiator, 7.5 lb (see Appendix 1), can be set against the overall cooling drag of 7 lb (see Table 2), thus making the total engine system virtually 'drag-neutral'. A further point is that, if this author's estimate of Collar's initial value for the combined exhaust and radiator thrust ($T_j \approx 120$ lb, see Appendix 1) at maximum speed is correct, and given the Rolls-Royce value for exhaust thrust of 70 lb (see above), then the thrust contribution from the radiator is of the order of 50 lb.

The efficacy of the Meredith Effect depends on the area ratios employed in the ducts used. A greater area ratio for the entry duct produces a greater radiator drag reduction; similarly, a greater contraction ratio for the jet efflux increases the jet thrust. The drawings of the Spitfire's radiator duct provided by Reference 10 indicate area ratios around 2, these being limited by the duct's placement beneath the wing. In the case of the P-51 Mustang, in contrast, the radiator's placement at the underbelly of the fuselage offered room for significantly greater area ratios, perhaps of the order of 4. Consequently the Mustang benefitted rather more from the Meredith Effect than did the Spitfire. Indeed, Atwood [(38)], who was involved in the Mustang's design, claims that the greater implementation of the Meredith Effect here was more significant in terms of drag reduction than the use of the so-called laminar flow wing. A further interesting feature of the Mustang's radiator is that its entry scoop is clear of the fuselage skin so that the entry flow is not contaminated by the fuselage boundary layer, a feature absent on the Spitfire. Ludwig [(39)] claims that this beneficial item in the Mustang's design was suggested by Beverley Shenstone, presumably when he was working for the British Air Commission in the United States.

Two final points are worth mentioning concerning the Spitfire's propulsion system. One is that production Spitfires did not begin to receive De Havilland variable-pitch propellers until

May 1939 [11] (from the 78th production aircraft onward). These propellers improved take-off and climbing performance. The second point is that in 1939 the Air Ministry upgraded its requirement for aviation fuel from 87 Octane to 100 Octane. This enabled higher supercharger boost pressures to be used, thereby increasing engine performance. In both cases the year, 1939, is significant here. Whilst the Chain Home radar system was unquestionably crucial for what was soon to follow, the above two important technical advances also arrived just in time. This indicates that the preparation for what was to become the Battle of Britain in the summer of 1940 was a fine-run thing.

4.3 The Turning Radius Problem

Before discussing this, a description of this author's initial involvement might be of interest and, indeed, cause some amusement. Some years ago, this author and his colleague at Manchester's Victoria University, Dr Peter Lamont, were approached by Dr Stephen Bungay who was writing a history of the Battle of Britain [40]. Dr Bungay had been puzzled by certain statements in the popular aviation literature concerning the turning radii of the Spitfire and the Bf 109. The one source of information immediately to hand quoted limited extracts from an internal Farnborough report stated to be BA 1604; this turned out to be a report on seaplane performance. After a fruitless search through the ARC's wartime publications, any further literature search was abandoned in favour of analysis since a quick response was needed. The essential steps in this analysis are given in Appendix 2 and lead to the result that the radius, r, for turning flight at constant speed in the horizontal plane is given by

$$r = 2w/(\rho\ g\ C_L \sin \varphi). \qquad (2)$$

Here w is the wing loading, ρ the air density, g the gravitational acceleration, C_L the lift coefficient and φ is the bank angle. The results based on this relation were published in Reference 41. After publication, entirely by chance it was discovered that the elusive Farnborough report was BA 1640, not BA 1604; moreover, this report by Morgan and Morris [13] had then been published by the ARC. However, although dated September 1940, this had not been published until after the war, presumably for security reasons.

The important point to note from equation (2) is that turning radius, r, is directly proportional to w, the wing loading. The Spitfire's semi-elliptical wing of substantial area helped in this respect, producing a wing loading around three-quarters of that for the Bf 109: 24.8 lb/ft^2 for the Spitfire, 32.2 lb/ft^2 for the Bf 109 [13]. The results of both investigations [13, 41], compared in Table 4, reflect this; the quantity **n** in that table is the ratio of lift to weight (see Appendix 2). It should be noted that, whereas the altitude selected in Reference 41 is 10,000 ft, in Reference 13 this is 12,000 ft. According to equation (2), the decrease in air density, ρ, with increased altitude will lead to increased turning radii and this is also reflected in Table 4's results. In both investigations it is assumed that the turning flights occur at maximum power and at maximum, or near-maximum, C_L values. However, Reference 13's investigation benefitted from the provision of accurate data both for the Spitfire and for the RAE's captured Bf 109E-3, the latter having been subjected to extensive testing. Reference 41, in contrast, had to rely on less reliable data coupled to a certain amount of guesswork. For example, the C_L values chosen are less than those of Reference 13 and such differences lead to differences in the speeds as registered by the pilot's air-speed indicator (V(ASI)). Nonetheless, both investigations lead to the conclusion that the Spitfire is superior to the Bf 109 in turning performance. Moreover, the Spitfire proved superior in other respects, as Reference 13's extensive investigation of its Bf 109 indicates.

Table 4 Turning Data

	Radius/ft	n	Bank Angle φ°	C_L	V(ASI)/mph
Spitfire (41)	687	2.7	69	1.35	85
Spitfire (13)	696	2.7	68	1.47	126
Bf 109 (41)	853	2.7	68	1.40	94
Bf 109 (13)	885	2.1	62	1.60	118

The results obtained by Morgan and Morris (13) are derived from analyses by Gates (42, 43). In 1932 Gates (42) produced a general analysis of sustained turning motion which includes change of altitude (spiralling motion) and the possibility of sideslip. In 1942 he used this analysis to investigate various dog-fight scenarios (43). For the simpler case of sustained turning motion at constant altitude discussed here, Gates's analysis reduces to that outlined in Appendix 2.

Morgan and Morris (44) later considered the influence of altitude on the Spitfire's turning performance, extending their calculations to an altitude of 35,000 ft. As stated earlier, the turning radius increases with altitude and, to counter this, Morgan and Morris (44) suggest the use of flaps, settling on an optimum flap angle of 30°.

5. The Spitfire's Performance at High Subsonic Mach Numbers

Reference 1 included a graph showing the variation of the drag coefficient at zero lift, C_{D0}, with Mach number for both the Spitfire PR XI and the P-51 Mustang. It is here reproduced as Figure 4 for ease of reference. The graph was taken from Jeffrey Quill's contribution to Reference 7, the data, dated January 1944, having been obtain from Farnborough's wartime high-speed flight test work. However, there is a problem with this graph; one would not expect variations in C_{D0} to occur prior to the onset of compressibility effects. Thus the initial gradual rises in C_{D0} are suspect and may have arisen from data reduction errors in dealing with the original flight-test results. This matter is resolved, however, in the large compendium (45) of Farnborough's wartime high subsonic research work published in 1950 and edited by William Austyn Mair (1917 - 2008). The rather busy graphs of Figure 5, taken from that publication, show no increases in C_{D0} until critical Mach number conditions are encountered.

Critical Mach number conditions occur when, at some point in the flow field, the local flow velocity reaches the local speed of sound. These conditions usually occur on wing upper surfaces, at which point shock waves begin to form there and create an addition drag, wave drag, causing an increase in C_{D0}. As shown in Figure 5, the onset of this compressibility

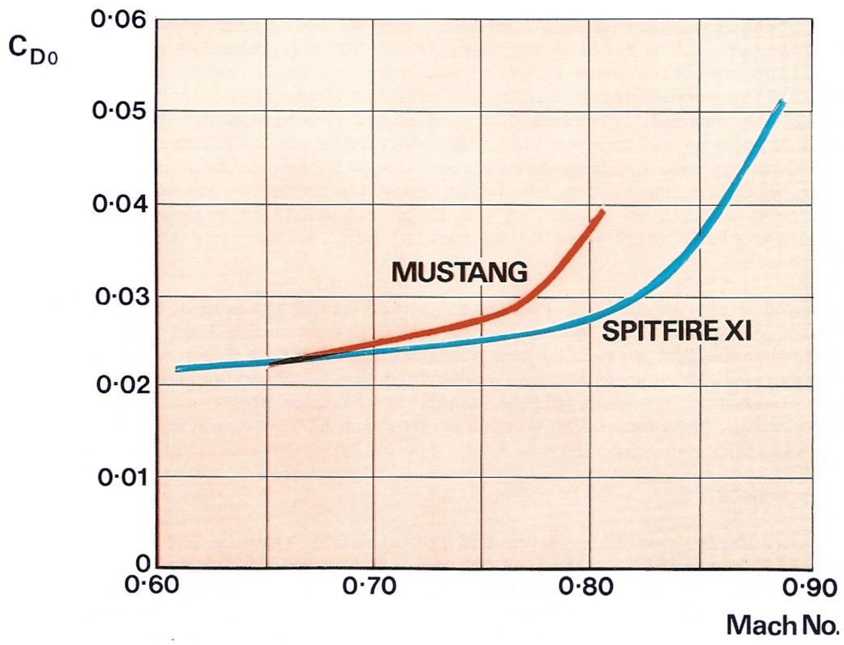

Figure 4 Variation of C_{D0} with Mach number for the Spitfire PR XI and the P-51 Mustang, from Quill's contribution to Reference 7

effect occurs well below the point at which the flight Mach number approaches unity, and depends on the shape of the aircraft. A wing's t/c ratio is thus important here and the smaller its value, the higher is the aircraft's critical Mach number. As Figure 5 shows, the Spitfire with its thinner wing benefits in this respect compared with the Mustang with its slightly thicker wing. However, there is a further feature to be seen in both Figures 4 and 5 which is that the Spitfire's rise in C_{D0} is more gradual than that for the Mustang. At the risk of being

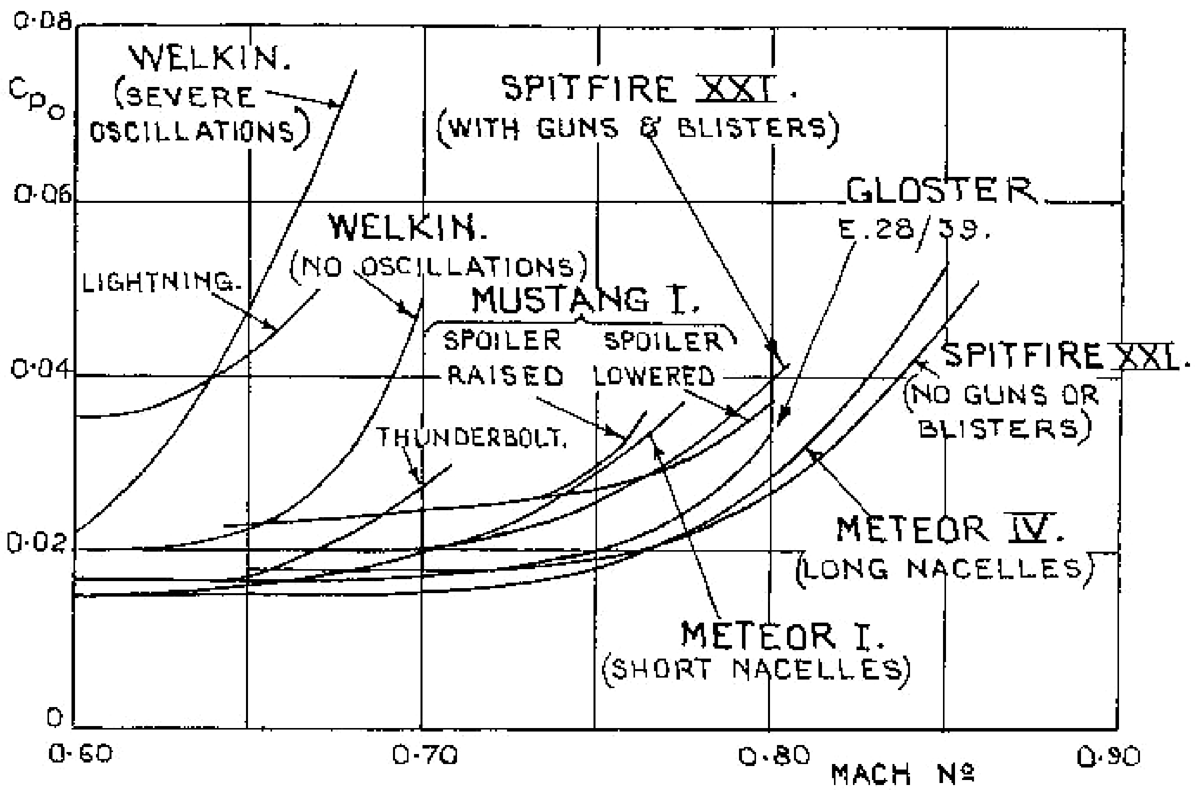

Figure 5 Variation of C_{D0} with Mach number for various aircraft [45]

over-simplistic, a clue to this difference in behaviour might lie in the basic ideas associated with the Area Rule.

In the past the discovery of the Area Rule for flight around the speed of sound had been attributed to Richard Travis Whitcomb (1921 – 2009) whilst working at the NACA. His first paper [46] on this appeared in 1952 and the work culminated in his report of 1956 [47]. However, it is now know that the concept originated with Otto Frenzl (1909 – 1996) around 1943 whilst working at Junkers in Dessau (see Reference 48 for a review of his work). The Area Rule's basic idea is that, from nose to tail, the variation of an aircraft's total cross-sectional area (including the wing and tail assembly) should be as smooth as possible. Depicting this variation as a graph, any sudden changes in its gradient should be avoided if possible since such changes will lead to increased wave drag and an increase in C_{D0}. If such changes do occur, they should be as small as possible. A frequently cited example is the behaviour of the prototype Convair F-102 Delta Dagger (1953/54) which proved incapable of achieving supersonic flight. The remedy provided by Whitcomb's work was to waist the fuselage to compensate for the delta wing's cross-sectional area, the result being that, in plan view, the fuselage acquired a Coke Bottle shape. As a result, the F-102 was able to achieve supersonic flight.

Applying the above idea to the Spitfire and Mustang, graphs of cross-sectional area variation aft from the nose would show smooth variations until the wings are reached. At that point sudden bumps in the curve will occur and thus create increased wave drag. But the bump for the Spitfire, with its thinner, broader chord wing of elliptic taper, will be more gradual than that for the Mustang with its slightly thicker, smaller chord wing. Basically the rate of change of cross-sectional area around the region of the wing is less marked for the Spitfire than is the case for the Mustang. Furthermore, for the Mustang there will be an additional small bump due to the underbelly radiator intake. This argument might, it is suggested, provide a clue to the difference in high subsonic behaviour of the two aircraft. That said, it is nonetheless remarkable that, as shown in Figure 5, there should be so large a difference in C_{D0} behaviour between the aerodynamically cleaner Mk 21 Spitfire with its cannon and ammunition blisters removed and the standard version.

The switch from the PR XI Spitfire of Figure 4 to the Mk 21 of Figure 5 was due to two serious accidents involving the former Spitfire mark. On the first occasion, in May 1944 as part of an instrumentation check, the Spitfire was in a very steep dive up to a Mach number of 0·89 [45]. A loss of oil pressure to the constant speed propeller unit resulted in the propeller severely over-speeding. The reduction gear and propeller were torn away, thus preventing any further tests from being made on this aircraft. On the second occasion a month later, as Price relates [11], the replacement PR XI suffered a burst supercharger and engine fire. On both occasions the pilot, Squadron Leader Martindale, not only survived but also managed to save the flight data. This stands as a notable tribute to his skill and courage; clearly the RAE's high subsonic flight test programme involved exacting and dangerous work. The final replacement for the PR XI, the Mk 21 with guns and blisters removed, is judged in Reference 45 to be not quite as clean aerodynamically as the PR XI.

Two versions of the Gloster Meteor appear in Figure 5, the Mk I and one labelled Mk IV. However, the latter is an interim type denoted in Reference 45 as “Meteor IV”, signifying that it is a pseudo-Mk IV, aerodynamically close to the production Mk IV but in fact a modified Mk I. Having experienced buffeting at high subsonic Mach number with the Mk I, the RAE deduced from in-flight wool tuft tests that the problem lay in the appearance of shock waves around the short engine nacelles which infected the wing flow there. The RAE suggested lengthening the nacelles, a measure adopted for the “Meteor IV” and production Mk IV, and this alleviated the problem. It is seen in Figure 5 that the “Meteor IV” approaches the cleaner Spitfire’s performance and does rather better than the first British jet-propelled aircraft, the Gloster E28/39.

As to the other aircraft appearing in Figure 5, the P-38 Lightning scarcely makes it onto the graph, whilst the twin-engine high-altitude Westland Welkin (maximum t/c ratio 21%) produced occasional oscillatory behaviour – hence the two graphs - which, as far as this author is aware, was never resolved. There remains the P-47 Thunderbolt and, in commenting on this in comparison with the Mustang and the Spitfire, Quill [(25)] remarks that this was “in the ‘also ran’ category”. According to the Area Rule idea outlined above, the Thunderbolt, with its blunt nose and consequent sudden jump in cross section area there, might be expected to do badly.

In the above discussion there is no intention to suggest that aircraft such as the Spitfire and the Mustang were in any way Area-Ruled. With the benefit of hindsight it could be argued that the nacelle lengthening on the Meteor was in accord with Area Rule ideas, but it should be stressed that at that time the RAE, indeed the Allied Powers, had no knowledge of the concept then being explored by Otto Frenzl. As to the Spitfire specifically, when asked in the discussion following his lecture recorded in Reference 7 if compressibility effects were considered in the Spitfire’s design, Alan Clifton replied, “Nothing, I’d say!” Later he amplified that remark by stating that in the late 1920s Supermarine had evolved guess-work corrections for compressibility effects on the tip sections of the Schneider Trophy aircrafts’ metal propellers. He concluded with the statement, “However, we certainly saw no reason to apply this data to the Spitfire wing.”

6. Concluding Remarks

This paper has attempted to describe the basic aerodynamic ideas incorporated in the Spitfire’s design. The emerging picture is one in which few aerodynamic compromises occurred and, indeed, the latest aerodynamic ideas available in the early 1930s were successfully incorporated. Yet this picture, aerodynamically speaking, is not complete; as later marks of Spitfire emerged, changes in their aerodynamics, though often slight, did occur. Moreover, other aspects of the Spitfire story are equally worthy of investigation: structure and aeroelasticity, stability, control, general handling qualities, engines and propellers. All of these contributed to the Spitfire’s phenomenal development [(17)] in which its power and weight both roughly doubled whilst its maximum speed rose by about 100 mph. Perhaps other writers might take up the challenge of addressing these matters.

Since its first appearance in early March 1936, the Spitfire has now reached the venerable age of eighty, a circumstance which prompted the Spitfire Seminar's celebration mentioned at the beginning of this paper. Through its longevity and historical associations it has become, one might say, a National Treasure to be displayed at times of major celebration, a part of the story the British like to tell about themselves. Its crucial importance, particularly during the dire circumstances of the summer of 1940, is unquestionable. As Salisbury (21) points out, its highly distinctive planform provided a considerable boost to civilian morale, particularly in the south-east of England, but had quite the opposite effect on German pilots. British airspace was seen to be defended, and effectively so. This alone stands as a considerable tribute to Mitchell and his design team at Supermarine. A further tribute, and again a considerable one, is the ability of the Spitfire's design to answer the exacting developmental demands placed upon it. As to Mitchell's design team itself, a striking feature is its loyalty to Mitchell, even more so to his memory. And, whilst individual team members freely gave credit to others for their contributions, Clifton to Shenstone, Shenstone to Joseph Smith and so on, none showed any sign of claiming credit for themselves.

Left somewhat dazed after this heady trawl through a limited part of the copious Spitfire literature, perhaps this academic, now almost as old as the Spitfire itself, might be allowed to bow out with a touch of levity. To plagiarize two well-known slogans and with an apology for the repeated classic split-infinitive, the Spitfire's achievement of a Mach number of 0.9 toward the end of its wartime career touched those parts (of the flight envelope) which other aircraft could not reach, enabling pilots to boldly go where no (air)man had gone before.

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Appendix 1

Collar's Calculations

The following analysis, not present in Collar's paper [26], is this author's attempt to provide the basis for Collar's calculations.

For an aeroplane flying straight and level at constant speed, V, the thrust, T, must equal the drag, D, and lift, L, must equal the aeroplane's weight, W. Thus

$$T = D, \quad L = W.$$

However, there are two contributions to T, one being from the engine/propeller combination, T_e, the other being the jet thrust provided by the engine exhaust and heat regeneration in the radiator, T_j. Thus

$$T_e + T_j = T = D,$$

$$(T_e + T_j)V = DV.$$

For a propeller efficiency, η, and engine power, P,

$$T_eV = \eta P.$$

The drag, D, can be written in terms of the drag coefficient, C_D, as

$$D = ½\, \rho\, V^2\, S\, C_D,$$

where ρ is the air density and S the wing planform area. Thus,

$$\eta P + T_j\, V = ½\, \rho\, V^3\, S\, C_D.$$

It follows that

$$\eta P /(V^3) + T_j/(V^2) = ½\, \rho\, S\, C_D.$$

Writing $\sigma = \rho/\rho_0$, ρ_0 being the sea-level air density, then

$$\eta P /(\sigma V^3) + T_j/(\sigma V^2)= ½\, \rho_0\, S\, C_D.$$

However,

$$½\, \rho_0\, S\, C_D\, (100)^2 = D_{100},$$

which is the drag at sea level at a speed of 100 ft/s. Consequently,

$$10^4\, \eta P/ (\sigma\, V^3) + T_{j,100} = D_{100},$$

$$T_{j,100} = 10^4 T_j/(\sigma\, V^2), \qquad \text{(A1)}$$

the latter being the exhaust/radiator thrust at a speed of 100 ft/s. Thus,

$$10^4\, \eta P/ (\sigma\, V^3) = T_{e,100} \qquad \text{(A2)}$$

is the engine/propeller thrust at 100 ft/s.

This last result (equation A2) is Collar's starting point for his calculations [26]. He uses it in what he calls "the usual procedure" to scale the data for the maximum speed and corresponding engine power conditions at 18,000 ft altitude to a common speed of 100 ft/s at sea level. This yields his estimates for engine/propeller thrust in lb for both aircraft, as shown in the table below. To these he adds his estimates for $T_{j,100}$, the thrust due to the exhaust and the radiator's heat regeneration at sea level and at the speed of 100 ft/s. He says these estimates have been obtained by scaling an unstated value, but common to both aircraft, at the maximum speed conditions. Using equation (A1), that unstated value can be retrieved by scaling up the values for $T_{j,100}$ shown in the table to the maximum speed conditions. This yields a T_j value of about 120 lb for both aircraft.

Collar's totals given in the table below are then those totals appearing in the drag 'balance sheets' of Table 2, to which he gives a probable error of ±3 lb.

	Spitfire	Hurricane
Engine/propeller thrust, $T_{e,100}$	51.5	73.0
Exhaust & heat regeneration, $T_{j,100}$	7.5	9.0
Total	59.0	82.0

Appendix 2

Turning Flight

For an aeroplane in straight and level flight at constant speed, the lift, L, opposes the aeroplane's weight, W = mg, m being the aeroplane's mass, g the gravitational acceleration. Suppose now that the aeroplane banks through the angle φ from the horizontal whilst maintaining its speed, V, and its altitude. Since the lift force continues to be perpendicular to the plane of the wings, the lift is now directed at the angle φ to the vertical. The lift force's vertical component, L cos φ, is consequently now less than the weight, W. To correct this, the wing incidence angle must be increased so that the lift force increases to the extent that

$$L \cos \varphi = W.$$

There is also the lift force's horizontal component, L sin φ, which provides the centripetal force causing the aeroplane to follow a circular path in the horizontal plane. A body moving in a circular path of radius r experiences a continual acceleration towards the circle's centre, a centripetal acceleration, equal to V^2/ r. Consequently, since force is equal to the product of mass and acceleration,

$$L \sin \varphi = m V^2/r.$$

The lift force can be written in terms of the lift coefficient, C_L, as

$$L = \tfrac{1}{2} \rho V^2 S C_L,$$

ρ being the air density and S the wing planform area. Rearranging, we obtain

$$r = 2w/(\rho\ g\ C_L \sin\varphi),$$

where w = W/S is the wing loading. The crucial point is that the radius of turn is directly proportional to the wing loading.

We wish the aeroplane to retain its altitude so as to turn in a horizontal plane. As explained earlier, it is then necessary to increase the lift, L, by increasing the wing incidence angle. Yet this entails an increase in the value of C_L. Two potential aerodynamic problems arise here. The first is that C_L might be increased to its stall value, a difficult situation for the pilot. The second is that the C_L increase increases the induced drag, its coefficient being given by equation (1) of Section 2, that is

$$C_{Di} = k\ C_L^2 /(\pi\ A),$$

A being the wing aspect ratio equal to b^2/ S, where b is the wing's span. The total drag coefficient for an aeroplane can be written as

$$C_D = C_{D0} + C_{Di},$$

where C_{D0} is the drag coefficient at zero lift. For the fighters under discussion, a typical value of C_{D0} is around 0.02, the aspect ratios being close to six. For straight and level flight at high speed the induced drag contribution is very small since the aeroplane is flying at a low incidence/low C_L value (see the values in Table 2 in Section 3). However, in the turn, taking a C_L value comfortably below stall, say $C_L = 1$, then C_{Di} has a value around 0.05. Consequently, in executing the turn whilst maintaining the speed V, the aeroplane's overall drag coefficient has increased from a little over 0.02 (see the results quoted at the end of Section 3) to around 0.07 and the drag force has increased in that proportion. The question then is: has the aeroplane sufficient engine thrust/power to sustain this speed?

There are further difficulties in turning motion and these are associated with the load factor, n, given by

$$n = L/W = 1/\cos\varphi.$$

Gates [(42)], for example, points out that in a sustained turn a pilot will experience blackout in vision at a value of n around 6 ($\varphi \approx 80°$) whereas at a value of n around 10 ($\varphi \approx 84°$) structural failure is likely to occur.