# On the aerodynamics of the Miles M.52 (E.24/43) – a historical perspective

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## ABSTRACT

The aerodynamic design of the Miles M.52 experimental supersonic aircraft is reviewed relative to the state of knowledge at its time of 1943 – 1946. Drawing on widely-ranging material, much not previously published, this study enlarges upon, and in places amends, previous accounts of the project.

Based on advice collated from the fields of aerodynamics and ballistics, Miles conceived an original and forward-looking design, backed by an extensive test programme. Novel solutions to new requirements in the areas of structures and systems provided a robust airframe, showing fair prospects of being stable and controllable throughout the specified flight envelope. An equally innovative power plant was devised for it by Whittle's company, Power Jets Ltd.

Contradictory predictions of its performance were made by Miles and the RAE, through differences in their estimations of fuselage drag. A new evaluation suggests that the available information would have shown the aircraft to be capable of providing vital aerodynamic data for the transonic and early supersonic regimes at a time when no other sources were available, though further engine development would be needed for it to reach its full potential.

## NOMENCLATURE

| | |
|---|---|
| $A$ | frontal area; cross-sectional area presented to flow |
| $c$ | aerodynamic mean chord of wing, amc |
| $cg$ | centre of gravity of aircraft |
| $C_d$ | drag coefficient of aerofoil section |
| $C_D$ | overall drag coefficient of wing or aircraft |
| $C_F$ | overall skin friction (boundary-layer drag) coefficient of surface |
| $C_L$ | overall lift coefficient of wing or aircraft |
| $C_M$ | overall moment coefficient of wing or aircraft |
| $d$ | diameter of body of projectile |
| $D$ | total drag of aircraft |
| EAS | equivalent air speed; speed at sea level giving the same stagnation pressure |
| $f_R$ | ballistic resistance coefficient |
| $g$ | acceleration due to gravity |
| $G$ | Prandtl – Glauert multiplier for effect of Mach number |
| K | constant in Equations (10) and (11) |
| $L$ | length of projectile head or aircraft nose, calibre |
| $l_v$ | derivative of rolling moment coefficient due to sideslip |
| M | Mach number |
| $n_v$ | derivative of yawing moment coefficient due to sideslip |
| rpm | rotational speed, revolutions per minute |
| $r$ | radius of body of projectile |
| $R$ | radius in definition of ogive |
| Re | Reynolds number |
| $S$ | gross plan area of wing |
| $t$ | maximum thickness of aerofoil section |
| $X$ | function of Mach number, Equation (8) |
| $V$ | true flight speed |
| $\alpha$ | incidence angle |
| $\beta$ | half-angle at apex of section |
| $\gamma$ | function of Mach number, Equation (4) |
| $\mu$ | Mach angle, rad |
| $\rho$ | air density |
| $\theta$ | inclination of local surface to free-stream direction |

### Suffixes

| | |
|---|---|
| $b$ | at base |
| $f$ | for fuselage |
| $i$ | induced by vortex motion due to lift |
| $max$ | maximum value |
| $o$ | at zero lift; at low speed |
| $wemp$ | for wings plus empennage |

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Other terms are defined where they first appear in the text. Having regard to the historical viewpoint and consistency with quotations, contemporary terminology and units are used throughout. Units were mostly based on the pound-foot-second system, though speed was usually given in miles per hour (mph).

## 1.0 INTRODUCTION

In February 1946, the decision to cancel the Miles M.52 project annulled the second of two significant initiatives made by the British aviation authorities around the beginning of the 1940s. If they had been part of a coordinated programme, these could have given an early indication of the full scope of a new era in aeronautics, that was opening with the introduction of turbine jet propulsion. The ending of this project, before its completion, has long been considered to be one of the most unfortunate decisions in the unfolding development of aviation[1,2].

The first initiative had been the order in 1939 to build the Gloster E28/39, Britain's first jet-propelled aircraft. Events surrounding this have been well summarised[3,4] and a historical review is given elsewhere of the basic decisions taken on its aerodynamic design[5]. From the outset it had been realised that the new type of engine, to be provided by the company Power Jets Ltd, founded by Frank Whittle in 1937, would be able to take this aircraft into the hitherto unexplored region of high-subsonic flight. Through the collaboration that ensued between Gloster's Chief Designer, (Wilfred) George Carter and his small team and researchers at the Royal Aircraft Establishment (RAE) and the National Physical Laboratory (NPL), the regular use of the Mach number would pass out of the research centres into the industry. In level flight, the E28 could reach a Mach number of about 0·75, at a time when contemporary front-line fighters were barely reaching 0·5.

This was possible because the turbojet engine would evade the growing penalties accompanying the supercharging of piston engines, and the fall-off in efficiency of propellers as the blade tip speed approached the speed of sound. The E28 tests were confirming Whittle's calculations that the performance of the jet engine would be improved with altitude, due to the beneficial effect of the falling air temperature on the cycle efficiency, and with speed, due to the further enhancement caused by ram compression in the intake. The balance of forces was leading to higher speeds and lower fuel consumption as altitude increased, with the benefit of much reduced installed power-plant weight.

Although at this time the E28 was the only British aircraft showing this enhanced performance, and was being powered by one-off experimental engines, both Whittle and Carter were confidently looking forward to further advances. With best performance now occurring in the stratosphere, where the speed of sound was at its lowest, the next step would surely be into supersonic flight. Speaking at an RAeS meeting in 1937, Whittle had criticised those who considered the speed of sound to be an insuperable barrier to flight[6], and Carter had later written that it would be a natural progression for aviation to pass through what he called the 'sonic gateway' into this regime[4].

Few were yet ready to share this optimism, and so the second initiative was an unexpected and ambitious one, when in the autumn of 1943 the decision was made to commission a manned aircraft capable of flight at 1,000mph, or about M = 1·5 in the stratosphere. An instruction to devise a prototype machine to this requirement was placed with the company of Phillips & Powis Aircraft Ltd (Miles Aircraft Ltd from 5 October), with a turbine power plant to be provided by Power Jets Ltd. There was no experience of the conditions that would be encountered in flight at speeds beyond those being reached by the Gloster E28. Yet through an appraisal of what might be expected there, a design quickly evolved, of a form that was radically different from anything previously imagined in Britain. This is illustrated by the model shown in Fig. 1. Original features

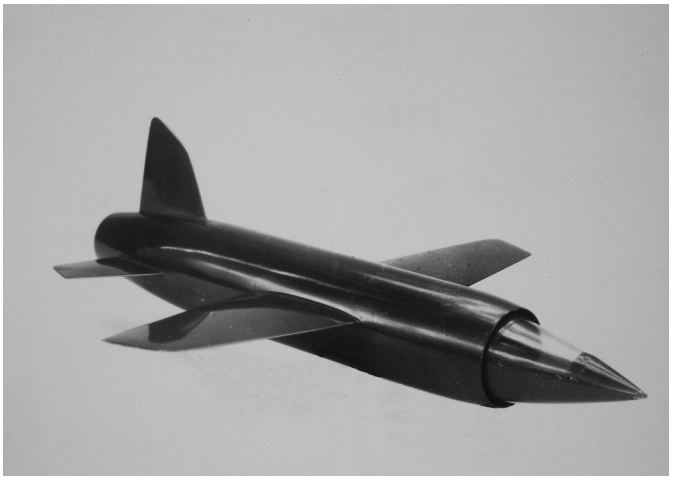


Figure 1. Model of the Miles aircraft to Specification E.24/43 (FAST Museum).

brought into play included the sharply-pointed ogival nose; the embedded cockpit, enclosed within a pressurised, jettisonable capsule ; the annular air intake, with boundary-layer bleed; the very thin, sharp-edged wing and empennage, with angled tips; the unbalanced, power-assisted controls, with an all-moving tailplane and numerous other innovations internally.

The unhappy circumstances of the termination of the project have tended to obscure the boldness of the thinking that led to this format and the technical advances embodied in it. Supersonic flight would not become a regular occurrence for a decade, and it would be mainly brought about then by exploiting the properties of new types of wing planform. Yet the aerodynamic considerations that arose in the design of the M.52 were not an unproductive diversion. Not least, they focused the attention of key individuals onto a regime of flight that had previously been unconsidered. If they had not been allowed to languish, these emerging ideas might justly have been seen as an early and necessary contribution to the unfolding of aeronautics in Britain. Although the events surrounding the aircraft have attracted much comment subsequently, its aerodynamic design is rarely considered in detail. For the present study, it is held to be of especial interest to examine how such a novel design evolved, when there had been no previous prospect of supersonic flight, and such information as might be relevant to that was dispersed and unco-ordinated.

## 2.0 THE ORIGIN OF THE PROJECT

### 2.1 Facts and fears

In 1943, though still comprising independent companies, the aircraft industry was being brought closer together by the exigencies of war. The pressures were very great, and work was necessarily focused narrowly on securing immediate returns, and on those developments with perceived time-scales that might have an influence on the war's outcome. There was no reason to consider flight at supersonic speeds, when it was hard enough, with the approach of the transonic region, even to secure small margins over the increasing performance of aircraft fielded by the enemy.

Research results and advice on emerging problems were provided impartially as needed by the staff of the Royal Aircraft Establishment, now greatly enlarged, as an arm of the Ministry of Aircraft Production (MAP), with backing from the National Physical Laboratory. The Aeronautical Research Committee (ARC), with membership drawn from the establishments, the industry and those

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of the research community still remaining in academia, provided an independent forum for reviewing novel ideas. Throughout the inter-war years it had been a vital channel for the dissemination of research information, through the publications of its Reports and Memoranda (R&M). As an independent advisory body to the aeronautical authorities, it was now very influential on the chances of a radical idea being taken forward.

Whittle's turbojet engine and its first airframe had received that support, and the Gloster E28 first flew in May 1941. Over the next four years, it proved to be an outstanding asset, both in verifying the advances in performance that would be achieved with a succession of engine developments, and in supporting research into flight in the high-subsonic region[5]. But it had been recognised that an aircraft with a single engine of the early Whittle type would not have the range and payload capacity required for the air defence role. And so the twin-engined Gloster Meteor had been ordered already in 1940, to become the first jet-propelled type to enter service with the RAF.

By 1943, intelligence sources were providing evidence that jet- and rocket-propelled aircraft were under development in Germany also. Then in May, material revealed by a German prisoner-of-war raised the fear that this work might be being extended into the supersonic regime[7]. References were made to work on propulsive ducts, known as athodyds (aerothermodynamic ducts, or ramjets, today), one example cited being identified as the 'Do. Geräte'. No specific results were forthcoming for these, but the processes involved in the operation of propulsive ducts were already well-known, showing that although they would be too inefficient to be of practical significance at low speeds, they might become of interest if speeds ever became high enough. The effort being put into their development seemed to indicate that this point had been reached. The prisoner had mentioned an aircraft with a cruising speed of 800mph and maximum speed of 1,000mph at 59,000ft. This was said to use mixed power-plant, consistent with the idea that these high speeds might have been obtained with something like the propulsive duct, having been supplemented by other types of engine at lower speeds. Although the prisoner had not himself been involved with aeronautics, this report was taken very seriously, as some of the other information that he had given was already known and verifiable. On the initiative of Ben Lockspeiser of the MAP, meetings of individuals from the RAE, NPL and relevant ministries were promptly convened to consider the implications of this report. The topic was given the highest security classification of the time – Most Secret.

The propulsive duct was an engine of the utmost simplicity, which had the potential of very low weight, and the ability to utilise a variety of fuels, but it could function effectively only when in motion at substantial speeds. Its potential as a supplementary power source had been reconsidered in 1940 by Arnold Hall and Hayne Constant of RAE[8]. This review showed that the propulsive efficiency would reach only a few percent at 500mph, but at 1,000mph the higher ram compression in the intake should increase the efficiency sufficiently to make supersonic flights 'of short duration' possible.

Three key papers, composed in May 1943, provide a background to the consideration by Lockspeiser's group. A new study undertaken by the RAE Engine Department provided some additional figures for the prospects of propulsive ducts if supersonic flight could be achieved[9]. At 1,000mph, it was confirmed that a propulsive efficiency similar to that of conventional power plants at subsonic speed might be possible, and this would improve progressively at higher speeds, due to the growing pressure rise behind the shock wave that would form around the intake.

In a paper by Ernest Relf of the NPL an outline was given of the potential performance of a generic type of supersonic aircraft[10]. In this initial study, the effects of a fuselage had been neglected, so the results would really be applicable only to an all-wing aircraft, but the main interest was on the use of the linear theory of aerodynamics in supersonic flow, which had been published in the inter-war years by Jakob Ackeret of Zürich[11]. It was shown by this that a wing of very thin section would be needed to give a reasonable lift/drag ratio, but that this could be maintained over a significant speed range. Further, for an aircraft that was small and light, the thrust requirement would not be prohibitive, probably being within the reach of the propulsive duct, and perhaps even the emerging jet engine. However, warning that many areas were not well understood, Relf mentioned specifically that, on the aerodynamic side:

- the effect of aspect ratio on wing characteristics in supersonic flight was obscure, though was 'possibly unimportant'. If that was so, the making of wings with very low thickness ratio might become easier, through the larger chordwise dimensions that would be allowable when low aspect ratios could be used,
- the transition from subsonic to supersonic flight could give problems of stability and control. A major factor was that the aerodynamic centre of the wing was predicted to move substantially, from around the quarter-chord point in subsonic flight towards the mid-chord position when the flow was supersonic. There might be other problems in this region that were currently unknown;

and on the propulsion side:

- differences between the requirements at low and high speeds would be likely to mean that different methods of propulsion would be needed for take-off and climb on the one hand and for acceleration and supersonic flight on the other; and
- since the propulsive jet stream would have to be supersonic in the high-speed condition, a convergent/divergent propelling nozzle would be required at the exhaust, and the shape of that would probably have to be variable in flight.

A complementary study was provided by staff of the Aerodynamics (Aero) Department at RAE[12]. Wings were again characterised by the Ackeret theory and thrust provided by propulsive ducts, as represented in the Engine Department report but with some corrections for losses that had not been considered there. This time, an estimate of fuselage drag was included, based on an interpretation of known figures for the resistance of cones, of a form that might represent a possible nose shape.

This paper showed how the boundaries of aviation would have to expand if supersonic flight was achieved. It spoke of an aircraft of which half of its take-off weight of 20,000lb was the fuel load, having a range of 1,500 miles, with cruise at an optimum height of 100,000ft. If the supposed behaviour of a type of biplane proposed by Adolf Busemann was assumed, an increased range to 3,000 miles was indicated, at 88,000ft. But the conclusions echoed some of the warnings of the NPL report, with comments such as that

- there were 'at present no facilities in England for supersonic work of this kind on aircraft',
- no consideration could be given at the time to 'the difficult problem of transition from subsonic to supersonic speeds', and
- this might have to be achieved by use of 'auxiliary engines (or alternatively towing) in the first stage of the flight' to get to a fair altitude (about 30,000ft) at subsonic speed and then combining the available thrust with the component of the aircraft's weight in a dive to overcome the 'peak in drag coefficient which occurs just below M = 1'.

With these preliminary views to hand, the meeting of Lockspeiser's group on 4 June decided that a standing committee should be formed, and an action programme formulated. The driving force behind this decision was stated explicitly – that 'We in this country had not studied the problems of aircraft flight at supersonic speeds,

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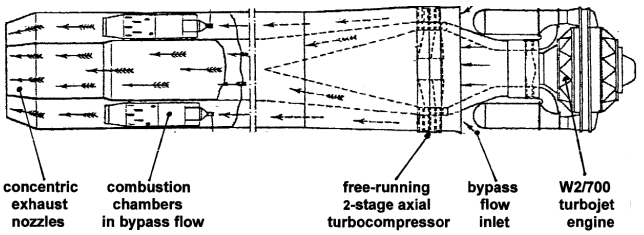



Figure 2. Power Jets Ltd sketch of the M.52 power plant based on the W2/700 turbojet engine with the No 4 Design Thrust Augmenter (The Miles Aircraft Collection).

but it was clear that there were many difficult problems, and if Germany had, in fact, solved them, then that country was years ahead of Great Britain and the USA'[7].

### 2.2 An experimental aircraft

The sense of urgency raised by fears about this issue is shown by the decision to hold Progress Meetings monthly, despite the pressures of other work. Committee papers were headed 'Problems of Supersonic Flight', and although the group which met to discuss them never had a formal title, it was inevitably referred to as the Supersonics Committee. Over time, its membership grew in an ad hoc manner, becoming formidable as leading specialists from the research establishments and universities were drawn in. Lockspeiser wrote subsequently that he had tried to seek out 'the best brains in the country, irrespective of where they come from.' Only a few figures from industry were invited to join. Remarkably, Miles was not represented, and in fact seems not to have been told of the existence of the Committee throughout the duration of the project.

Frank Whittle was a member from an early stage. In addition to encouraging some proposed work on propulsive ducts, he told the Committee about the 'augmented' jet engine, on which he was working. In his patent of 1936, a bypass flow had been added to that of a core engine to increase the total mass flow-rate and to moderate the overall velocity increase by merging the two streams at the exhaust nozzle. This would produce an increase in the propulsive efficiency, by optimising the overall change in momentum flux. But adding further combustion in the bypass duct (which had also been included in the patent), could provide a large temporary increase in thrust when needed. This might now be required when passing through the transonic region. A simplified sketch of the arrangement as later transmitted to Miles is shown in Fig. 2. The possibility of applying this variant of turbojet propulsion gained ground with the receipt of a paper by Alan (A.A.) Griffith, now at Rolls-Royce, showing that he had studied a specific power plant of this kind, in which the bypass mass flow-rate was four times that of the core engine[7]. But, as with the general approach in the Engine Department at RAE, Griffith thought that the core engine would have to be of the axial compressor type. It was concluded that 'such a power plant is admittedly some years away, though perhaps no more so than the supersonic aeroplane itself''.

However, in the discussions of the Committee, ideas about procuring a possible experimental supersonic aircraft were already converging. RAE tabled some sketches of possible configurations, and a Whittle augmented engine, with a second combustion stage in the bypass flow, began to be seen as the preferred option for propulsion. There were cogent reasons for this. Though heavier and more elaborate than a propulsive duct might have been, in this plant the core engine would give sufficient thrust at low speeds to obviate the need for auxiliary means for take-off and initial climb. It would not require the development of unfamiliar technology, as the additional components required for the augmenter would largely resemble those that had been researched for existing designs. The series of Whittle engines that had been installed in the Gloster E28 had proved to be reliable and their performance very close to predictions[5]. A bypass version of the W2/500 engine, with an aft fan integral with the power turbine, had been running for test purposes[13]. Moreover, Whittle was already working with thrust augmenters, having design principles set out in further patents of 1940 and 1942. No other turbine power plant of comparable capability was in prospect at the time.

At the November (7th) meeting of the Committee there was a reference to 'the Miles high speed aircraft that had been proposed'[7], seeming to indicate that the suggestion had come from Miles. Although the firm had some time earlier been in contact with Power Jets, where supersonic flight was regarded as inevitable, the proposal had arisen from its response to an outline specification for a 'very high speed experimental aeroplane', put forward by MAP to Mr F.G. Miles at a meeting on 9 October. The requirement was very brief, essentially setting a target performance of a maximum level speed of 1,000mph at an altitude of 36,000ft. This height was effectively that of the tropopause, at which the speed of sound became stabilised at its lowest value in the accessible atmosphere, of about 660mph. A small monoplane aircraft was envisaged, of all-up weight 5,000 to 6,000lb, with sufficient fuel to climb to 40,000ft and cruise for ½ hour at 700mph. Little else was specified, except that it would need to have an all-moving tailplane, as a means of dealing with trim changes in the transonic region, where conventional elevators were already proving to be ineffective, when used on existing aircraft in the pull-out from steep dives.

Thus it came about that, at the eighth meeting, within half a year of the first discussion having taken place, a decision was taken to proceed with a research aircraft that would be capable of flight at more than double the top speed of contemporary fighters, and well beyond the speed of sound[7]. A further twist to this remarkable requirement was that the machine was to be ready within nine months.

## 3.0 MILES AND POWER JETS

The order to begin work did not go out to competitive tender. The MAP was co-ordinating work throughout the industry, and could assess factors such as a company's available facilities and the workload, and the proven capability of its chief designer and project office staff for generating innovative ideas and seeing them into fulfilment. The contract for the E28/39 had been placed on the Gloster Aircraft Company on a basis such as this[3]. But after the E24/43 project had run its course and become generally known, there was some surprise that it had been Miles Aircraft Ltd, associated by most people with the manufacture of training aeroplanes, that had been chosen to realise something so far ahead of contemporary knowledge and experience. It proved, however, to be a sound choice.

The leading personalities of the company were the Miles brothers, Frederick (usually known as 'FG') and George. Having learned to fly in the 1920s, FG founded aero clubs and flying schools, and formed small companies to build aircraft of his own design, for the private pilots of the time. In 1932 he went into partnership with the firm of Phillips and Powis Ltd, to build light aircraft at their site at Woodley, near Reading. Later he became Technical Director and Chief Designer to the firm, joined in design work by his wife Maxine (generally known as 'Blossom'). Their single-engined Hawk aircraft found a ready market, and this and succeeding versions were notably successful in the air races and long-distance flights that were popular at the time[14,15]. Having obtained official Design Approval for the firm, Miles developed the Magister primary training aircraft for the RAF Reserve, followed, as war approached, by the Master intermediate trainer. These, with some other types, became the company's principal wartime contribution, of more than 5,000 aircraft, on which a large number of RAF flying personnel were to graduate. In 1941 FG acquired control of the company and reconstituted it in a

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form that was to become Miles Aircraft Ltd, with his brother George as Technical Director. 'Blossom' Miles continued her involvement in design, and they were joined by able newcomers such as Dennis Bancroft and Don Brown, with others on the staff who had been with FG since earlier times.

The Miles team possessed much energy and determination, and showed remarkable fertility of invention and innovation. They submitted a stream of original proposals for systems and projects to the Ministries responsible, from the commencement of rearmament until well after the end of the war. These were often ahead of their time, proposals being rejected, only to see the need for them conceded later. They designed and built their own wind tunnel, but FG, especially, was always keen to get new ideas tested at full scale in the air, and twice the Company was reprimanded for having built and flown experimental machines without first obtaining permission to commit the necessary materials to manufacture them. Both FG and George were experienced pilots, and often made the first flights of experimental and prototype aircraft designed by the firm.

It might be said that so little was known at the time about the situations that the requirements for E.24/43 would throw up, that Miles were as well placed as any other company to take it on. But they also had the advantage of a long history of close involvement with RAE, and their qualities seem to have been respected there. In 1935 this was marked by their provision of the single-engined Miles Hawcon aircraft, (intermediate between the Hawk and Falcon), that was purpose-built for in-flight research. It could be fitted with a series of interchangeable wings, for the investigation of the effects on the aerodynamics of variables such as aspect ratio and taper ratio[16]. This was continued later with the supply of a modified Miles Falcon Six, with which the effects of wing thickness ratio were investigated, up to the high figure of 30%[17]. In 1938, the tiny Miles Hobby, built for King's Cup racing, was sent to the RAE for testing at full scale in the 24ft wind tunnel. This allowed the characteristics measured in the tunnel to be compared with those obtained in flight on the same airframe at the same Reynolds number, throwing light on corrections required for the differences in factors such as the turbulence levels in the two situations. Other co-operative work was continuing in 1942, involving flight research on boundary layer control by suction, the use of spoilers for lateral control and high-lift devices, culminating in a combination which achieved the notable value of 3·2 for the maximum $C_L$[18,19]. The Company also gained a reputation for rapid response to a need, exemplified by the design, construction and first flight of the prototype M.20/1 fighter in just 65 days.

As the firm made the sometimes arduous transformation into the large-scale serial production required in wartime, the flow of innovative ideas continued[15]. The Experimental Department was strengthened, the wind-tunnel was in service and the Miles Aeronautical Technical School, later to count several significant figures in aviation among its alumni, was flourishing. Miles had been approached by Power Jets with a view to possible collaboration, and a visit to its site at Rugby was made by George Miles and Don Brown in November 1942[20]. There they viewed the latest engine developments and brought away details of the dimensions, weights and performance, which began to colour their ideas for new projects. For Power Jets, Flt Lt W.E.P. Johnson proposed at that stage to approach MAP for permission to effect a working agreement with Miles. A second visit was made in June 1943[20].

Though, by its general approach, the Miles team was temperamentally well-suited to take on the novelty of the supersonic project, it would also have to get along with the formidable figure of Frank Whittle. As his company was contracted to provide the power plant, Whittle took a close interest in the requirements for meeting the E.24/43 specification. At the first discussion on the project with Miles representatives at Power Jets on 17 October 1943 (wrongly minuted by PJ as 17 September)[21], he offered some typically bold suggestions on the design of the aircraft. The Miles team had more conservative ideas and were 'silently horrified' at Whittle's proposal for use of a biplane or multiplane format, with the tips joined as a means of obtaining sufficient structural support to allow wings to be used having the very low thickness ratio that would be needed[22]. But on a more practical level, they were shown a W2/700 engine on test, and in the discussion were given estimates of the likely dimensions and weights of an installation with this engine plus the latest form of thrust augmenter.



Figure 3. Two-stage turbocompressor wheel for No 4 Augmenter (FAST Museum).

The No 4 Design Augmenter (sometimes Augmentor) that Whittle devised for the W2/700 power plant followed experience with the No 3 version that had already been run with a modified W2/500 engine. This now had a free-running two-stage turbine rotor, driven by the core engine exhaust, carrying a ring of compressor blades outboard of the turbine blades on each disc (Fig. 3). With these outer blades, the bypass air was compressed before being ducted towards a further set of combustion chambers, from which the hot products expanded out through an annular exit surrounding the central nozzle of the core engine. The general arrangement as shown to Miles is reproduced in Fig 2. The overall thrust obtained at 36,000ft was estimated at this point to be 1,620lb at 500mph, rising to 4,100lb if the projected speed of 1,000mph was achieved[13]. Provision was considered also for the use of a brief period of reheat in the core engine exhaust, should that be needed.

This arrangement of the propulsion system for the aircraft was to remain throughout the project. But a fundamental change to the circumstances of Power Jets took place early in 1944, with the policy decision to found the National Gas Turbine Establishment, after which Whittle's influence on gas turbine development would inevitably diminish. As a first step, the assets of Power Jets would be bought out, and it would be formed into a separate nationalised company, Power Jets (R&D) Ltd, operating on its existing site at Lutterworth. For aeronautical applications at least, it would not in future be allowed to design and build complete engines, as this would be regarded by the burgeoning turbine engine industry to be state-sponsored competition. Though designs and test work for several further engines had been in hand, the W2/700 would, in effect, be the last complete aircraft engine of the Whittle era. The autobiographical account of Whittle's work was in most respects highly detailed, but the supersonic project and the augmented W2/700 power plant were mentioned there only in a brief footnote[23].

## 4.0 PRELIMINARIES

When the contract to produce the aircraft to Specification E.24/43 was placed, the project acquired the firm's designation M.52, which will be used hereafter. Within the small team, headed by Leslie

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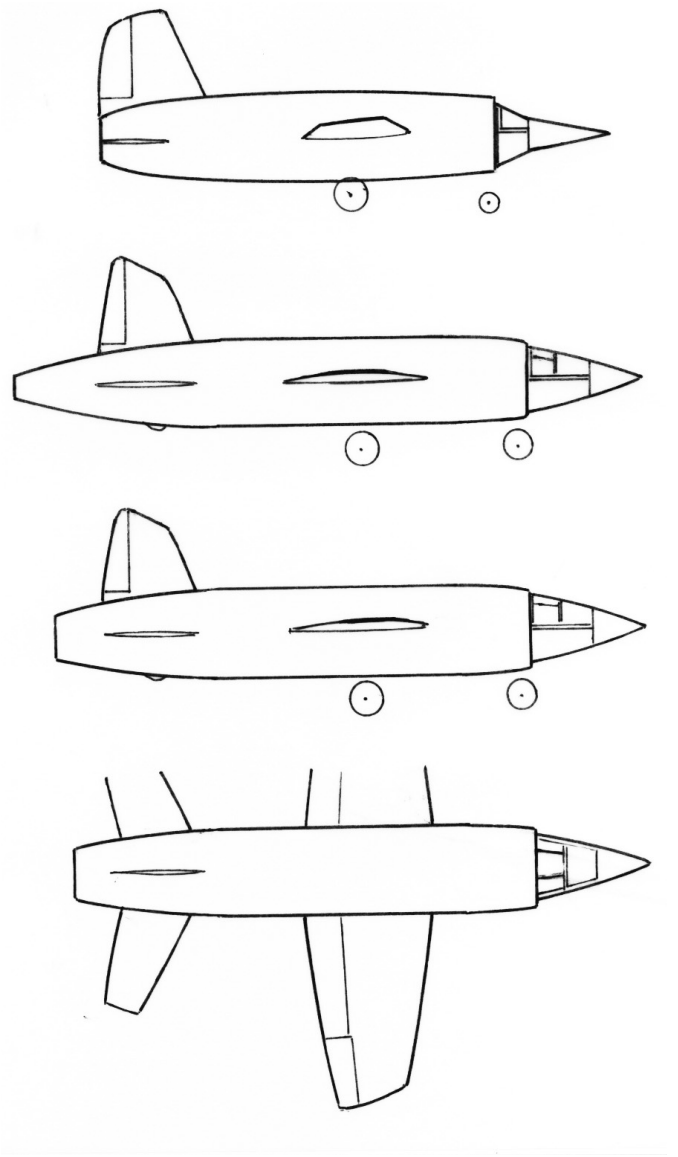


Figure 4. Evolution of the Miles M.52. Earliest known format (top) and final designs, with W2/700 engine only (upper) and addition of the No 4 Augmenter (lower)

(Redrawn by author from Miles Aircraft Ltd drawings – The National Archives).

('Toby') Heal, set up to undertake the design, the contribution of Dennis Bancroft, the company's Chief Aerodynamicist, would be crucial. To reach the specified performance, the aircraft would have to operate in a stable and controllable manner in three distinct regimes of flight – subsonic, transonic and supersonic. Though these formed an overlapping sequence, each would present very different aerodynamic challenges. It was not obvious that the characteristics needed for them could be combined satisfactorily in a single airframe.

Designing aircraft was always at the leading edge of technological advance. Although for the most part, designers could rely on the accumulated theory and experience of the past, they would expect that there would be new areas of uncertainty in any project. These would be addressed by seeking out information and advice from persons with expertise in those areas, by trial manufacture and rig testing, and by observation and measurement in wind tunnels and in flight. This systematic process was now followed in the case of the M.52. The teams from Miles and Power Jets set out at once to consult with researchers about the current state of knowledge that could be applied to the project. Those contacted were in the aerodynamics departments at the RAE and the NPL, and subsequently at the Armament Research Department (ARD) of the War Office at Sevenoaks, where the country's leading expertise in ballistics was located. But theirs was a radically new requirement, that no-one had imagined to be imminent, involving the first advances into two new regimes of flight. They were to find that there was no consolidated body of knowledge to which they could refer. Relevant information was sparse and scattered, and awareness of it was confined to a mere handful of people, among whom a consensus of opinion had not yet emerged.

Despite these uncertainties, it is clear from the record that Miles quickly appreciated the main features of what could be learnt from the specialists, and shrewdly incorporated these into an Outline design. Figure 4 includes a profile of the earliest general arrangement to be found, dating from 7 January 1944, about three months after the initial approach by MAP[24]. It had been decided to adopt an orthodox arrangement of fuselage, wings and empennage, as this familiar layout would 'permit more accurate design analysis' and allow more readily for the 'errors in assumptions' that would be inevitable in the sparse state of knowledge about the novel conditions in which it would operate[25]. As first conceived, the aircraft would have a short, tubby fuselage, with proportions resembling those of a projectile, fitted with unswept wings of uniform thickness ratio 7·5%. The lifting surfaces all had straight edges, except for the trailing edge of the rudder, and the tips of all surfaces were cut back, such that the trailing edges were shorter than the leading edges. The pilot's cabin was enclosed within the nose, surrounded by an annular air intake, and terminating forward in a slender conical spike.

Refinements would be required over time, with some significant changes of detail, but the basic arrangement would remain. Figure 4 shows also the two forms of the Final design[24]. A Preliminary version, with changes to the nose profile and the lifting surfaces, came between these. The rear fuselage was lengthened, resulting in a greater fineness ratio, more suited to the subsonic part of the flight, and better closures towards the exhaust outlets for two different power plant arrangments. The position of the empennage was unchanged. The wing, now tapering to a thickness ratio of only 4% at the tips, had an effective average ratio of 6·2%. The tailplane was swept back, and the shape of the cut-off of the wings and fin was reversed, so as to make the trailing edges the longer ones. Two flying prototypes were to be built, externally differing mainly at the aft end, where the rear fuselage was shorter and wider when accommodating the outer nozzle of the bypass duct of the augmenter. The first, powered by the W2/700 engine alone, would undertake handling tests in the take-off and landing phases and flight up to high subsonic speeds. The second prototype, fitted with the thrust augmenter and perhaps additional reheat, would be expected to pass through the transonic region, in a shallow dive if necessary, and onwards to achieve the design speed of 1,000 mph.

The leading dimensions of the M52s, in their final form, are given in Table 1. Details of the design decisions that led to their strikingly forward-looking shape are considered in Section 6. A brief review is first given of the background of the state of relevant knowledge around the time of their conception. The significant further developments in understanding that took place during the life of the project in the next two years are included in the description of the design.

## 5.0 THE KNOWLEDGE BACKGROUND

### 5.1 The subsonic region

A critical decision, taken very early in the concept stage, was that the M.52 should be able to take off and land normally. Launching at altitude from another aircraft was considered briefly, as this could reduce the fuel load to be carried, by not having to make provision for take-off, initial acceleration and climb-to-height[14]. The weight of the undercarriage might be saved also, if a satisfactory skid

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arrangement could be devised for landing. The decision to go for conventional take-off and landing would enable a much more extensive flying programme to be undertaken, but would also be in line with the character of the Miles brothers. They had been flying throughout their adult lives, and would be likely to favour the test pilot's cautious exploration of the flight characteristics of this new type of aircraft, from a beginning with the first taxying trials. A more gradual approach also had the best chance of being able to show that, despite the high performance of a supersonic aircraft, it could have acceptable characteristics in the ordinary circumstances of take-off, landing and low speed flight.

Design principles for these subsonic parts of the flight envelope were well established, and need not be reviewed here in any detail. The layout of the M.52 was basically conventional, and for the most part it would have been no more difficult to ensure that it was stable and controllable throughout this regime than for other contemporary types. However, some design features, dictated by the requirements of other phases of the flight, would react on the low-speed behaviour. In particular, the wings and empennage had to have sections that were much thinner than those in current practice, with sharp leading edges and cut-off tips.

There had been experience with thin sections in the early days of flight, these having been modelled on the wings of birds. More recently, the Gloster IV entry for the 1927 Schneider Trophy contest had used a section, based on the RAF 25 profile of thickness ratio 6·14%, but given sharp leading edges[26]. At that time, there had been concerns that separation would develop at the leading edge for anything more than trivial incidence (angle-of-attack), giving high profile drag and an early stall, and these matters were revisited thoroughly for the M.52. There was nothing on record on which to base expectations about how the profile drag would develop, how the induced drag would be affected by the unusual tip shape, or on the form of the high-lift devices that could be used to allow reasonable speeds to be used on take-off and landing. Though the earliest wind-tunnel tests for the project made by Miles were to investigate the low-speed characteristics of the wing, and more extensive ones were made later, it was decided at an early stage that the chosen wing form should be tested at full scale in flight. It had long been F.G. Miles's conviction that 'Full-scale tests should precede, run parallel with, and follow all other forms of research'[15]. And so, the Falcon M3B aircraft, which had already been flown with different sets of wings for experimental work at RAE, was returned to the Company, where it was fitted with wings and tailplane that replicated the form of those planned for the M.52. Outlines of the wind-tunnel and flight tests and their results are given later, in Section 6.

The basic requirements for an aircraft to be stable in flight had been established in the earliest days, and were formalised in the extensive work of Hermann Glauert and Sidney Gates at RAE in the inter-war period. Summaries of the theory and methods for obtaining the relevant derivatives for a rigid aeroplane were set out quite fully in the textbooks of the late 1930s[27-30]. These were sufficient to ensure that the relevant moments would tend to restore steady fight after a disturbance, but methods for the analysis of the dynamic behaviour, particularly as it would affect handling, were still under investigation. Aeroelasticity was becoming a major component of research, and at the practical level the flexibility of parts of the airframe could be taken into account to some extent, in the calculation of quasi-static quantities such as the speeds for lifting surface divergence and control reversal. Dynamical effects like control surface flutter were contained largely by mass-balancing and a range of empirical treatments acquired during flight testing.

**Table 1**
**Miles M.52 Principal dimensions and data**

Miles Aircraft Ltd drawing refs: D.5200000, D.5299962, D.5231008
(D.S. Bancroft, private communication)

| | | | |
|---|---|---|---|
| All-up weights | (18in duct and 200gals fuel) | | 6,763lb |
| | (37in duct and 193gals. fuel) | | 7,812lb |
| Fuselage | | length | 35·6ft (18in duct) |
| | | | 33·6ft (37in duct) |
| max diameter | | | 5·0ft |
| slenderness | | | 0·14 |
| wetted area | | | 443ft$^2$ |
| Wing | | gross area | 142ft$^2$ |
| | | nett area | 108·7ft$^2$ |
| | | aileron area | 4.86ft$^2$ each |
| | | span | 26·875ft |
| | | mean chord | 5·28ft |
| | | aspect ratio | 5·09 |
| | | root section | 7·5% biconvex |
| | | tip section | 4·02% biconvex |
| | | dihedral | 2° |
| Tailplane | | gross area | 53·5ft$^2$ |
| | | nett area | 31·2ft$^2$ |
| | | span | 14·25ft |
| | | mean chord | 3·49ft |
| | | aspect ratio | 4·16 |
| | | root section | 6% biconvex |
| | | tip section | 4·25% |
| Fin and rudder | | total area | 18·5ft$^2$ |
| | | rudder area | 6·3ft$^2$ |
| | | height | 4·58ft mid chord |
| | | mean chord | 4·04ft |
| | | aspect ratio | 1·13 |
| | | root section | 7·5% biconvex |
| | | tip section | 4·48% biconvex |

## 5.2 The transonic region

What later became known as the transonic region began at speeds where compressibility of the air caused departures from the familiar low-speed aerodynamics to become significant, especially as they affected structural load cases and flight safety. Problems were being encountered in power dives by front-line fighter aircraft, when heavy buffeting and difficulty (sometimes impossibility) of recovery was experienced, due to large changes of trim. But the first relevant investigations had been prompted much earlier by concerns about the fall-off of efficiency of propellers in this regime[31]. Results were reported in NACA papers from 1922 to 1931, at speeds up to M = 1·5 by Sylvanus Reed[32], and up to 1·08 by Lyman Briggs and Hugh Dryden[33,34]. In Britain, tests were made up to M = 1·3 by George Douglas, William Perring and others from 1923, of which those of 1927 and 1928 are particularly relevant here[35-41]. An example from the latter, showing lift and drag coefficients for a 10% thick aerofoil with a biconvex section, is shown in Fig. 5[37]. Results such as these clearly revealed the characteristics that were to be troublesome with wings when flight entered the transonic region, more than a decade later. For a given incidence, the lift coefficient rose at first as the speed increased, and then fell rapidly after passing through a maximum (in this case at around M = 0·7). This collapse was known in Britain as the 'shock stall', by analogy with the loss of lift at the limiting incidence experienced at lower speeds. This was accompanied by a steep increase in the drag coefficient, reaching a peak shortly after M = 1. The strong influence of the incidence on the onset and extent of these effects was also noted.

In 1930, Glauert showed that the initial rise in lift coefficient was to be expected, due to a progressive enhancement of pressure coefficients when compressibility of the air is taken into account[42]. A coefficient at Mach number M would be raised, relative to its value

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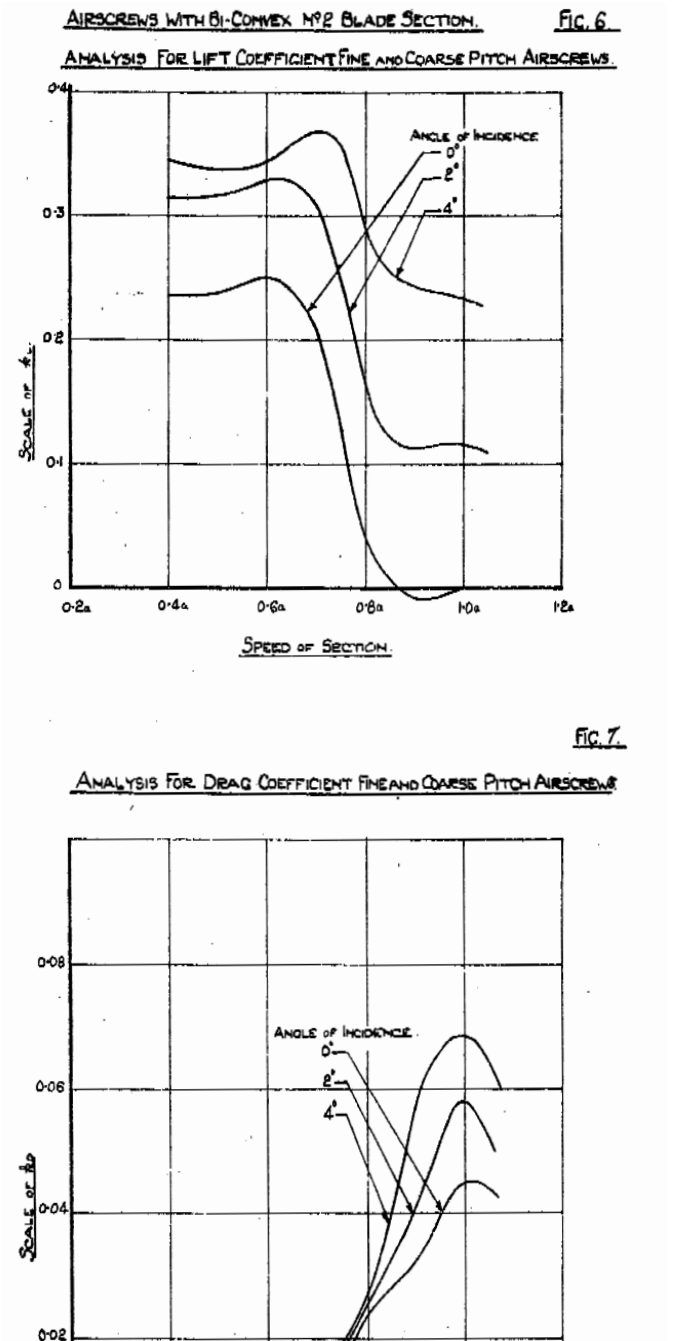



Figure 5. Lift and drag coefficients of biconvex section, 10% thickness ratio[37] (note $a$ = sonic speed ; $k_L = C_L/2$ ; $k_D = C_D/2$).

in incompressible flow, by a factor ($G$ here), later known as the Prandtl-Glauert factor, through having been found earlier, but not published, by Ludwig Prandtl :

$$G = 1/(1 - M^2)^{1/2} \quad \ldots (1)$$

This would apply to the lift and moment coefficients, though in inviscid theory there could be no effect on the drag coefficient. Clearly, the prediction would break down on approaching M = 1, where $G$ would tend to infinity.

Work at NACA, reported in 1935 by Eastman Jacobs at the 5th Volta Congress in Rome, illustrated what was happening in the earliest stages of the collapse of lift[43]. While the forward speed was less than M = 1, the flow over the section could reach sonic velocity locally in regions of low pressure, and shock waves were likely to appear there. The high local pressure rise at the foot of the shock could cause the boundary layer flow to separate, producing an increase in drag, reflected mainly in the losses in the wider wake. There was a general redistribution of pressure around the section. This resulted in an overall loss of lift and a rearward movement of the aerodynamic centre. It was at first taken for granted that these effects would occur when the local velocity at some point first reached the speed of sound, defining the critical forward speed or Mach number.

The work on propellers in Britain and in the United States had shown empirically that the onset of these effects could be delayed by using blade cross-sections that had small thickness ratios and sharp edges. It was now necessary to reconcile these requirements with current practice for wing sections, for which theory had guided development in the opposite direction, towards aerofoils with substantial thickness ratios and rounded leading edges. These forms had provided the principal means of obtaining large values of lift/drag ratio, with a wide working range of incidence. The main thrust of work throughout the late 1930s and 1940s then became the search for new aerofoil sections that would maintain good lift characteristics while delaying to higher forward speeds the first appearance of sonic velocity on the section.

An influential paper to the RAeS in 1937 by Christopher Lock of the NPL summarised the position at that time[6]. Using a procedure developed by his colleague Ernest Relf to determine the critical speed, he showed the extent to which it was delayed through limiting the disturbance of the flow by using thin sections. He also demonstrated that a body of revolution would have a higher critical speed than an aerofoil of the same profile. This was due to the greater freedom for the flow to expand in the third dimension, as it moved over the body, noted in potential flows, and later known as '3D relief'. In Lock's predictions, the pressure coefficients were first determined from incompressible flow theory, and then enhanced for the effects of compressibility by applying the Prandtl-Glauert factor. Numerous refinements were developed subsequently around this theme, both for predicting the flow and for relating it to the free-stream Mach number. But in due course, it became clear that the forward speed could rise considerably above that at which sonic velocity occurred locally before shock waves appeared and the drag began to rise steeply. The point at which this happened would be known later as the drag-divergence speed.

There were indications that a rearward movement of the point of maximum thickness, relative to current practice, would be advantageous. Herbert Piercy at NPL had begun developing sections that had maximum thickness at the 40% chord position, giving more gradual acceleration of the air. The first section of this kind to come into use, designated EC, after the elliptic and cubic equations used to define the shape, was already flying on the Gloster E28/39 in 1943, and would shortly enter service on the F9/40 Meteor[3,5].

Progress was seriously hampered by the absence of facilities for systematic studies in the transonic regime. In a closed wind tunnel, the appearance of any local shock waves on the model caused this point to become a virtual throat, on which the flow became choked and the upstream Mach number could not then be increased[44]. For models of any reasonable scale, contemporary wind tunnel work was limited to speeds not much above M = 0·8. Attempts to obtain higher speeds through full-scale flight trials in steep dives were already experiencing limitations due to the onset of hazardous load fluctuations or excessive control forces to maintain trim and effect

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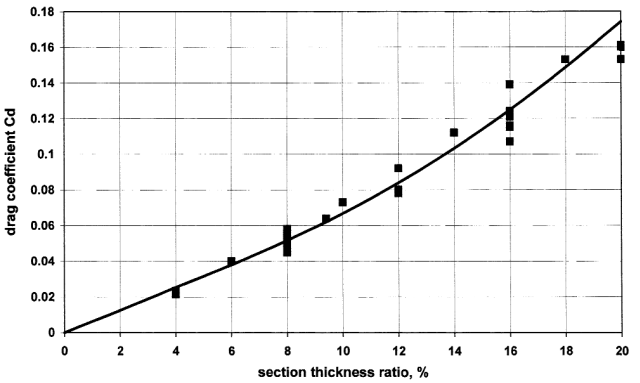



Figure 6. Drag coefficient of thin, sharp-nosed sections at M = 1·08, $C_L$ = 0·15[32-34].

recovery[45]. The E28, with thickness ratio of 12% at the wing root and 9% at the tip, ultimately reached about M = 0·82 in these trials, but a specially-prepared Spitfire XI unarmed photo-reconnaissance aircraft, with a wing of about the same thickness ratio, but of more slender form overall, had exceeded M = 0·9. Though systematic study could not be carried out in this way, these trials leant weight to suspicions that wing sections designed to delay the onset of the drag rise might be subject to a steeper rise, once it had begun.

Validation of linear theory had shown that there were good grounds for expecting wing drag coefficients to fall again in the supersonic regime. But just as subsonic theory included factors containing the term $1/(1 - M^2)^{1/2}$, as in Equation (1), so supersonic theory contained the term $1/(M^2 - 1)^{1/2}$, so that predictions were caused to rise to unrealistic levels in approaching M = 1 from either side. In the matter of drag, the general opinion was that there were no data that might indicate the values to be expected at the peak of the drag coefficient that must appear somewhere between high subsonic and low supersonic speed, or the Mach number at which that would occur. Without a theoretical foundation, or normal wind-tunnel work, the only source that might have been drawn upon at the time would be empirical evidence in the results of the earlier propeller tests at transonic speeds.

Results at speeds up to M = 1·08 had been obtained by Briggs and Dryden with a free jet, emerging from a convergent-divergent nozzle. Figure 6 shows a compilation of drag coefficients of aerofoils at this speed, extracted from their results for a lift coefficient of 0·15, that would be typical for the M.52 wing at around sonic speed at 36,000ft. The absolute values would be questionable, as their technique involved model sections of 1in chord, tested in a free jet of only 2in diameter. But the high internal consistency of the results had suggested that their relative values might give valid guidance for the selection of sections for good performance in propellers. Though obtained with many variations of shape, Fig. 6 shows the results to be coherent, and to display very distinct general features. It would be clear that the thickness ratio of the section continued to be the principal determinant of the drag in the vicinity of sonic velocity. Sections with sharp leading edges usually gave somewhat lower drag coefficients than those of the same thickness with more rounded nose shapes, but for the thinnest sections, the shape was less important, and thickness became the dominant factor there.

The results of Douglas, Perring and others were obtained from measurements of the change of linear and angular momentum flux of the air at various radii across the propeller disc. With the blades rotating in a large free stream, in principle the absolute values were more likely to be valid, and they were compatible with the measurements of thrust and torque, made at the same time. Their drag results were generally higher than those of Briggs and Dryden, but the tests were fewer in number, and confined to a narrow range of thickness ratio. They were also more scattered, so that a clear correlation was not evident. However, as propeller tests seem to be the only source of drag data for lifting surfaces available for the transonic region, more reference will be made to them later.

On a visit to RAE at the beginning of the project, the Miles team had been advised to expect a transonic peak value of '10 to 20 times' the low-speed drag coefficient[22]. The basis for that was not given, but it might have been recalled from the earlier work on propellers, with reference to sections with thickness ratios of 15% and above, that were current on contemporary aircraft. As it became clear that much thinner sections would be selected for low drag in the supersonic regime, RAE estimates of the transonic drag rise were in the range of five to six times, as reported later.

Uncertainty about the peak drag coefficient would remain until ways of exploring the drag rise systematically could be developed. Efforts were being made to find a basis for a new generation of wind tunnels, able to operate routinely in the transonic range, though little scope for this work could be found in Britain at this critical time. But a potentially important indication had been given in the 1939 translation of a paper by Antonio Ferri of the Guidonia research centre near Rome, which was in circulation as an NACA Technical Memorandum[46]. This had been presented originally to the Lilienthal-Gesellschaft in Berlin in October 1938. Ferri had reverted to using an open jet, arguing that the boundary conditions would be more appropriate than those of a fixed tunnel wall. The jet cross section was 450mm wide by 280mm high, which, when used with aerofoils of 40mm chord gave much better proportions than those of the Briggs and Dryden open-jet experiments. Measurements of lift, drag and moment were reported for a section of 6·25% thickness ratio up to M = 1. Ferri found the outcome to be 'quantitatively in agreement with that obtained for similar airfoils in propeller experiments', with specific reference to a presentation by Douglas at the 1935 Volta meeting. Moreover, his drag results for this thin section did not manifest the 'marked increment found elsewhere at M = 0.8 spoken of as a regular sound barrier'. This paper also reported some high-quality measurements of the pressure distribution around a thin aerofoil in the region up to sonic velocity, the only ones known to have been published at the time. The potential implications of Ferri's work will be considered later.

Progress in tunnel development continued in the US and the leading edge of transonic work was to remain there subsequently[43]. However, well aware of the lack of transonic data, the Supersonics Committee began to institute a supporting programme, part of which was intended specifically to address this deficiency.

### 5.3 The supporting programme.

The Committee had no budget allocation, and although it seems that its senior members would be able to carry some supplementary investigations within their own programmes, these would have to be fairly limited. Most of this work was intended to provide a better understanding of the performance of propulsive ducts, and those parts are not relevant to the present study. However, there is a certain historical interest in noting that Eastman Jacobs of NACA was invited to attend the Committee's meetings, while on a visit to Britain in 1943[7]. Though one of the world's leading experts on transonic aerodynamics, his main contribution on this occasion was to propose a test vehicle for small propulsive ducts, propelled by the standard British 3in solid-fuel rocket. This was intended to provide measurements of the drag coefficient of a typical duct, obtained mainly from optical and radar observations of the trajectory of the vehicle after burn-out of the propelling rocket. Duct shells were designed and made by Power Jets to Jacobs's requirements, and in due course some numbers of these were fired. The maximum speed reached was about M = 1·14, lower than had been hoped, but the expected drag peak had not appeared, and it was assumed that it had not been reached. A study concluded that higher Mach number could be obtained only by the addition of booster rockets. But the

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scatter of the test results was such that a very much larger programme would be needed, if the finer detail of the drag curve was to be revealed, and this part of the programme was discontinued.

In October 1943, Ronald McKinnon Wood of MAP proposed that aerodynamic characteristics of wings in transonic flow could be measured by attaching specimens to a torpedo-shaped vehicle that would be released from a parent aircraft at high speed and altitude, and a firmer proposal for a test vehicle along these lines was worked up by Ronald Smelt of the Aerodynamics Department of RAE[47]. Behind its ogival nose there was a long cylindrical section, to which the test specimen could be attached. It was suggested that a wing of 6ft span and 20in chord should be carried, so that its drag would be a reasonably high fraction of the total drag. Alternating measurements of dynamic pressure and acceleration would be transmitted by telemetry throughout the trajectory, which would be traversed in about 50s. As with the propulsive duct tests, the static pressure would be obtained from height measurements, taken by radio altimetry or from the ground by radar and kiné-theodolite tracking. Smelt considered that the likely accuracy of the derived drag values would be comparable with those of wind-tunnel testing at the time. A secondary objective of the proposal was to 'provide a free flight basis of comparison for tunnel methods' when the latter had been developed sufficiently to give reliable results at transonic speeds.

Through the British Air Commission in Washington, it was arranged that a programme of dropped-body tests should be mounted on a joint UK/US basis. Tests had a slow start in both countries, as it seems that unstable behaviour of trial vehicles prevented any consistent results from being obtained for a long time. It is reported that in US circles the British trials were being called the 'undropped body' tests[48]. This was not without justification, as with many delays, the first results were not presented to the Committee until August 1945, after the M.52 design had been finalised, and these were for the bare vehicle without any test specimen. A drag coefficient of about 1·0 on the projected area was indicated in the vicinity of M = 1[7].

Though Barnes Wallis of Vickers-Armstrong was later to be a member, there appears to have been no awareness in the Committee of the work he was doing at this time devising deep-penetration bombs, intended for the destruction of military installations and civil engineering works by producing shock waves in the ground. In May 1944, six prototypes of the first 4,000lb 'Tallboy' bombs were drop-tested at the Orfordness Research Station. Released from a Lancaster 1 bomber at 18,000ft and 190mph, these performed very consistently, reaching a terminal velocity of 3,300ft/s, or nearly M = 3, after a time of fall of only 34s[49]. Wallis had evidently arrived at a very low-drag shape. However, their more predictable trajectory than those of the other free-drop tests was probably due to the use of spin-stabilisation, which had been obtained by off-setting the tail fins at 5° to the axis. It is uncertain whether, without this feature, an adapted bomb of this form would have behaved suitably for use in aerodynamics research.

It was realised somewhat belatedly that methods of analysis and measured data relevant to the project existed within the Ordnance Committee (later Board) of the War Office, and their leading ballistician, James Maccoll, was invited to join the Supersonics Committee. Further, there were supersonic wind tunnels, the largest with a working section 11in square, in use for ballistics research at the NPL[50,51]. These were operated by the Engineering Division, and it had apparently not previously been considered that they would be of use in aeronautics research. It was agreed that the larger tunnel could now contribute to the programme on supersonic flight, though tests could be run only at discrete values of Mach number, since a separate convergent-divergent nozzle was required for each test condition. There were difficulties in procuring the castings required for making these. Having regard to all the circumstances, it was found that the lowest supersonic Mach number at which reliable data could be obtained was about 1·4. At speeds nearer to sonic, the bow wave, reflecting from the tunnel wall, would interfere with the flow around the model and could disturb the boundary layer. Even if, for a small model, the reflected wave passed behind it, effects on the wake could propagate upstream to affect the boundary layers when the wake flow was subsonic. Thus, with the tunnels then in use, there was still a transonic region, from about M = 0·85 to 1·25 or above, that remained out of reach.

In due course, the Committee caused another tunnel, of the blow-down type, to be constructed at RAE, operated by utilising the stored compressed air of the chamber that surrounded the (subsonic) High Speed Tunnel. And so, this part of the supporting programme had some influence in the development of supersonic tunnels in Britain. William Hilton, a leading wind-tunnel expert at NPL, asserted later that 'at the end of the War British supersonic tunnels were a match for anything the Germans had to show'[51], though this has not been generally accepted.

The small scale and fragmented nature of the supporting programme seems unworthy of its distinguished proponents, but has to be seen in relation to the conditions of the time. After four years of 'total war', every part of British society was at full stretch, with a scarcity of manpower and resources of all kinds, such that departures from the acknowledged priorities could occur only by diversion from some other urgent requirement. The work of the Supersonics Committee was directed by powerful advocates, occupying senior positions, who regularly offered to take matters requiring urgent inquiry into the programmes of their own establishments. However, if demands for supplies arose, no priority could be claimed, because the high security classification prevented the nature of the project from being disclosed. In consequence, the Minutes of the meetings frequently include reports of frustrating delays, due to unavoidable postponements and the scarcity of materials and manpower. It is plain that resources of a scale to match the aspirations of the Supersonics Committee just could not be mustered at that time, so that nothing had emerged that would assist in the design of the M.52. Probably the only action that would have lasting consequences was the improvisation that gave the RAE its first supersonic wind tunnel.

## 5.4 The supersonic region

### 5.4.1 Bodies

As projectiles were the one known instance of objects moving at transonic and supersonic speeds, there would inevitably be resort to the available information on these. It had been known since the work of Benjamin Robins in 1746 that the resistance they experienced from the air took on a different form at velocities beyond the speed of sound[52]. Ernst Mach had shown in the late 19th century that a detached shock formed ahead of projectiles at low supersonic speeds, and that the shock wave became attached at higher speeds if the nose of the body was pointed[43]. This was inclined to the direction of motion at what is now known as the Mach angle, given by

$$\mu = \mathrm{Sin}^{-1}\,(1/M) \qquad \ldots\ (2)$$

The nature of the flow around projectiles continued to be studied by ballisticians but, while flight aerodynamicists could see no relevance to aeronautics, work in the two fields continued almost independently over the inter-war period. Only a very few individuals, happily some of high stature, maintained a bridge between them.

Ballistics experts developed firing tunnel methods for observing the motion of bullets and shells at close quarters, and by the 1940s had determined the aerodynamic drag of bodies of various shapes over a wide range of speeds[53]. These required interpretation, since a mere assembly of drag measurements, such as could be collected at the time of the M.52 project, would show great variation. Some key results of drag coefficient that are relevant to this study are shown in Fig 7. In the definition used in aeronautics, this is based on the drag per unit frontal area. The corresponding quantity originally used in

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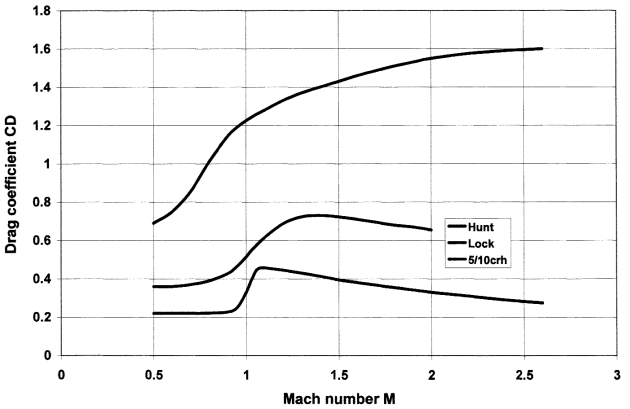



Figure 7. Drag coefficient of projectiles, frontal area basis, for various head shapes (top to bottom, flat-fronted[53]; rounded[6]; ogival[59].

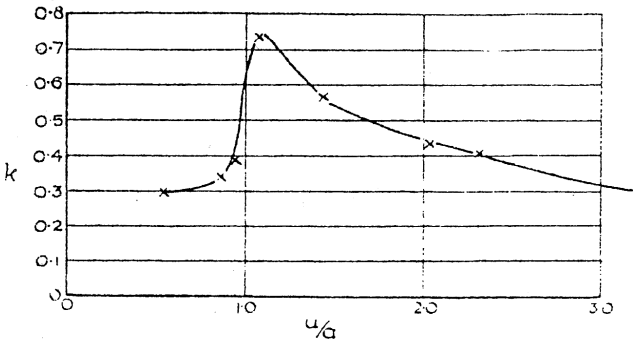



Figure 8. Resistance coefficient of sharp-nosed projectile[54] with permission, Royal Society. (note k = $\pi C_D/2$).

ballistics was the resistance coefficient, defined, for a projectile of maximum radius $r$ or diameter $d$, by

$$f_R = D/\rho V^2 r^2 \text{ or } D/\rho V^2 d^2 \quad \ldots \quad (3)$$

so that this must be multiplied by $2/\pi$ or $8/\pi$ respectively, to give the drag coefficient $C_D$, based on the frontal area. The principal difference in the results from different sources was that some rise to a sharp peak in the vicinity of M = 1, while others have a generally higher drag, which increases more gradually, and reaches a maximum at significantly higher speed. It would emerge that this was mainly due to differences in the shape of the nose, or 'head' of the projectiles.

The earliest supersonic wind-tunnel measurements made in Britain were those of Thomas Stanton at NPL[31]. Though not revealed until later for security reasons, he began operating in 1921 a tunnel of the induced-flow type, in which a series of convergent-divergent nozzles enabled test section speeds to be obtained that ultimately extended up to M = 3·25. Only tiny models could be used, to avoid interference effects, but the quality of Stanton's experimental technique was exceptional, and in 1924 he clearly showed the characteristic form of the drag curve for a sharp-nosed projectile, reproduced in Fig. 8[54].

The nature of the flow that gave rise to this form was clarified by work on the test case of a conical body in an axial supersonic stream. In a paper of 1933 (Geoffrey) G.I. Taylor and James Maccoll gave a method of calculating the pressure coefficients for cones and compared the values with measurements that had been made by Stanton in 1928[55]. With a nozzle insert that gave a Mach number of 1·7, tests were made with minuscule cones, of overall length 0·3 to 0·74in, and employing pressure tappings with dimensions as small as 0·0047in. The agreement with theory was excellent.

Maccoll later extended the range of these calculations and further revealed the nature of the supersonic flow field around bodies[56]. A drag coefficient falling with increasing Mach number was associated with an attached shock at the nose. For cones, the smaller the apex angle, the lower the Mach number at which the shock first became attached. The detached shock, which always formed immediately beyond sonic speed, being a normal shock, produced a large pressure rise in front of the nose and gave much higher drag. When, as in the upper curve of Fig. 7, the projectile has a flat nose, the shock remains detached, and the drag is very high and continues to rise at higher Mach numbers. The effects of using rounded and pointed noses were essentially the same as with aerofoils – the sharper the nose, both the initial drag rise on the subsonic side and the peak drag on the supersonic side moved closer towards M = 1.

In 1932, Theodore von Kármán and Norton Moore provided the first general method for calculating the total drag of a projectile, modelled as a slender body of revolution with a defined head profile and a cut-off base[57]. This used a process involving superposition of the flow from sources and sinks onto the free-stream flow, as originally devised for incompressible flow by William Rankine in the 19th Century. In their treatment, the equivalent of the sources became conical fields when the free stream was supersonic. Numerical integration of the resolved pressure distribution over the body could now give values for the supersonic drag for heads of any shape. The results for conical heads were confirmed by the work of Taylor and Maccoll to be in reasonable agreement with their exact solutions. von Kármán and Moore applied their method to a projectile with a head of ogival shape, showing that the drag was less than for a cone of the same mean half-angle at the nose, as noted in the results given by Hunt[53]. Also in this paper they gave the first attempts to estimate the contributions to the overall drag from other sources. In the case reported, the proportions of the overall drag of a projectile, obtained from the three sources of skin friction (boundary layer drag), head drag and base drag, were about 9, 43 and 47% respectively, in the vicinity of the peak at M = 1.

Elaborating this method, Hsue-Shen Tsien, of the same laboratory, later give results for lift and moment when the body was at incidence[58]. It was shown that the lift of slender bodies in supersonic flow was proportional to incidence at small angles, so that this and the associated stability derivatives could be calculated. The drag would remain approximately as given by the method of von Kármán and Moore.

The earliest ogival head shape for projectiles was that of the tangent ogive. This is formed by rotation around the body axis of a circular arc, beginning tangentially to the cylindrical part of the body, and intersecting the extended centreline to form the tip, giving a head length $L$. It is customary in ballistics to give lengths in units of the diameter of the body, or calibre, so a head shape formed by an arc of radius $R$ in calibres is designated an $R$ calibre radius head (Rcrh). The length $L$ of the head in calibres is then given by

$$L^2 = R - 0{\cdot}25 \quad \ldots \quad (4)$$

A more pointed head shape, with a lower drag than that of the tangent ogive, was used to define that of a new Standard Projectile, detailed in the 1940 Ballistic Tables of the War Office[59]. This is known as the 5/10crh, the notation $R_1/R_2$ now specifying two radii $R_1$ and $R_2$ in calibres, to construct the ogival form. The length of the head is defined by $R_1$, as in Equation (4), but the head shape is formed by an arc of the larger radius $R_2$, passing through the same beginning and ending points. This shape is called a secant ogive, from a function appearing in one of the trigonometrical expressions for defining it. These two head forms are shown to scale in Fig. 9. The smaller curvature of the secant ogive gives a more pointed head shape than for the corresponding tangent ogive, with a smaller angle at the tip. For the 5/10crh, the tip length remains 2·18 calibre, but the half-angle at the tip is 19·3º, compared with 25·8º for the tangent form.

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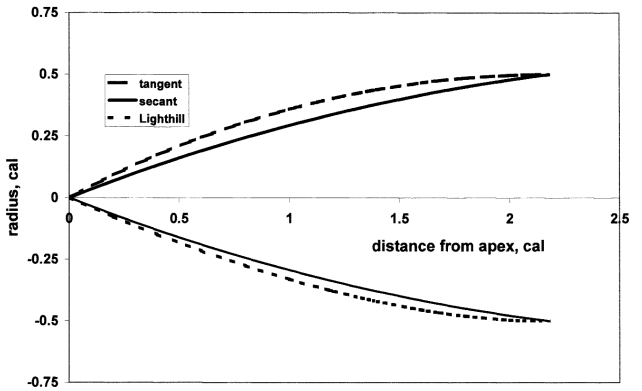



Figure 9. Ogival head shapes for projectiles.
(outer: 5crh tangent; intermediate: minimum wave drag form[60], 1945; inner: 5/10crh secant).

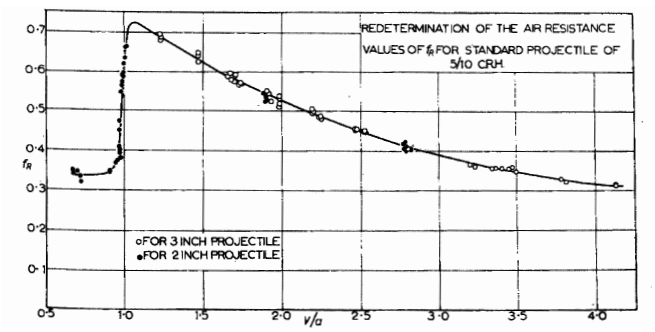



Figure 10. Firing tunnel data for the resistance coefficient of the New Standard Projectile, 1940, (5/10crh nose)[61] [note $f_R = \pi C_D/2$].

The pioneering work of von Kármán and Moore, though giving results similar to those of the exact calculations of Taylor and Maccoll, had assumed that the flow was only slightly perturbed and its characteristics were isentropic and irrotational. The subject was hugely widened later by (Michael) James Lighthill, who went on to show that the wave drag of any slender body of revolution could be determined analytically[60]. He also showed that there was a minimum wave drag for such a body. For application to projectiles, that had cylindrical bodies, he identified a practical head shape that would give a wave drag close to the minimum possible value. The form he derived falls between those of tangent and secant ogives, and is also illustrated in Fig. 9. His theoretical value for the minimum possible wave drag coefficient is $1/L^2$. This was in agreement with the evidence from ballistic measurements such as those given by Hunt, showing how the drag fell with increasing values of head length[53]. For the proportions of a 5crh nose the minimum wave drag coefficient would be 0·211. The nearest analytic form to the minimum drag shape that Lighthill could find, as shown in Fig. 9, has a wave drag coefficient of 0·237

The overall peak $C_D$ for the Standard Projectile, including skin friction and base drag, is 0·457, and comes at M = 1·07. The Ballistic Tables show the curve of resistance coefficient against Mach number up to M = 4·5. In a paper contributed to an AGARD study after the war, Maccoll shows this, as reproduced in Fig. 10, together with the experimental data points from which it had been constructed[61]. (The consistency of the measured results is remarkable, especially in the narrow region of the very steep increase immediately before M = 1. To researchers in aeronautics, attempting to obtain aerodynamic data in the transonic region, it must have been a revelation to find that methods of such precision had been long in use in a related field).

Thus it was, that by the 1940s, a practical head shape had been defined for projectiles that had been shown to give a very low drag in supersonic flight. Further, this was backed by methods by which the lift, drag and moment coefficients and other properties could be estimated for any slender body in this regime, though only for speeds beyond that at which the nose shock became attached.

### 5.4.2 Wings

The first measurements of the aerodynamic characteristics of aerofoils in supersonic flow were made at the NPL in 1928, also in Stanton's small tunnel, as another part of the investigation of sections for propellers[62]. To avoid interference effects, only tiny aerofoils, of 0·3 and 0·5in chord, could be used. Despite the difficult experimental conditions, balance arrangements were devised that gave consistent results for lift, drag and pitching moment. Taylor subsequently showed that these were in good agreement with the Ackeret theory for the corresponding unsymmetrical biconvex section, if it was assumed that there had been a minor error in the incidence settings[63]. Later, Stanley Hooker showed that this was in fact due to a second-order term arising from the camber, which had been predicted by Busemann[64].

The particular presentation of the theory by Taylor had been for biconvex sections, but the significance of the results was most readily illustrated by reference to the characteristics of a thin symmetrical double-wedge aerofoil. This was the basis of a summary by Relf, circulated to the ARC Aerodynamics Committee in May 1943, when the possibility of supersonic flight was about to be considered seriously[65]. Relf represented the Mach number effects in terms of the group

$$\lambda = 4/(M^2 - 1)^{1/2} \quad \ldots (5)$$

Then, if $\beta$ is the semi-angle of the wedges, and $\alpha$ is the incidence of the section to the free stream, the resultants of the pressure distribution over the four flat surfaces of an infinite wing gave the simple forms

$$C_L = \lambda\alpha \quad \ldots (6)$$

$$C_D = \lambda(\alpha^2 + \beta^2) \quad \ldots (7)$$

$$= (C_L{}^2/\lambda) + \lambda\beta^2 \quad \ldots (8)$$

and $C_M = 0$ about the mid-chord point $\ldots (9)$

Equation (8) showed that the pressure distribution produced a profile drag that was not zero at zero lift, and that the overall drag coefficient had a parabolic form in supersonic flight, with a $C_L^2$ term, superficially similar to that of the induced or vortex drag in subsonic flight. However, both parts of the expression were now functions of Mach number.

From Equations (6) and (7), the lift/drag ratio was given by

$$C_L/C_D = \alpha/(\alpha^2 + \beta^2) \quad \ldots (10)$$

which is independent of M. It has a maximum value of $1/2\beta$ when $\alpha = \beta$. Since, for a thin symmetrical double-wedge aerofoil, $\beta$ is equal to the thickness ratio, this clearly showed how, for an acceptable performance in the supersonic regime the wing section must be very thin. To obtain a maximum $L/D$ of 10, for example, it would have to become practicable to devise a wing with a thickness ratio of 5%.

It was seen in Equation (6) that the lift coefficient of a surface in linear theory is proportional to the incidence $\alpha$ and the quantity $\lambda$, given in Equation (5). To obtain the magnitude of the lift, the coefficient will be multiplied by a group containing $V^2$, and so in the stratosphere where the speed of sound is constant, to $M^2$ also. It

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followed that all forces and moments for a given situation are then proportional to the group:

$$X = M^2/(M^2 - 1)^{1/2} \quad \ldots \quad (11)$$

This varies very little over the range of Mach number of interest here. Thus the lift and moment of a wing at a given incidence in supersonic flow would be almost invariant with speed. And so, if the wing loading could be chosen to give an incidence at which the lift/drag ratio was at its maximum, there would be a practical speed range over which the wing drag would be near the lowest achievable.

In general, Taylor showed that, for a part of a surface inclined at a small angle $\theta$ to the free-stream direction, the local pressure coefficient is proportional to $\theta$. It follows that, for a general aerofoil section, the expression for lift, Equation (6), is universal, but the term $\beta^2$ in Equations (7) and (8) should be replaced by the mean square of the inclination of the surfaces, averaged over the chord. For a symmetrical biconvex section, this is $4(t/c)^2/3$. The drag is increased by a third relative to that of the double-wedge aerofoil, so that for a section with a thickness ratio of 5% the maximum $L/D$ ratio is reduced to 8·68. Getting a satisfactory performance from a wing in supersonic flow was not to be easy.

Further validation of the theory was obtained in 1939 by Ferri at the Guidonia research centre, using the supersonic wind tunnel that had been built to a design supervised by Ackeret[66]. This was a closed-circuit tunnel, in advance of anything in Britain at the time, with a working section of 400mm x 750mm (15·5in x 29in) cross-section[67]. Tests had been made on seven different aerofoils, of double-wedge and biconvex sections, symmetrical and unsymmetrical, with chords up to 60mm (2·4in) and of various thickness ratios, at Mach numbers of 1·85 and 2·13. The texts of the reports on this work were obtained, and translations were available within the MAP[68]. For practical values of $C_L$, agreement with the linear theory was good in all cases, departures eventually emerging at higher values, due to complex interactions appearing at the trailing edge of the section.

Thus, at the time of the M.52 project, there was a well-validated theory for the basic characteristics of thin wings at supersonic speeds. However, there was a lower Mach number limit to its validity, set by the requirement that the shock wave at the leading edge should be attached. The theory for supersonic flow over a wedge had been known for some time, and showed that this first occurs at a much higher Mach number than on a cone of the same included angle, evidence of the 3D relief in this regime of flow. Although, for a biconvex section of say 6% thickness ratio, the included angle was only about 7º, the leading edge shock on a wedge of this angle would not be attached below a Mach number of about 1·2. Thus, from high subsonic speeds up to a point such as this, there remained a region for which theoretical predictions could not be made, and measurements were very few – essentially only those from early propeller tests and the open-jet measurements of Ferri. Assessment in this gap would therefore have to be filled on the basis of information that was very limited, though not totally absent, as has been supposed.

# 6.0 THE DESIGN DECISIONS

## 6.1 The layout

Concerning the M.52, then, it was being said that 'this aircraft is being built to obtain the information necessary for it to be designed'. With no previous example to go on, Miles could range widely in the search for a suitable starting-point. The memorandum on the wartime activity at Miles records that several possible layouts were considered in framing the Outline design[25]. As well as the biplane and the multiplane arrangements suggested by Whittle, these included tailless layouts and the canard configuration, with a large foreplane, of which Miles had unique design and flying experience, with its Libellula (dragonfly) aircraft[14]. The preferred layout, around which the design had quickly crystallised, was however more orthodox, as was shown in Fig. 4, and the design decisions that underpinned that can now be reviewed in relation to the state of knowledge available at the time.

## 6.2 The fuselage

Miles looked to projectile experts for advice on the shape for the M.52 fuselage. This emerged finally with the form of a central cylindrical body, about 3·5 calibre in length, with a shroud surrounding the annular air intake at the front and a boat-tail afterbody terminating with the exhaust nozzles. The ogival nose of the pilot's capsule projected about 1·35 calibre beyond the intake. For the first prototype, with the Whittle engine alone, the overall slenderness ratio was 7·2, and was 6·8 for the second, with the bypass arrangement and thrust augmenter, the afterbody being cut off earlier in that case due to the larger final diameter at the outer exhaust exit. These proportions were not hugely different from those of contemporary practice for subsonic flight, though the outer diameter of the fuselage was large, being determined mainly by that of the power plant that it had to accommodate. The overall diameter of the basic W2/700 engine was 3·5ft, but with the surrounding duct for the bypass air, and provision for the fuel to be carried in the fuselage near the cg, so as to minimise trim changes, the outer diameter was fixed at 5ft. With the frontal area already larger than desired, it had been decided to locate the pilot within the nose of the basic envelope, so that there was no projecting canopy to increase the area further, or to break the lines of the ballistic form.

To finalise the fuselage shape, and to obtain an estimate of its contribution to the drag, consultation took place with Maccoll, then at the Armament Research Department (ARD). He informed a visiting group from Power Jets that 'for good projectiles, the total resistance could be divided approximately as to '40% head resistance, 50% base suction and 10% skin friction', somewhat in line with the theoretical estimates that had been made by von Kármán and Moore[69]. The presence of the exhaust streams would be expected to cause the base drag component to be lower than for projectiles, but this did not diminish the importance of keeping down the head drag by a suitable choice of shape for the nose. Power Jets had already made some drag measurements of a bullet in their own small supersonic wind tunnel, obtaining a coefficient of 0·7. Maccoll said that for a shorter bullet with a similar head shape, their value had been 0·596, suggesting that these had been for blunt-nosed rounds. His advice was to use the well-understood 5/10crh ogival form, which for the Standard Projectile gave the lower peak drag coefficient of 0·457. it is not clear if he produced the new Standard Drag Function from which this was obtained.

Miles pointed out that, as is seen in Fig. 9, a secant ogival profile for the nose is not tangential to the cylindrical body where it intersects it. They were assured that this would have little effect on the drag[25], but on the M.52 the nose shape was not in fact continued back to the point of intersection, and there is no discontinuity there. The rear bulkhead forming the closure of the back of the nose capsule was placed a short distance inside the intake shroud, and the sharp forward edge of the shroud at the front of the body was curved over it to form the lip of the annular air intake. At its rear, the shroud blends smoothly with the cylindrical section of the fuselage. The resulting shape of the surfaces at the nose of the M.52, which contributed much to the aircraft's purposeful appearance, is shown approximately in Fig. 11. The nose form is that of a 5/10 secant ogive, and the intake shroud is shown following the profile of the 5crh tangent ogive. This gives a pleasing shape to the shroud, one likely to maintain a low supersonic drag, though it is not known whether it was formed exactly to that curve.

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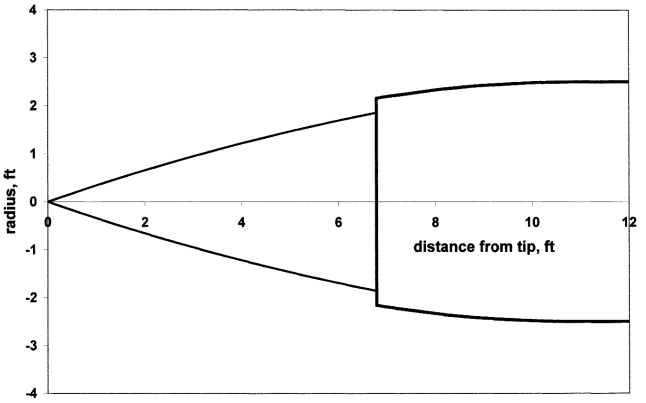



Figure 11. Outline form of the nose capsule and intake shroud for the Miles M.52, showing ogival shapes.

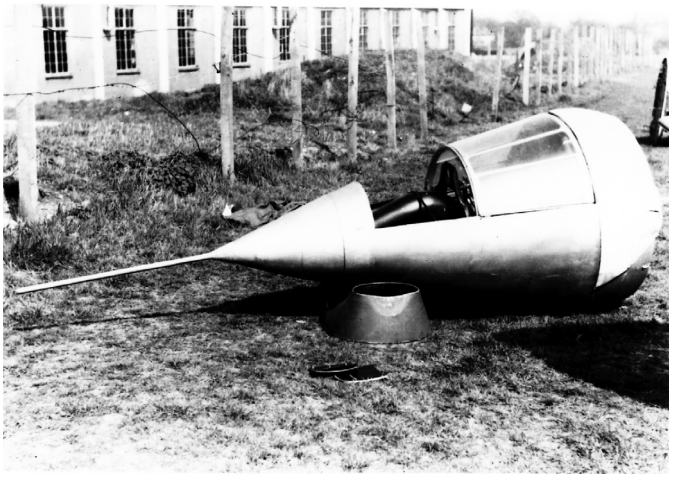


Figure 12. Mock-up of the pilot's capsule for the Miles M.52, awaiting disposal (The Miles Aircraft Collection).

With this arrangement, the flow passing over the nose would still be supersonic at the intake, where in most circumstances it would experience a final normal shock at the entrance. In the expanding internal space that followed, there would be a further, subsonic part of the pressure recovery ahead of the entry to the compressor of the engine, and to the bypass duct, when the augmenter was fitted.

Engines of the Whittle type were not designed to take air directly from an intake duct, but from a plenum chamber surrounding the casing of the rotating parts. In the M.52, this chamber terminated at a bulkhead in the vicinity of the wing main spar fitting. When the augmenter was used, the second flow of air was ducted directly from the intake to the axial compressor inlet via an annular duct close to the fuselage skin. There is a reference also to a bleed, 0·5in deep, being in place, by which the slower-moving air from the boundary layer on the nose was to be removed, and 'taken through the fuselage and discharged to atmosphere through an annulus around the exhaust exit'[70]. The exact arrangement for this cannot be distinguished in the available drawings, and its origin remains obscure, though at this time a boundary-layer bleed would have been an innovation.

Miles took the view that the design of the intake and the exhaust nozzles were part of the responsibility of the engine supplier, but inevitably that reacted onto the fuselage design in general. Collaborative work on these items continued throughout the life of the project and was continued for a further time at Power Jets. The highest efficiency for the annular intake in supersonic wind tunnel tests had been given as 73% with boundary-layer bleed[7]. However, values in the range 80-90% were being reported later[71]. It is probable that, for efficient operation over the full speed range of the specification, a variable area intake would be required. Though this was discussed at various times, it seems that a design had not been finalised by the end of the programme, and the surviving drawings show no details of how it might have been arranged.

Photographs of the nose of the mock-up show that there was to be a long pitot tube projecting forward from the nose of the pilot's capsule (Fig. 12). As mentioned in a report by US officers who visited Miles in 1946, this was 'an approximately two foot long, thick walled tube acting as a pitot head and shock wave splitter'[72]. If the latter term had been heard at Woodley, it might imply that Miles had been aware already that the overall isentropic efficiency of compression would be higher across two consecutive weaker shocks than across one shock with the same overall pressure rise. A shock on the front of the pitot tube would contribute to this, providing an initial pressure rise ahead of the main shock pattern beginning at the nose of the capsule. It will be recalled from Fig. 4 that the Outline design sketch of early 1944 showed a more substantial spike projecting ahead of the conical nose. If that had been intended to have a similar purpose, it would have shown considerable foresight, though there is nothing on record to confirm this.

While the position of the pressure cabin in the nose should have given a good forward view, that was limited by the frames of the canopy, and distortion would be a risk when the transparencies lay at such a shallow angle to the pilot's line of sight. The space available within the cabin was small and of awkward shape, requiring the pilot to be of slight build and to sit practically on the floor in a semi-reclining position, with feet raised above the level of the seat pad. To make the pilot more comfortable in his unusual posture, flexible supports were provided for the thighs and lower legs, arranged to collapse out of the way temporarily to allow entry and exit from the cabin[70]. Pressurisation of the capsule at altitude was provided by a system similar to that devised by Westland Aircraft for its high-altitude fighter Welkin. With a maximum differential of 2·5lb/in$^2$, a cabin pressure equivalent to 35,000ft could be maintained to an altitude of 65,000ft. As the cabin air was taken from the compressor of the engine, arrangements were made to cool it to 20ºC by first passing it through a heat exchanger, with heat transfer to evaporating ammonia[70].

It had been recognised that in an emergency at high speed, the pilot would not be able to leave the cabin in the normal manner, so the capsule was made to be jettisonable. Explosive charges would sever the supporting structure, disengaging all connections and releasing the capsule. The stagnation pressure of the air in the plenum chamber behind it would then give an initial forward impulse to the rear bulkhead of the capsule, to assist its separation from the fuselage. Without the weight of the capsule at the front, the rest of the aircraft would rear up, and drop back quickly. After falling clear, the capsule was to be slowed in a stable configuration by a small drogue until a speed and altitude were reached at which the pilot could push away the canopy and leave safely, to make a final descent by his own parachute. Stability of the capsule, when in free flight, required it to be ballasted so that its cg was ahead of the aerodynamic centre, and wind-tunnel measurements were made by Miles, to establish the requirements for that[73]. Ballast was assisted by placing well forward in the capsule the 'automatic observer', a multi-channel instrumentation package to be provided by RAE, by which essential aerodynamic and engine data were to be continuously recorded in flight, together with some strain gauge readings, probably including stresses in the main wing spars. Protection provided in the hope of recovery in the event of an emergency separation of the capsule prefigured the 'black box' flight data recorder that was to become ubiquitous in later years.

For the case of emergency escape without separation of the capsule, such as might be necessary at low altitude, studies were made in the RAE Blower Tunnel, with a model of the M.52 and a dummy having the appropriate scaled inertia to represent the pilot[74]. These tests showed that the canopy would clear the aircraft after release in flight,

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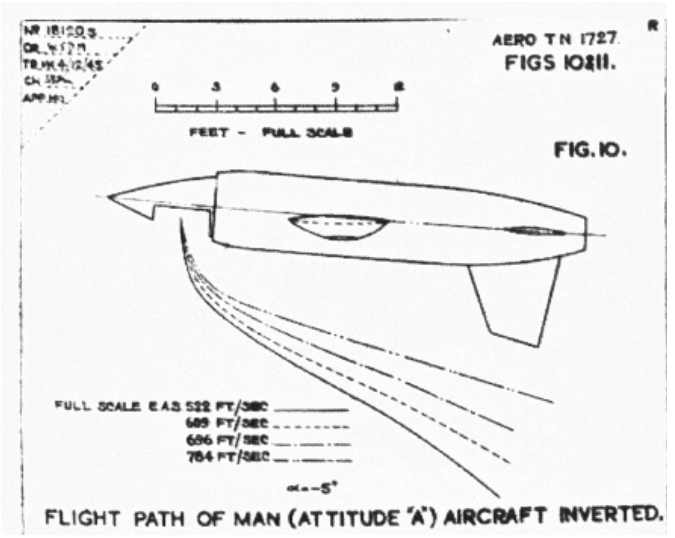



Figure 13. Model tests in the RAE Blower Tunnel, showing trajectories of the pilot, after leaving the capsule *in situ*, aircraft in preferred inverted orientation[74](FAST Museum).

but that at speeds below 500ft/s the airflow would not carry the pilot high enough to avoid risk of injury, from striking the sharp edges of the tail. If, however, the pilot could first roll the aircraft into an inverted position, with the direction of gravity now giving assistance, escape would be 'relatively easy and safe'(see Fig. 13).

Extensive calculations were made of the wheel loading expected during the ground runs at take-off and landing[75]. A robust tricycle undercarriage was designed by Messier, with special wheels and tyres from Dunlop, suitable for operation at ground speeds up to 200mph[70]. As there would be insufficient depth to accommodate the main wheels in the wings, these had to be retracted into the fuselage, resulting in the narrow track of 4·5ft, which would not have made for easy handling on the ground. A less than desirable arrangement had to be made for the nosewheel also, which was mounted in the nose capsule and retracted forward, into a well located between the pilot's legs.

It was known that for projectiles, the high proportion of the overall drag contributed by the base drag was due to the large turbulent wake formed after separation at the sharp cut-off at the rear, and that this could be reduced by a suitable choice of boat-tail to terminate the base of the body. The shape chosen for the M.52 afterbody had the profile of a tangent ogive, with generating radius of 7·5 calibre, which was truncated at different lengths, according to the diameter of the exhaust nozzles required for the different engine installations. At this large generating radius, and being also foreshortened, the afterbody shapes were not greatly different from blended shallow cones, with a half-angle of about 8–10º. Possible effects on the base drag of the fuselage are considered in the evaluation of performance, given later in Section 9.

### 6.3 The wings

With the early indications of the loading and the maximum lift coefficient that could be obtained in the landing condition, it was decided that the smallest practicable wing that would give an acceptable stalling speed would have a gross plan area of 142ft[2,76]. A span of 27ft was chosen to give outboard ailerons sufficient authority, and then the aspect ratio was 5·1, not greatly different from established practice for subsonic flight. The wing was mounted slightly above the centre-line of the fuselage, giving a nett area of 109ft$^2$. The large difference between the gross and nett area would raise questions on the continuity of lift across the fuselage, and interaction between the flows over the wing and fuselage at the junction.

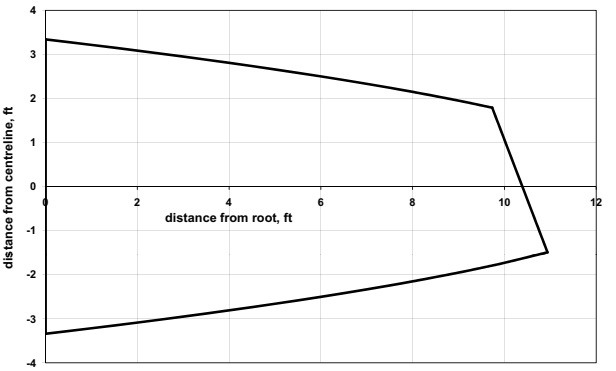



Figure 14. Plan-form of the wing for the Miles M.52.

But the proportions were similar to those of the Gloster E28, which also had a wide fuselage to accommodate the first Whittle engines to be flown, and its flight characteristics had been exemplary, up to speeds of M = 0·75, which had not been reached before in level flight[5]. With the turbojet engine and thrust augmenter fitted, the wing loading of the M.52 at maximum take-off weight would eventually be 58lb/ft$^2$, more than twice that of the E28. The maximum usable lift coefficient for the sharp-edged section was found to be lower than expected, at 0·7, giving a flaps-up stalling speed of around 180mph EAS. With flap, the minimum take-off speed would be somewhat lower, though to reach it, a distance of 4,560ft would be required to clear a 50ft obstacle[14]. These conditions would have represented a challenge to the pilot, but were considered to be within a tolerable margin of current experience.

A straight wing planform was chosen for the aircraft, with a taper ratio of about 2:1, which, with an aspect ratio of 5·1 would be expected to give a reasonably efficient spanwise lift distribution in the subsonic range. The section would be the symmetrical biconvex one, with the maximum thickness at mid-chord. This shape, with a somewhat larger included angle at the apex than for a double wedge section, would be likely to have better low-speed behaviour, though at the expense of what was thought to be a 50% higher drag coefficient in supersonic flight[76] (actually one-third higher at zero lift, as shown earlier). By enclosing a larger cross-sectional area, this shape would also be better able to support bending and torsional loading. Even so, structural limitations dictated that the minimum thickness of the wing at the root could not be less than 6in, corresponding to a thickness ratio of 7·5% there. This would require the convex curve of the wing skins to have a radius of about 22·4ft at this point. Though it was at first supposed that the thickness ratio would be uniform along the span, it was found that the thickness could be allowed to taper linearly to some lower value at the tip, while remaining within the wing strength requirements.

The final plan shape of the wing has been described as somewhat 'elliptical'[77], though it was not a design choice in itself, but a result of the values chosen for the taper ratio and the linear taper in thickness. This is a consequence of the idea, patented by Miles, of building the wing with rolled skins of an equal and constant radius of curvature throughout. No double-curvature, involving expensive pressing of heavy-gauge material, would be required, and the method of assembly would be simple and accurate. The plan shape is defined where the edges of the two bounding cylinders come together. Different sources give the tip dimensions variously, but Bancroft records that it was the tip chord that was chosen first, to give a reasonable taper ratio in plan, and then the linear taper in thickness gave a tip ratio close to 4%. With this value, the resulting plan shape becomes as shown in Fig. 14. The effective mean thickness ratio has also been variously given, but would in any case be slightly different for lift and moment than for drag. The effective value for drag is probably of most significance. In the supersonic

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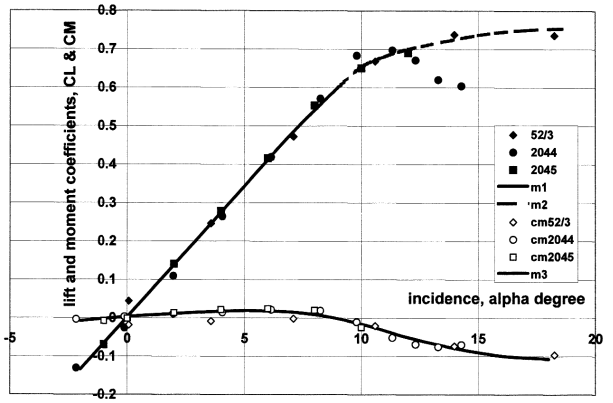



Figure 15. Low-speed lift and moment characteristics of the Miles M.52 wing[80-82].

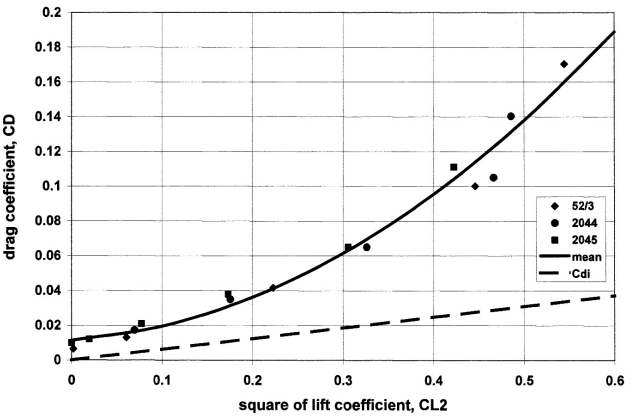



Figure 16. Low-speed drag characteristics of the Miles M.52 wing[80-82]; dotted line – expected vortex drag contribution.

region, the drag coefficient would be dominated by the wave drag component, as in Equation (7), and thus would be roughly proportional to the square of the thickness ratio. Then, at any given spanwise position, the contribution to the drag by an element of the wing would also be proportional to the chord there. As so, the effective mean thickness ratio for the whole wing works out at about 6·2%, somewhat higher than the average of the root and tip values, since it is weighted by the square of the thickness and the chord, which are greater towards the root. This will be used as the representative value when considering other aspects of the design.

The wing was given no washout, and the final value of the dihedral was 2º. It is reported that the wing was normally rigged at 0º 33' relative to the fuselage centreline[78], though the rigging angle could be adjusted on the ground over a range of 2º if required from flight experience.

A notable feature of the final form of the M.52 wing was the cut-off at the tips, at an angle to the direction of the flow. This was initially a reflection of the conjecture by Busemann that, if any vortex motion was induced at the tip in supersonic flow, this would be confined to the region within the Mach cone behind it. If the material of the wing lying within the cone were then removed, the wing would experience no effect from any change in the flow that might originate at the tip. It would then operate in essentially two-dimensional flow throughout. However, the angle of cut-off actually employed on the M52 wing was around 22º to the flow direction, rather than the smallest angle it could take by Busemann's argument, which would be about 42º at the maximum Mach number of 1·5, by Equation (2). Further, it was decided to cut off the tip in the outward-going direction, the opposite of that proposed by Busemann, and a reversal of that shown on the early Outline design. This change was made on the grounds that the maximum length of trailing edge was needed, to provide sufficiently large areas for the flaps and ailerons. Some unspecified wind-tunnel tests were made, to check the effect, finding that 'the cut-off with the longer trailing edge had only slightly increased the drag over the other method'[76].

The origin of the actual cut-off angle, which is a prominent feature of the M.52, as it was employed on the fin also, remains obscure. Bancroft stated that it was made at the Mach angle from the tip[76], and this has been repeated by numerous subsequent commentators, while others state that it was intended to keep the wing within the shock wave from the nose[1,79]. But although this wave would come close to the wing-tips at the maximum M = 1·5, with the shallower angle actually used, the tip would not be parallel to it.

It would be essential at an early stage to have an understanding of the low-speed characteristics of the chosen wing design. Measurements were made by Miles in their own wind tunnel, and these were supplemented by two sets made by RAE[80,81,82]. It was expected, from wool-tuft observations, that flow separation would

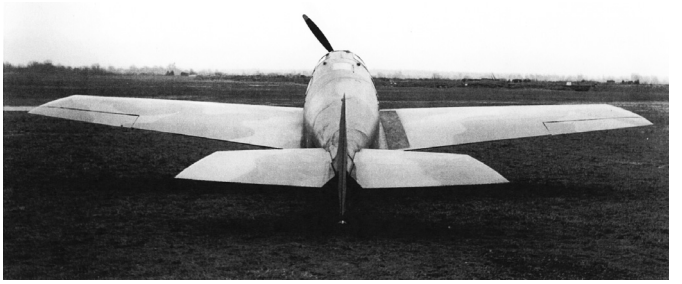


Figure 17. Miles Falcon Six (the 'Gillette Falcon'), fitted with thin, sharp-edged wings and all-moving tailplane (The Miles Aircraft Collection).

begin at quite small values of incidence, and the effects were clear. The mean lift curve for the available data, shown in Fig. 15, is linear only up to about 9º incidence, with a maximum usable lift coefficient of 0·7. The slope is 3·9 per rad, somewhat lower than would normally be expected for an aspect ratio of 5·1. Also shown in Fig 15 are the quarter-chord moment coefficients, which at first are practically zero, as would be expected for a symmetrical section, but become progressively more negative as the stall develops, indicating an accompanying rearward movement of the aerodynamic centre. Drag coefficient measurements, gathered together in Fig. 16, show a gradient that increases with increasing $C_L^2$, the values becoming hugely greater than could be attributed to induced or vortex drag, the conventional line for which, with constant gradient, is shown dotted in the Figure. These basic characteristics are similar to those of a flat plate[83], mainly reflecting the effects of the separation at the leading edge, spreading to cover the whole section at high incidence.

The low-speed behaviour of the thin wings and tailplane were to be confirmed in tests at full scale and at a Reynolds number similar to that of high-speed flight at altitude. Suitable replica surfaces were mounted on the Falcon aircraft L9705, returned to the Company from RAE, to which it had been supplied for research involving interchangeable wing sets (see Fig. 17). Fitted with its thin, sharp-edged wings, it was generally referred to as the 'Gillette Falcon', after the familiar manufacturer of razor blades (a 1946 report in *The Aeroplane* gives the nickname as the 'Breadknife' but this seems to have been mistaken[78]). A modified main undercarriage was fitted, with attachments to the fuselage instead of the wings, so as to leave the wing air flow as free as possible of any obstruction.

Test flights of L9705 in this configuration were begun in August 1944, flown by the Miles test pilot, (Flt Lt) Hugh Kennedy. It was found at first that if the nose were raised by more than a small amount, the increase in drag would be too great to allow take-off

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conditions to be reached. The aircraft was therefore allowed to accelerate without rotation until it could be flown off. With practice, procedures were worked out by which the Falcon could be flown routinely, and the flight characteristics of the new wing were duly established, over a range that would be appropriate for the M.52. These were substantially as shown by the wind tunnel tests[84]. The maximum lift coefficient for the clean wing was confirmed to be about 0·7, and the stall developed gradually, extending over a considerable range of incidence. Measurements of the profile drag were made in flight with a pitot rake, but although the results obtained were internally consistent, the values were lower than those found in the wind tunnel tests. However, drag measurements obtained from partial climb testing were in closer agreement, perhaps indicating that the pitot rake had not fully intercepted the wake.

The low-speed behaviour was considered to be 'exceptionally good', with the controls remaining effective throughout, including into the stall[76]. With split flaps at the full 65º deflection, the maximum lift coefficient found in wind-tunnel tests had been 1·29, a satisfactory value in view of its influence in the selection of wing area, in which reference to the landing condition is a key element. However, during the design of the wings for the M.52 itself, it was considered that there would be significant advantages for the strength and weight if a plain flap were used instead. Finally, it was concluded that the highest lift coefficients that could reasonably be employed would be about 0·7 on take-off and 1·1 on landing. Because of the implications for the speeds in these conditions, RAE made some investigations of possible alternative high-lift arrangements, as reported below.

Having heard about these matters, the Supersonics Committee suggested fitting the wings with a removable sleeve or 'plaster' for early flight trials, to give the wing a temporary rounded leading edge and more conventional characteristics. Miles made some wind tunnel tests with such a sleeve, and saw the further possibility of turning it into a jettisonable slipper tank for carrying extra fuel[85]. However, as in due course flight had been managed safely with the sharp wing and empennage on the Falcon, by their own test pilot and others at Farnborough, the idea of temporary modification of the section was not pursued.

The RAE wind-tunnel work on possible improvement to the low-speed behaviour of sharp-edged wings examined modifications of the basic shape, of a kind that might produce higher lift, while suppressing some of the separation that was responsible for the high profile drag of the sharp-edged section[86]. The front 16% of an 8ft span test wing was arranged to move forward and rotate, becoming a leading-edge slat, and the rear 25% to move aft and rotate, to form a slotted semi-Fowler flap, as shown in the central part of Fig. 18. With the best combination of movements of the slat and the flap, a maximum $C_L$ of 2·02 was obtained, with a stall incidence of 17º, much as expected for a cambered wing of conventional section with similar geometry. The early part of the profile drag slope $dC_D/dC_L^2$ was lowered to about 40% of that of the basic section, and extending to much higher $C_L$ with a constant value, indicated that the flow separation had been reduced considerably. The practicality of fitting these devices to a thin wing section was not pursued. However, it was suggested that quite similar results could be obtained more simply, by articulating the wing about spanwise hinges at around 15% and 70% of the chord, as shown in the lower part of Fig. 18. Test results showed that $C_{Lmax}$ of 1·84 could be obtained, with an overall $C_D$ never more than twice the ideal $C_{Di}$. It was asserted that 'an aircraft embodying this flap arrangement would be free from the disadvantages associated with the early stall of sharp-nose sections'. The potential for an improvement in low-speed performance, and the possible raising of the flight ceiling by having what, with small settings, would effectively be a variable-camber wing, was striking. It can hardly be doubted that this would have appealed to Miles and been followed up, if the programme had continued.

Tests up to high subsonic speeds were made by RAE on wing + body and whole aircraft models in 1944 and 1945[87,88]. The primary

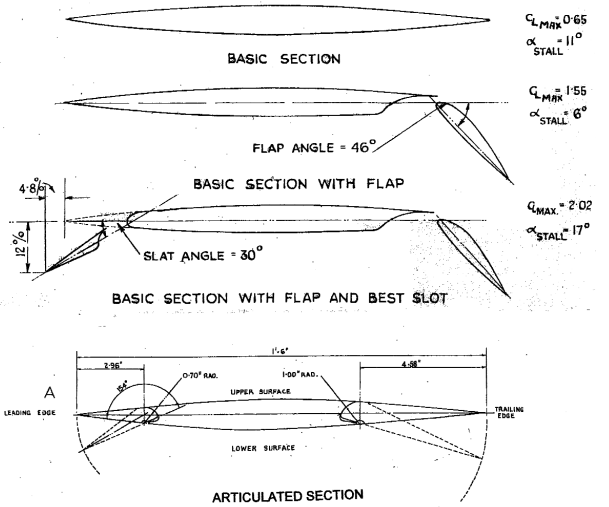



Figure 18. Possible high-lift arrangements for wings with sharp-edged biconvex section[82].

purpose of such tests is to establish values for stability derivatives, and they will be considered in that context in Section 6.6. Here it is noted that the tests further consolidated the conclusions concerning the basic lift, drag and moment characteristics of the wing, as reviewed above. With the aid of wool tuft tests, they also confirmed the observations that had been made earlier in the Miles tunnel, that the stall developed over the wing progressively. Beginning at the root at incidence of 6º, it had spread over the inner half of the wing by 9º, but the tip did not stall until 12º incidence was reached[89].

In the tests in the RAE High Speed Tunnel, results were obtained up to M = 0·82, made possible by streamlining the model strut supports (sting mounting was not yet in general use). An increase in lift curve slope with increasing M was noted, being in fair agreement with the predictions of the Prandtl-Glauert rule of Equation (1). Although it was considered that the tunnel corrections were unreliable for drag coefficient values beyond M = 0·7, there was encouragement in the finding that the thin wing did not exhibit the 'shock stall' at any M up to 0·82[88].

In December 1943, just after the Miles project had started, further confirmation of the applicability of the Ackeret theory was provided by the Aerodynamics Division of NPL[90]. Measurements of lift, drag and moment coefficients were made on a wing with an 8·7% thickness ratio, with a section that had originally had the form of a double wedge, but now had the discontinuities at the mid-points removed by blending the front and rear wedges together with circular arc profiles. This formed a more practical section, closely resembling the biconvex one. Tests were made from M = 0·4 to 0·82, and at two supersonic values, M = 1·21 and 1·45. Though for a thicker section than would be used on the M.52, these provided additional supporting material on the characteristics to be expected for sharp-edged wings in the high subsonic region, as well as showing satisfactory agreement with the linear theory in the supersonic one. From the drag measurements, it was further indicated that in supersonic flow the skin friction contribution was of the same order as would be given by the contemporary procedures based on boundary-layer theory for subsonic flow.

During the currency of the M.52 project, continued pioneering work by Lighthill and others was to clarify greatly the uncertainties that had been reported earlier by Relf about the nature of the flow over a supersonic wing and the generation of lift and drag by it. If information could not be passed forward on a wing, it did not appear that the flow could be represented by the circulation model that had been so successful in describing aerodynamics in subsonic flow. Without circulation, there would be no downwash behind the wing, and no tip vortices. But as there was lift, it would have to be reacted

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on the surroundings somewhere. Lighthill showed in a study in 1944 that in fully-supersonic flow this reaction took place on the air in the regions above and below the wing, as it passed between the fore and aft shock waves[91]. The extent of the shock waves in supersonic flow meant that the region of air affected was very great, so that the overall disturbance at any point was slight. The flow leaving the trailing edge was returned essentially to the same speed and direction as those of the incident free stream. And so, when considering the flow at the tailplane, as required for stability calculations, there would be 'no appreciable downwash' to be represented behind the wing.

In another study in November 1944, Lighthill gave a procedure for determining more precisely the distribution of lift and drag over a finite wing in supersonic flight[92]. Among much else of a more fundamental nature, this showed that a rotational flow was induced in the air behind the tips, as in classical circulation, but that the disturbance was relatively weak. This meant that the decision to cut off the tips of the M.52 wing in the outward-going direction would not be troublesome. Lighthill concluded specifically that 'though it would be economical in the strictest sense to follow Busemann's plan and cut off the ends of the span at Mach angle, almost any consideration which may deter designers from doing this (such as the presence of ailerons etc) may be held to be of greater weight'.

### 6.5 The empennage

A prominent feature of the early Outline design was the large fin, of total area 24ft$^2$, made necessary by its short moment arm on the tubby fuselage. Significant changes were made to the empennage as the design evolved. For the Final version, the total fin area was reduced to 18·5ft$^2$, and the direction of the cut-off at the tip was changed, as on the wings, so that the trailing edge had the greater length, to give more effectiveness to the rudder. For the tailplane, which was all-moving, and so without elevators, this consideration was not relevant. Its most notable feature is perhaps that, originally straight, it was now swept back, at about 20º on the mean chord line (24º on the quarter-chord line). This ensured that, in subsonic flight the aerodynamic centre of the tailplanes was near to the axis about which they rotated, so as to minimise the hinge moment. In supersonic flight, the centre would move aft of the axis and although the hinge moment would become large, it would act in a restoring direction. At the root, it was necessary to locate the axis at the mid-chord, to provide enough depth for the fittings. The tip was kept perpendicular to the spar line, so that due to the sweepback, the leading edge was longer than the trailing edge, the reverse of the form of the other surfaces. There was no aerodynamic significance to this difference.

The sections of the components of the empennage were of similar biconvex form to that of the wings. The mean thickness ratio of the fin and rudder was about the same as for the wings, while the tailplane was somewhat thinner, with 6% thickness ratio at the root and 4·25% at the tip. The total plan areas, as fractions of the gross area of the wings, were 0·25 (nett) for the tailplane and 0·13 for the fin and rudder. The corresponding values for the Typhoon, a front-line fighter entering service at the time, were respectively 0·14 (nett) and 0·09, so these proportions for the M.52 were significantly larger than would be typical of contemporary practice for subsonic aircraft. No doubt a large area for the tailplane was desirable, to enhance its authority in the transonic region, but the value intended by Miles was initially somewhat smaller, and after wind tunnel tests it had been raised at the suggestion of RAE by increasing the span by 10%. It was perhaps likely that there would also be uncertainty about the effectiveness to be assumed for the tailplane when the fuselage occupies so large a fraction of its overall span (finally 28%).

At the time of its specification for the M.52, there had been no flight experience with the use of an all-moving tailplane to provide the only means of control in pitch. As it was also to have the thin, sharp-edged biconvex section, wind tunnel tests were first made in the Miles tunnel to establish the basic characteristics. These were made on a specimen of rectangular planform and thickness ratio of 7½%, with a full-span elevator of 36·5% chord[93]. With the indication that longitudinal control would be satisfactory with this arrangement, trials with a thin, sharp-edged tailplane were undertaken at full scale with the Gillette Falcon. After the initial experience with conventional elevator control, a fully all-moving tailplane was fitted[94,95]. It is reported that the low-speed characteristics of the aircraft with the all-flying tailplane were not found to be significantly different from those with elevators.

### 6.6 Balance, control and stability

The origin of the all-moving tailplane has been much discussed. Its inclusion in the E.24/43 specification showed that it had been put forward from an informed source, probably at RAE. At the time, the idea was being considered for dealing with the trim and stability changes experienced by aircraft in a dive. But flight trials with the arrangement giving a sufficient range of movement to provide both trim and control, as on the Gillette Falcon, seem not to have begun until early in 1945[95,96]. With the all-moving tailplane directly connected to the pilot's control column as intended for the M.52, many hours of flying were made with the Falcon, to measure the aerodynamic characteristics with the unique combination of the wing and tailplane. Panoramic cameras on the ground were used to record the behaviour during take-off and landing. A film was made, showing the techniques recommended to be followed in these phases of flight. Essentially, the requirement was to keep the wing operating at small values of incidence, so as to prevent the development of the large profile drag due to separation.

The aircraft was then transferred to RAE, where it was flown by several pilots from Aero Flight, including Eric Brown, who reported similarly to the Miles pilot, Hugh Kennedy, that it was 'not a particularly pleasant aircraft to fly'. This was attributed in part to the tailplane not being mass-balanced, so that any vertical acceleration, for example due to gusts, produced reactions at the control column. But, noting that many of the characteristics were specific to how the new wing and empennage forms had been implemented on the Gillette Falcon, Brown disagreed with the comment of the CO of the Flight, that by extension the first pilot to fly the M.52 might easily be landed in 'considerable trouble'. Brown, who believed that he himself would be that first pilot, considered instead that the disagreeable effects would be negated when irreversible power controls were fitted. He concluded that 'there was no reason to think the bi-convex wings would produce any great handling problems in the low-speed part of the flight envelope'[79].

At the time when the M.52 was being designed, the requirements for stability of a new project were addressed mainly by reference to the tail volume coefficients, and to consequential measures such as the longitudinal static and manoeuvre margins. The adequacy of these quantities could not be assessed in absolute terms, but only in relation to those of existing types where the behaviour after a disturbance was considered to be satisfactory in flight. Methods involving the moment derivatives for the three principal axes were coming into use, but although dynamic factors related to handling were known to researchers, they were not yet developed sufficiently to be taken up generally. It was usual to expect modifications to be required to a prototype, after experience with the earliest flight trials, though reliance was being placed increasingly on measurement of factors related to stability in wind tunnel testing.

It was reasonable that Miles should have chosen design variables for the M.52 that would be expected to lead to high stability in yaw and roll, compared with current practice for small aircraft. But from their first wind tunnel tests with a 1/6·154 scale model, RAE considered that, when taken together, these would have disagreeable consequences[87]. It was suggested that changes should be made that would reduce the derivatives $n_v$ and $-l_v$ to between half and a third of their original values of around 0·1. Miles accepted the advice that

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practical measures to achieve this would be to lower the dihedral from 5º to 2º, and reduce the fin area. This suggestion was undoubtedly a consequence of the extensive parametric studies of the response of fast-moving aircraft to disturbances by gusts and control movement, being made from 1943 onwards with aid of the Differential Analyser at Manchester University[97]. An early use of (analogue) computing for the analysis of coupled motions, this work provided practical guidance on the ranges of these two principal parameters, which, when taken together, would lead to responses which were comfortable and slow enough to be within the pilot's capacity to control.

For wind tunnel tests, the air intake was faired on the model, so that the internal flow was not represented. However, calculations were made at this time of the destabilising effect in pitch of the change of momentum of the air on entering the intake, due to a change of inclination of the fuselage axis relative to the flight path creating a moment arm for it relative to the position of the cg. This would have been a very early, and possibly the first, occasion when this effect was evaluated. The resulting forward movement of the neutral axis was not negligible, at about 0·02*c*. A further result of these trials was that the original setting of the wing chord line, at 2º to the fuselage centre-line, was reduced to 0º 33'.

It was found that the wind tunnel results were noticeably affected by the taper of the conical fairing that had been fitted to the model to round off the exhaust aperture. The initial value of 60º was considered to be too sharp, but with the 30º cone that replaced it, the effects were negligible. The difference was attributed to adverse effects on the flow round the fin. In the final design, the fuselage was lengthened, giving a similar effect due to a more gradual taper of the afterbody, though the position of the empennage remained unchanged.

Interpretation of wind tunnel data, especially estimation of scale effects, was of continuing concern. The M.52 shared with the Gloster E28 a fuselage width that was much larger relative to the span than on current aircraft types. Wind tunnel tests for the E28 had shown that the forward displacement of the neutral point due to the presence of the fuselage had been substantially underestimated by the (empirical) method then in use at the design stage[5]. However, that aircraft proved to be more stable in flight than had been indicated, suggesting that there were also likely to be significant scale effects involved in these assessments.

In the later tests of an M.52 model in the High Speed Wind Tunnel[88], measurements of moment and downwash, with and without the tail, indicated that the longitudinal stability was beginning to fall at higher Mach numbers, attributed to the increasing lift curve slope of the wing due to compressibility. Extrapolation suggested that stability would become neutral soon after M = 0·82, though this region could not be reached in the tests. Miles considered this determination to be too uncertain to be meaningful, but accepted as a precaution the RAE suggestion that the span of the tailplane be increased by 10% in the final configuration[25]. It was acknowledged that, once flight had become established in the supersonic region, stability would be enhanced by the rearward position of the aerodynamic centre. A higher stability was not expected to be troublesome in the supersonic parts of the profile, where the flight would be mostly straight and level, though the flight plan called for a pull-out within the transonic range if diving was necessary to pass through it. Finally, there was to be a zoom to high altitude to produce the deceleration at the end of a high-speed run.

Balance required that a suitable location should be obtained for the cg, such that trim could be maintained in all phases of the flight. The requirements for subsonic flight were well understood, and there was now enough confidence in the ability of linear theory to represent forces in the fully-supersonic region, to allow an estimate to be made of the most suitable location for the *cg*. From a consideration of the wind-tunnel results it was decided to locate it at 26% of the mean chord, and to maintain that position under all conditions of flight by drawing fuel appropriately from the group of tanks distributed about the central fuselage[70]. If there were to be any problem with the maintenance of control and trim, that was likely to arise mainly from unpredictable changes of the flow in the transonic regime.

A further application of linear theory, that would have been timely for the M.52 designers, was an RAE study in November 1943 by (Arthur) Roderick Collar of the forces and moments at supersonic speeds on a thin aerofoil with a hinged flap, either fore or aft[98]. This gave an important indication that the standard coefficients for control surfaces and high lift devices would have the same signs as in low-speed conditions, and although generally smaller, were of the same order of magnitude.

Calculation of hinge moments and control forces showed that at the highest speeds, they would become very large, reportedly up to 100 times those encountered on contemporary aircraft[25,70]. The pilot's unconventional posture, and the small control movements made necessary by the cramped cockpit, meant in any case that large control inputs would be problematic[99]. Accordingly, it was decided that power-assisted controls would be necessary, giving the important incidental advantage of irreversibility of the control movements. Electrical power on the aircraft was very limited, so although Miles might have employed the electrical actuators developed and patented by the Company, that had many applications, a source of hydraulic control was sought. Miles had misgivings about adapting a system suggested by MAP, and set about developing hydraulic power controls of their own design, specifically for the M.52. This employed hydraulic servo-actuators located close to the control surfaces, to apply the forces required to produce the movement demanded by the push-pull linkages, running in roller bearings, that connected with the cockpit controls. The control surface loads were then reacted locally onto the structure, with enough load at the cockpit controls to provide 'feel'. Hydraulic power was supplied by an engine-driven pump, with an accumulator in the loop, that would provide for temporary actuation in the event of engine failure. The ultimate back-up would be the residual reversion to manual operation, though only at low speeds would this be within the pilot's strength capacity. The system was being run extensively on a ground test rig, and before being cleared for installation was to have been proven in flight by trials in a modified Master II aircraft, though no details of these have come to light.

With power-assistance, irreversibility would make tab control inappropriate, and it was decided that, with control and trim in elevation being obtained through the all-moving tailplane, the other control surfaces could be simply hinged, without any aerodynamic or mass balance. Illustrations such as Fig. 19, of the all-moving tailplane as fitted to the full-size mock-up, show the planes rotating about an axis perpendicular to the fuselage centre-line, with a mechanical linkage in the form of a stirrup to maintain them precisely in unison. This linkage had to move in a very limited space within the fuselage, passing under the ducts of the power plant, which at this point filled most of the cross-section. A difficult compromise is seen here, which had resulted in the positioning of the tailplane on the centreline of the fuselage, perhaps not the most desirable position, as it would usually be operating in the wake of the wing. The two halves could not be joined with a stiff torque tube, as might have been possible if the tailplane had been mounted on the top of the fuselage or on the fin, but this would have led to other difficulties in arranging the actuating arm. With the stirrup arrangement, there was a potential for play and lack of stiffness to enter at that point. But in general the control system promised to be powerful and responsive, though it could not be known how the rapid changes in trim expected in the transonic region could be countered adequately until they were actually experienced in flight.

As a further measure against loss of trim, a pair of dive recovery flaps had been provided. These were 12in long by 3in wide, arranged to hinge outwards from the fuselage under the wing root junctions, to lie in the spanwise direction at the mid-chord position. (Underwing flaps had already been introduced on some fighter aircraft, where pilots had experienced difficulty in recovering from high-

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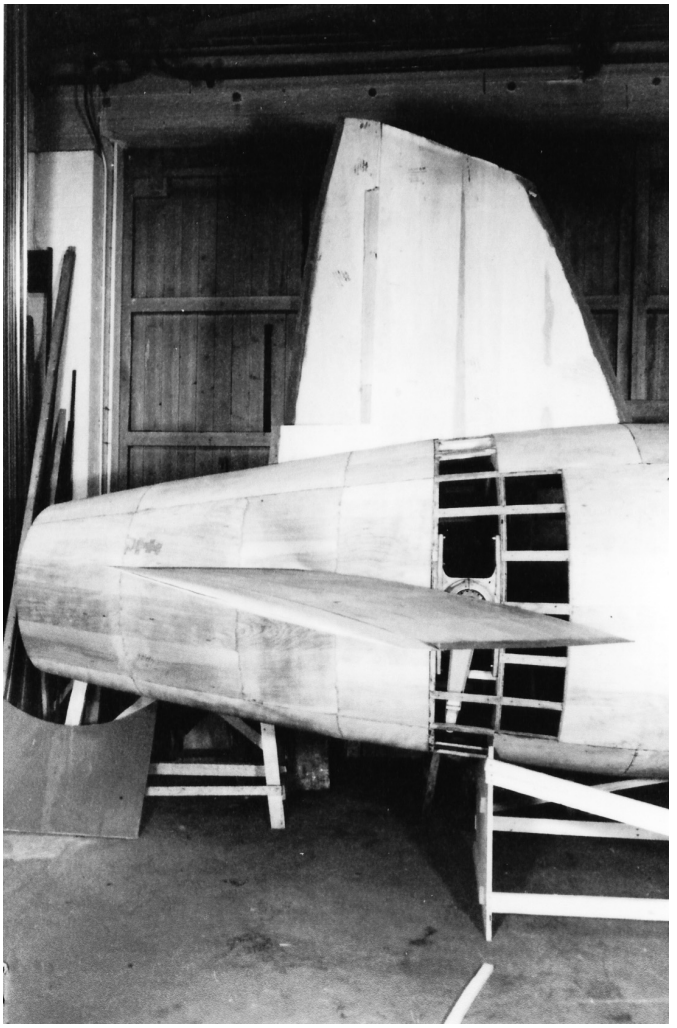


Figure 19. Mock-up of Miles M.52, showing all-moving tailplane in fully 'up' position (The Miles Aircraft Collection).

speed dives. The flap limited the loss of lift at the shock-stall, which otherwise reduced the downwash at the tailplane to the point of producing a nose-down moment that became impossible to counter through the controls). On the M.52, the recovery flaps were arranged to move out under hydraulic actuation when released by the pilot, and then become locked in position. Wind-tunnel tests were possible only up to M = 0·82, but it was concluded that the flaps would produce an adequate lift increment to ensure dive recovery, if opened at a high altitude, and although the aircraft would then have to be returned to base with the flaps still open, the resulting drag and trim change would not be serious at moderate speeds[88].

The proportions of the control surfaces of the M.52 were larger than for contemporary practice from the beginning, probably to provide a margin for the unknown conditions of the transonic regime. Modifications in the course of development did not materially change this. On the wings, for example, the total area of the ailerons was a fraction 0·072 of the gross wing area, compared with about 0·049 on the Typhoon, which had to be highly manoeuvrable as a ground-attack aircraft. The rudder occupied about one-third of the combined area of fin-plus-rudder. The corresponding value for the Typhoon appears to be larger, though on that aircraft it was not so easily defined, as the vertical tail surfaces were blended into the rear of the fuselage, making it uncertain what should be counted as 'fin'. (This difficulty continues to the present time, when several different conventions for determining fin area remain in use). Control movement had to be sufficient to provide for effective operation at the lowest speeds, with ranges of 15º for the rudder, 8º for the tailplane, and 9º up and 6º down for the ailerons[70]. Once in the supersonic regime, small control surface movement would provide forces more than large enough to provide the limited control required in the planned flight profile.

Early in 1944, Smelt produced a first view of the general trends of trim and stability to be expected as flight proceeded from subsonic to supersonic speeds[100]. This showed the combined results of the known effects of the aft movement of the aerodynamic centre of the wing, the disappearance of downwash at the tailplane and changes to the lift curve slopes. These would normally produce a nose-down trim change and an increase in longitudinal stability over the transition region as a whole. The trim change could be minimised by choosing design variables such that the tailplane incidence was positive at subsonic speeds, but the increase in stability was intrinsic in the basic equations. The illustrative values chosen by Smelt would not be exactly applicable to the M.52, partly because of the unusually low lift curve slope for its tailplane. However, there was a general indication that the range of tail incidence provided would be adequate to maintain trim, and no particular concern was expressed about controlling the aircraft in the conditions where the stability was increased.

The prospect of flutter, and fear of its unpredictability, were matters of concern during the flight trials of all new aircraft. Now the possibility of its occurrence at supersonic speeds had to be considered for the first time. In a lecture to the RAeS Historical Group, Collar later recalled someone from Miles visiting him when he was at RAE, urgently seeking advice on the wing stiffness that would be required in this connection[101]. As it happened, the symmetrical biconvex section of the wing and its symmetry in plan meant that the aerodynamic, flexural and inertial axes were all straight and coincident, and this represented the simplest case on which to pronounce. Although a reassurance could be given for this situation, Collar went on to investigate the more general case of supersonic flutter, for the case of a thin wing with forces as given by the Ackeret theory[102]. He found that, as with control surface parameters, the standard derivatives were expected to be of the same sign and similar magnitude to those already established for incompressible flow. In a postscript, however, he referred to other work, which was not published until more than a year later, suggesting that damping coefficients could become negative at high values of the frequency parameter when M was less than $\sqrt{2}$[103]. This was a 'somewhat disturbing prediction', which was not resolved during the life of the M.52 project. Cases of the flutter of control surfaces continued to arise from time to time in all regimes of flight, but the M.52 was to be an early example in which unbalanced surfaces and a stiff power-assisted control system would be expected to keep this in check.

It is seen then that the questions of stability and control had received extensive consideration in the design of the M.52, with due support from theory and wind tunnel, flight and rig testing. This resulted in several significant changes being made to aspects of the airframe. It was a normal consequence of experience that some design modifications would also be needed after the first flights of a prototype aircraft. But apart from the requirement for sensitive handling at take-off and landing, shown by the Falcon tests, it seems that the subsonic regime would produce no more uncertainties for the M.52 than were present in any contemporary design. New theoretical work continued to fill in the background for the determination of the moment derivatives for the lifting surfaces and the fuselage, required for the assessment of stability in the supersonic region. For example, Lighthill had shown that the lift curve slope for a slender body of revolution would be 2 per rad on theoretical grounds, and gave a simple expression for determining the position of the centre of lateral pressure, so that moments and their derivatives for fuselages could be assessed[60].

As in all other respects, the transonic region provided the most uncertainties, but it appears that such provisions as could reasonably be made for these at the time had been carefully considered. Only flight trials could show how well the rapidly-changing conditions that

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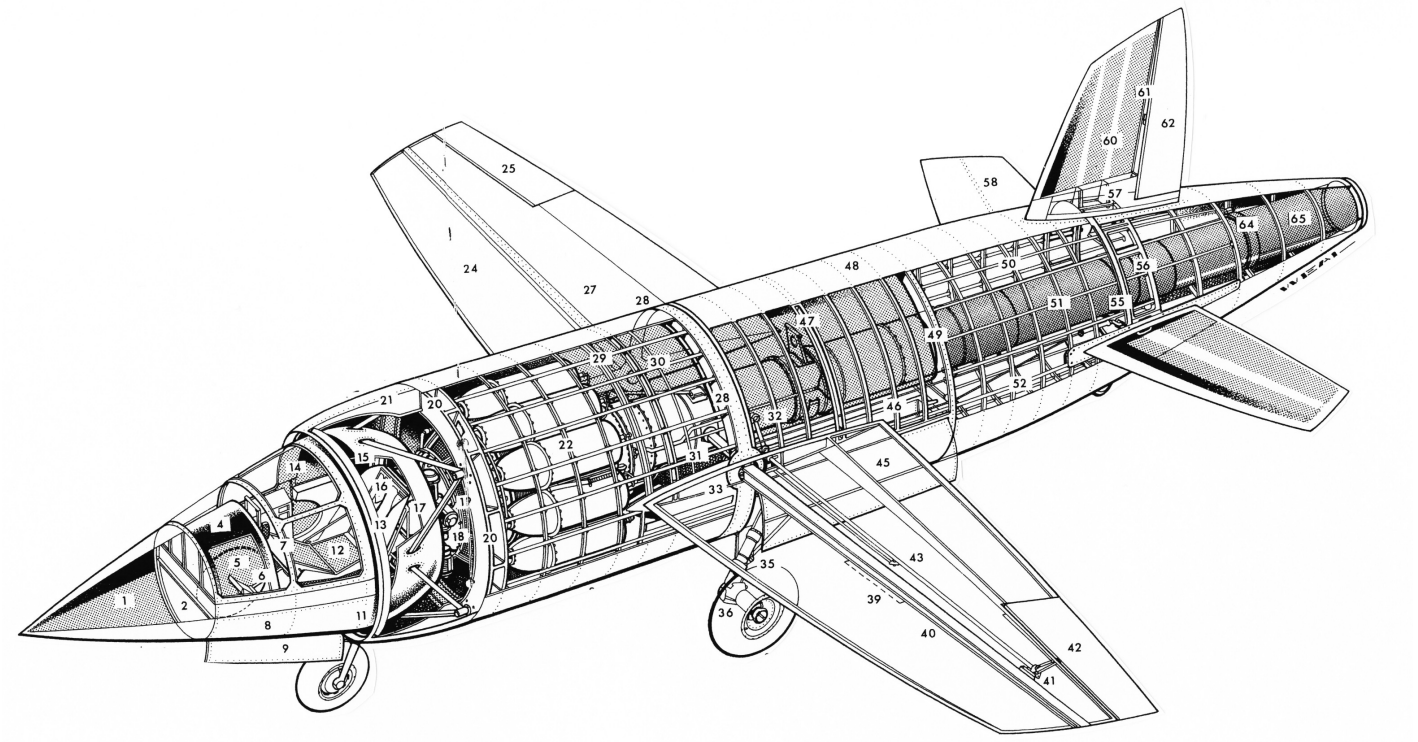


Figure 20. Cut-away representation of the Miles M.52[79] (Pilot Press, 1980).

were expected in this region would be handled. However, some reassurance might have been obtained if attention had been given to another observation in the 1939 paper by Ferri[46]. He noted that the region of fluctuating forces that began when shock waves first appeared came to an end before sonic forward speed was reached, and thereafter the behaviour was steady and repeatable. The indication for a section with the same thickness ratio as that of the M.52 wing was that, for the appropriate lift coefficient, the unsteady region would be confined to the Mach number range from about 0·86 to 0·94.

## 7.0 THE STRUCTURE

The primary purpose here is to review the decisions taken in the field of aerodynamics, but these requirements created new challenges for the structural design of the M.52 also. Some reference to the enterprising ways in which they were addressed is appropriate, as this throws further light on the thoroughness with which the design was carried out at all levels.

It is known that the design case laid down for loading was for a speed of 800mph EAS, with a load factor of 10[104]. The maximum normal accelerations were to be –3·4g and +6·7g, with a forward acceleration of +1·3g and a crash case of –20g. However, the surviving material is purely descriptive, with nothing having emerged on design calculations for the structure and systems to meet these conditions. Materials selected for the construction comprised forgings and extrusions in the aluminium alloy DTD363, castings in the magnesium alloy DTD289, and plate and sheet in the aluminium alloys DTD 603 and 646. EN40 steel was used for the inboard third of the spars for the wings and tailplanes, and for certain fittings.

A cut-away general arrangement of the aircraft is shown in Fig. 20. Although this gives little detail, especially of the wings and empennage, which were the subjects of Miles patents, it conveys a fair impression of the overall conception of the airframe and the challenges of creating a structure that would accommodate all that it had to contain. Whilst for the most part the structural design seems to have followed contemporary good practice, it was marked by notable originality at the component level[70]. Greater use than usual was made of castings and forgings, with care to maintain smooth stress paths, and to keeping jointing to a minimum at points of particularly high load transfer. Novel methods of construction were devised that would ensure that the shapes of the external surfaces of the airframe would be as close as possible to those that had emerged from the requirements of the aerodynamic design.

The thin wing, with its sharp edges, was a test for good structural engineering, but adequate bending and torsional strengths to meet the estimated loading were obtained, when heavy skins of 8swg (0·160in) thickness were used. Each skin was to be formed from a single piece, rolled to a constant radius. The main spar was located at 54% chord, the inboard third being a machined steel forging, terminating in a cylindrical stub which continued into bearings mounted on the fuselage frames. The arrangements at the root, necessitated by not being able to continue the spar through the fuselage, consisted of two forged and machined frames, joined by plate webs, with very robust detailing locally to allow diffusion of the wing loads out into the adjoining structure. The wing moment was reacted jointly on the bearing for the main stub and another for a smaller stub located at the nose of the root section, with a fitting that provided some adjustment on the ground. By this arrangement, the rigging angle of the wings could be varied through 2º, should adjustment be required during the proposed flight test schedule. Five subsidiary spars, full-chord ribs and nose ribs provided the rest of the internal structure of the wing, with extruded sections 4in wide forming the leading and trailing edges, where the thickness was least. Hollow magnesium alloy castings formed the tips. The sections were to be manufactured over-size, and machined to the finished size and shape after the wing structure had been assembled, prior to the fitting of the skins. Since the skins were rolled with single curvature, the spanwise generators of the shape were straight lines, which would allow this final machining of the supporting structure to be done on a planing machine. The three parts of the empennage were constructed in similar fashion, though with single spars, and skins of 12swg (0·104in) thickness.

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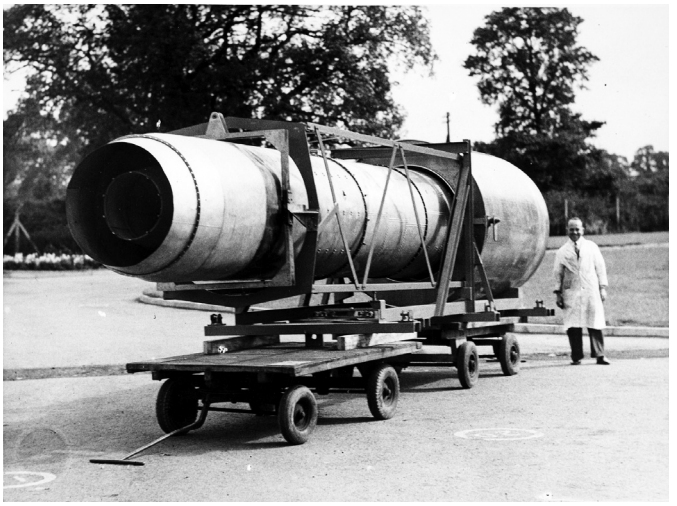


Figure 21. Power plant for the Miles M.52 (Power Jets Ltd W2/700 turbojet engine, with No 4 Augmenter, ready for static test[25]).

For the fuselage, a more conventional structure was adopted, with 22 transverse frames mounted on four longerons, carrying stringer-stabilised skins of 20swg (0·036in) thickness. The frames were also made over-size, and, being circular, could be finished to size on a large lathe, which, before being acquired by Miles, had originally been designed for machining the driving wheels of steam locomotives. Bancroft noted that the fuselage was required to contain a power plant that was 'basically a tube 36in in diameter and 18ft long, chiefly containing flaming kerosene under pressure at up to 1,700ºC [abs]'[22]. The formidable scale of this unit is shown in Fig. 21. An unobstructed internal space, some 40in in diameter, had to be provided, in which the power plant was supported by offset linkages to the frame members described as resembling 'bicycle spokes'. This arrangement enabled the thrust loads to be transferred to the airframe without significant relative movement, whilst allowing for that required by thermal expansion. The hot assembly was to be surrounded by some unrecorded thermal insulation, to protect the main structure, the control runs and other equipment from the effects of the very high temperatures of the power plant[70].

The assembly jig for the fuselage was built with great care, with its alignment verified by optical collimators. The power plant and all substantial internal components were to be loaded though the open front of the fuselage, with the jig mounted over a pit in a vertical position. The skin was to be fitted after all the internal equipment had been installed and tested, using flush pop rivets in which the steel compressing pins were retained, to maximise the strength in shear. Access panels were kept to the minimum required for filling the tanks and reservoirs for fuel, oil and hydraulic fluid.

It is clear that the method of construction intended by Miles would produce a very robust airframe, but one that would represent a higher proportion of the all-up weight than for contemporary service aircraft. This was pointed out by RAE[108], but could not be considered a significant handicap, due to the low amount of fuel required to be carried to meet the specified duty, and the minimum of equipment installed. It was of greater significance that the airframe would not only have the required strength and rigidity to meet the estimated loading requirements, but through the method of construction would show exceptional fidelity of shape.

## 8.0 ANCILLARY TESTING

It has been seen that wind tunnel tests, on component and complete airframe models, continued throughout the life of the M.52 project. Bancroft lists nine models constructed by Miles, for testing in their own tunnel and elsewhere[105]. Complete models ranged from the largest, the 4·37ft span steel model for testing in the RAE High Speed Tunnel (now in the Museum of Berkshire Aviation at Woodley) to the smallest, a 9in span brass and steel model for testing in the NPL 12in supersonic tunnel. Others were made at the establishments, including wing models at NPL and the largest one constructed, a ¼ scale wing model of 6·72ft span. Tunnel tests were supplemented by the air tests at full scale with the Falcon, and the operation of the power-assisted control system was to be validated by flight tests in a Master II. It has not been practicable to give full details of this work here, but all the principal documentary sources are listed in the References that follow.

The contract for the M.52 required two aircraft to be built, with components for a third. Later, a further airframe for structural test at RAE was added, and Miles were authorised to conduct their own tests on wing and tail specimens[104]. Records giving results of any of these tests would be of considerable technical and historical interest, but none has been found so far.

## 9.0 PERFORMANCE ESTIMATION

### 9.1 The objective

By way of review at this point, it could be concluded from the considerations above that the Miles M.52 aircraft had been soundly conceived, with good awareness of the current state of knowledge about the situations to be expected in the various regimes of flight in which it was to operate. Further, the requirements of the aerodynamic design had been well-matched in the realisation of its structure and systems, providing a robust airframe with fair prospects of being stable and controllable throughout the planned flight envelope. Uncertainties would remain about the behaviour to be experienced in the transonic regime, exploration of which would be part of its remit as a research aircraft. Being manned, and capable of normal take-off and landing, it was well-placed to undertake this investigation incrementally, spread over many flights.

Since it had been designed to a specification that laid down particular performance requirements, it was a necessary element in the proceedings to show by estimation that these were likely to be attained. As it was to be the first venture into supersonic flight, the critical factors were the estimation of the drag of the airframe beyond the limits of previous experience, and the prospects of the power plant being able to give sufficient thrust to exceed this. In the event however, there were to be some very serious disparities between the estimates of drag made by Miles and by the Aerodynamics Department at RAE, for which explanations must be sought.

### 9.2 Conflicting drag estimates

The most notable requirement of the specification was that the M.52 was to attain a speed of 1,000mph, or about M = 1·5 at 36,000ft. Then the first assessment needed was of the development of its drag over the full range of speed up to this point.

The methods used by the parties for the subsonic region were not quite the same, and their drag estimates differed somewhat, but in this region they were both low enough not to affect the outcome of performance predictions significantly. In the supersonic region, both relied mainly on thin-wing theory for calculating the wave drag of the wing and empennage, and the differences were again not such as to require closer examination at this point. It was likely that some different assumptions would be made about the wing drag that would be experienced in the transonic region, due to the perceived absence of relevant data, and these must be reviewed. But the most significant differences are found in the estimates of fuselage drag at transonic and supersonic speeds. Though these were based on

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ballistics data in both cases, they were interpreted in different ways, giving sharply-divergent results.

The initial approach to performance estimation taken by the staff at the Aerodynamics Department at RAE was set out in Technical Note Aero 1347 of December 1943[106]. This had been pessimistic about the prospects of the M.52 as originally conceived ; in particular, it had been concluded that the aircraft could only marginally exceed sonic speed, even in a steep dive. It was thought by Miles that this might unreasonably persuade the authorities that even the eventual design would not be able to achieve its specification. Objections to the RAE conclusions were put forward by Julian Hodge of Power Jets, but they were mostly criticisms of the use of the dimensions and features, of both airframe and power plant, for the early stages in the design, which were already out of date[107]. At this time, Hodge did not quote specific performance estimates by Miles or Power Jets, so that the more fundamental nature of the differences that were to emerge later between those of the Company and RAE were not yet exposed.

This first RAE paper was updated in July 1944 by Chrystelle Fougère, who had been co-author of the former with Ronald Smelt[108]. The conclusions were no more optimistic, and they were not revised again for the remainder of the duration of the programme. Hence, this version must be regarded as the considered opinion at RAE, particularly as to the estimation of drag in the transonic region. In the following, this report will be referred to as TN1470.

An outline of the procedures adopted by Miles is given in the final summary of the performance estimates recorded by Bancroft in March 1946, which will be referred to as PE.52[109]. By this time, the project had come to an end, so these estimates must also be assumed to be the considered view of the Company.

Miles had been advised initially by RAE to expect very high increases in drag coefficients on entering the transonic region, and this was reflected in the estimates of TN1470, where the rise in the overall drag coefficient was by a factor of about 10. From their interactions with NPL and ARD, however, Miles came to believe that the rise would be more like a factor of 2 to 4. It is not surprising that they felt more confidence in this, arising as it did from discussion at establishments where there was direct acquaintance with the transonic regime from their long-established programmes in ballistics. Although the Supersonics Committee might have provided a proper place in which these differences could be discussed and reconciled, Miles was not represented on it, so this did not happen. Since the matter has remained open, it is necessary to consider the ways in which the respective views developed, while remaining within the framework of the contemporary state of knowledge.

### 9.3 Fuselage drag

The methods first set out by RAE in TN1347[106] were changed in TN1470[108], and then the change was consolidated in some later studies[110-112].

In TN1347, it is stated that the drag estimate for the fuselage was obtained from the data for projectiles given by Hunt[53], with an unspecified adjustment for differences in Reynolds number and a 40% deduction for the absence of base drag, by reference to the paper by von Kármán and Moore[57]. This gave a peak $C_D$ value, at around M = 1·05, of 0·029 referred to gross wing area, equivalent to about 0·19 on the frontal area of the fuselage, for the dimensions of the Preliminary form of the design. Before reduction for the absence of base drag, the coefficient on which this was based would thus have been about 0·32 on frontal area. As the results in the reference cited are for projectiles with shorter nose forms than that of the M.52, this value derived by RAE was perhaps obtained by some extrapolation from these.

In TN1470, the approach had changed. It was now assumed that the drag coefficient based on the frontal area had a peak value of 0.7 at M = 1. This is said to have been suggested by 'supersonic tunnel results extrapolated', with reference to an unpublished ARC paper by Hilton of NPL, which has not been recovered. There is however some clarification when the topic is next considered in a further study by Smelt, of a possible successor aircraft, basically of the M.52 design, but to be propelled by a rocket engine[110]. Again, the drag of the fuselage is referred to frontal area, but the value is said to be 'taken directly from published data', now specifically referenced to the 1940 Ballistic Tables[59]. The possibility would have to be considered here that in TN1470, the resistance coefficient $f_R$ of ballistics (of which the peak would be about 0·7) had been mistakenly used for the drag coefficient $C_D$ as it would be understood in aeronautics (see Equation (3)). But in Smelt's further paper, it is acknowledged that the peak drag coefficient for a projectile with this nose shape would be 'only 0·5', so this possibility must be discounted. Yet, in this later study the value used for the peak coefficient is not reduced to 0·5, but is raised further to 0·8[110].

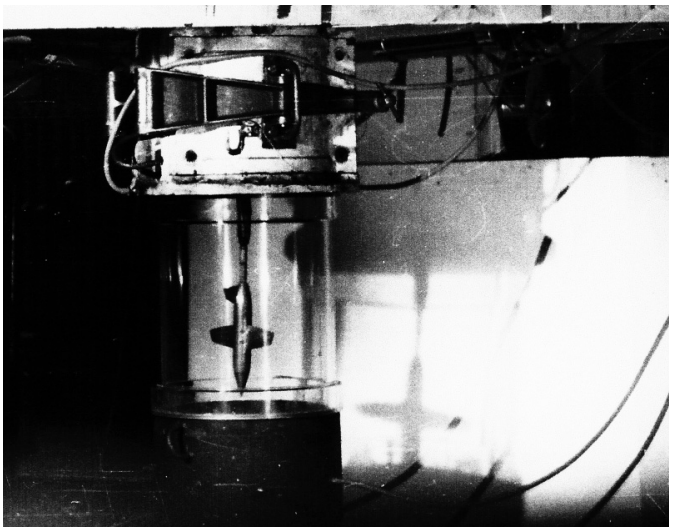


Figure 22. Model of Miles M.52 in the NPL 12in supersonic wind tunnel (The Miles Aircraft Collection).

The only justification given (for RAE having used a value of 0·8 for the drag coefficient, though knowing that the correct value was around 0·5), was that 'it is desirable to overestimate it rather than underestimate it'. For the study of the rocket-propelled aircraft, it was further acknowledged that the base drag component of the projectile data would not be present, but as in TN1470 also, no reduction in base drag was made. This use of the total drag coefficient of a projectile, with a value knowingly exaggerated, and without moderation for the accepted absence of base drag, was continued in further RAE studies by Nonweiler[111] and Brotherton[112] on other possible successor aircraft.

With an assumed peak drag coefficient of 0·7 on the frontal area, the contribution from the M.52 fuselage to the overall peak $C_D$ (referred to gross wing area) would be 0·096, and this is the value given in TN1470. It is more than three times the value estimated in the earlier paper, TN1347, though this increase passes without comment. The very high drag predicted by RAE for the M.52 is principally traceable to this contribution from the fuselage.

The Miles approach to the design of the fuselage began with advice from ARD to use the 5/10crh profile for the nose. Maccoll is also cited as the source of the drag data used by Miles, as would be expected. However, in the Miles final report PE.52 a different approach to the estimation of the fuselage drag is given[109]. This is said to begin with a result from tests in the 12in supersonic tunnel at the NPL. An illustration of the tunnel, with an early model of the M.52 in position, is given in Fig. 22. However, it cannot now be

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established whether the fuselage drag was derived from results with a complete model, or with one of the fuselage only. In the Miles paper on models, the description of the one for supersonic tests shows that it had a brass body and steel wings and empennage, so it could have been used in partly dismantled forms[105]. In Fig. 22, the model and its shadow show that at that point it is fitted with the fin, but no tailplane. The only reference to the results of these tests states that they were conveyed to Miles in a letter from Hilton, dated 28 June 1944, which has not been recovered[109].

As a different nozzle was required for each Mach number, tests could be made only at two values at this time. It is stated that a value of the drag coefficient at M = 1·45, the higher of the two obtainable, was used to define a datum value for the fuselage drag. Then, several corrections were applied to the measured value, to allow for changes to the length and diameter of the fuselage from those of the early version of the aircraft that had been modelled. However, these corrections are not given sufficiently clearly to allow the original measured value to be recovered, so that they cannot be directly compared with data reported earlier for projectiles.

The drag curve, from the start of the transonic region, was constructed using that of a 'bullet', which was then positioned by scaling it to pass through the adjusted point for M = 1·45. The curve used for this was the one reproduced in Lock's paper of 1938 and included here in Fig. 7[6]. At subsonic speeds, the curve was then blended into the curve that had been previously calculated for the early part of this region. When obtained in this way, the peak value of the fuselage contribution to the overall $C_D$, referred to the gross wing area, was about 0·027. This peak drag coefficient was not reached until well into the supersonic range, at M approaching 1·4.

This procedure is precarious. Firstly, the positioning of the fuselage drag curve would depend entirely on the value estimated from the model test at a single supersonic speed, through which it is scaled to pass. It was acknowledged that there had been a specific warning from NPL that the results from which this value was obtained were subject to uncertainty, since the tunnel had not been calibrated at that time[22,109]. [It appears that these tests were among the first to be made in this tunnel, following its conversion in November 1943 to supersonic operation, for estimation of the performance of the German V2 long-range ballistic missile[51]].

Secondly, and probably of greater significance, the shape of the drag coefficient curve, taken from Lock's paper, suggests that it had not been an appropriate choice to represent the drag of the M.52 fuselage. As shown in Fig. 7, this curve exhibits an early start to the drag rise, a moderate curvature and a late occurrence of the peak, with only gradual change on either side. These, and the high maximum value of more than 0·7, are the characteristics of a bullet with a rounded nose, and not a 'sharp-nosed' one, as it had been assumed to be[22]. With a sharp nose, its drag curve would have resembled those seen in Figs 8 and 10, with a much steeper initial rise, coming to a peak soon after M = 1, and a decrease of exponential shape thereafter. Unfortunately, no reference for the source of the curve was given in the paper by Lock, and so far its origin has not been traced, so that the precise head shape of the bullet to which it refers remains unknown.

The peak of the Lock curve comes at about the same Mach number as that of the adjusted point of the NPL wind-tunnel test, through which it was scaled to pass in the procedure reported. This has the result that the estimated drag coefficient values are lower than at this reference point at practically all other speeds. Further, they are then everywhere below the minimum value for projectiles, as calculated by Lighthlll for the wave drag only[60].

Awareness of Lock's curve might have arisen in discussion by Miles staff with personnel at NPL, including Lock and Relf. But there seems to have been no good reason to advise Miles to use anything in their estimates other than the drag data that they had been given by Maccoll, specifically for a body with the 5/10crh nose shape that they had adopted on his recommendation. This inconsistency remains unexplained,

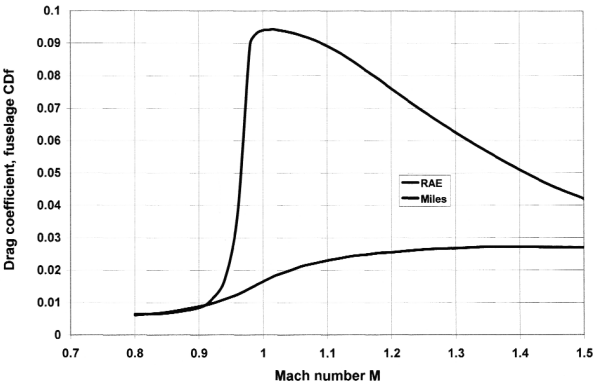



Figure 23. Drag coefficient of Miles M.52 fuselage, as estimated by RAE (upper curve)[108] and Miles Aircraft Ltd[109].

In Fig. 23, the two final predictions of fuselage drag coefficient are shown together. The effect of the different approaches to this matter by RAE and Miles caused their peak values to differ by a factor of about 3·5, and to occur at very different Mach numbers. It is clear from the respective records that they were driven apart through specific procedures that had caused the RAE prediction to be higher, and the Miles one lower, than the available information would indicate.

### 9.4 Drag of wing and empennage

Tests of a model wing at zero lift in the RAE High Speed Tunnel showed that there was no significant rise in drag coefficient up to the highest Mach number of 0·82 that could be used there[86]. By May 1944, these were supplemented by results from tests in the 12in tunnel at NPL with a smaller biconvex aerofoil with a thickness ratio of 7·5%. These covered the subsonic range up to M = 0·875, and individual results were obtained at the two available speeds of M = 1·25 and 1·45 in the supersonic regime[113].

For this somewhat thicker case, at a $C_L$ of about 0·2, around that of the M.52 in the high subsonic speed range at 36,000ft, the drag coefficient began to rise noticeably from about M = 0·7, increasing steeply to reach a value of around 0·04 at the limit of the subsonic tests. But It had already been recognised that wind-tunnel measurement of the onset of compressibility effects did not correlate well with those being obtained in the diving trials at full scale[44,114]. And so, although not specifically referenced, it is likely that for continuation into the early transonic region results would have been based on those from the diving trials with Spitfire and Mustang aircraft, following the example of TN1347. But the more significant problem was to estimate how high the wing drag coefficient would rise subsequently, and at what Mach number the peak would occur, before falling to the values that were again amenable to calculation in the supersonic regime.

It is seen from TN1470 that it was fully accepted by RAE at that time that the drag of a wing at around sonic velocity was determined mainly by the thickness ratio. However, the method employed to obtain a numerical value for the peak drag coefficient was to assume that the same procedure used for the drag contribution from the fuselage would apply also to the wing. That is, that the drag of any component, fuselage or wing, was proportional to its frontal area, and with the same constant of proportionality. For this to be so, the '3D relief', due to the greater freedom of the air to move around the nose of a body, would not be present in transonic flow. It is not recorded why it had been thought that this effect would disappear, though it was known that its presence in early compressible flow gave lower drag for a body of revolution than for a transverse 2D

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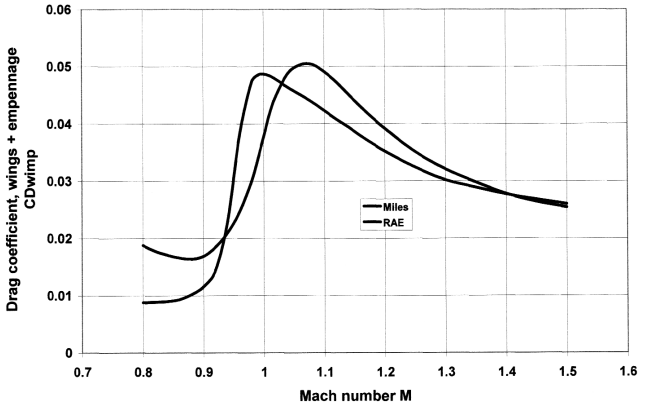



Figure 24. Drag coefficient of Miles M.52 wings plus empennage, as estimated by RAE (lower peak)[108] and Miles Aircraft Ltd[109].

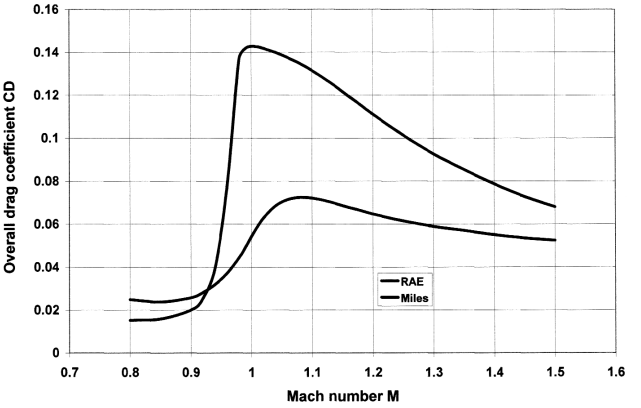



Figure 25. Overall drag coefficient of Miles M.52, as estimated by RAE (upper curve)[108] and Miles Aircraft Ltd[109].

cylinder of the same profile, like a wing, and in supersonic flow, it resulted in the drag coefficient of a cone being lower than that of a 2D wedge of the same included angle[6,55,56]. Unless the 3D relief was uniquely absent in the transonic region only, the peak drag of an essentially two-dimensional wing would be underestimated substantially by basing it on data for a three-dimensional body.

If, as assumed by RAE, the drag of any body in the vicinity of the sonic speed is proportional to the frontal area $A$ that it presents to the flow, the corresponding drag coefficient referred to the gross area of the wing becomes:

$$C_{D_o} = \mathrm{K}(A/S) \quad \ldots (12)$$

where for the peak drag the constant K was taken to be 0·7. It is readily shown that if that also applied to a wing, the standard drag coefficient based on gross wing area would reduce to:

$$C_{D_o} = 0{\cdot}7(t/c) \quad \ldots (13)$$

Thus, the peak drag would indeed be proportional to the thickness ratio, and on this basis, the value for the M.52 wing would be about 0·042 (6% thickness ratio had been assumed). RAE took it that only the nett area of the wing outside the fuselage would contribute to the drag, and obtained a final value of about 0·037. The contribution of the empennage was obtained on the same basis, giving a combined value of about 0·049. But two opposing factors have to be considered. As indicated above, a proper interpretation of the ballistic data would have given a much lower gradient K than 0·7 for the fuselage shape. Secondly, acting in the opposite sense, the 3D relief available to the flow around a fuselage would suggest that the gradient would be higher for an essentially 2D object like a wing. Though these contrary factors might cancel to some extent, it is clear that the procedure used by RAE was basically flawed, and without justification on physical grounds.

For the supersonic region, the peak value was blended with drag coefficients from the NPL tests at two Mach numbers on the 7½% thick wing, with guidance from Ackeret theory to obtain the shape of the curve over a range of speed. It is difficult to tell from the published curve whether any allowance was made for the actual wing being thinner, or for the contribution due to lift, as these would again act so as to cancel each other. RAE made no specific correction for skin friction, perhaps on the assumption that this would be suitably represented in the wind tunnel values. The final estimate of the drag curve for the M.52 wing was made by blending the low and high speed parts so as to pass through the peak value derived from the ballistic argument.

The corresponding estimate by Miles was also obtained by blending the curve of drag coefficient in the supersonic regime, given by the Ackeret theory, with a steeply-rising curve from the subsonic side. This was accepted to be a matter of judgment, in the absence of a physical basis to provide guidance as to the magnitude and location of the peak value. Although acknowledging that the NPL and ARD experience, which coloured their own conclusions, had been with bodies of revolution, like RAE they assumed that the transonic drag rise for wings would be of a similar order[109]. Numerical values can be read more easily from the Miles papers, from which it appears that a condition of zero incidence was assumed for the supersonic region. With a specific allowance of 0·004 for skin friction added, the peak value is estimated to be about 0·035, coming at about M = 1·07. With the corresponding value of about 0·016 obtained for the empennage, the combined coefficient for the lifting surfaces would be about 0·051. These values are referred to the gross wing area. Thus, although obtained by different processes, the peak value is similar to the estimate obtained by RAE.

The two estimates are compared in Fig. 24. Though the magnitudes are similar, the differing locations of the peak would have a significant effect. Being based on sharp-nosed projectile data, the RAE peak is placed close to M = 1, after a steep rise, while that of Miles comes later, as their assumed curve was generally more rounded.

Figure 25 shows the two overall drag coefficient curves predicted for the whole aircraft, represented by the sums of those in Figs 23 and 24. These correspond with the totals given by RAE and Miles. A (conventional) contribution of 7% was made by Miles for interference effects in the subsonic region, but there was no information at the time by which either party could base an estimate for that in supersonic flight.

## 9.5 Review

### 9.5.1 *The 'McColl' curve*

RAE and Miles Aircraft Ltd both went on to give curves derived from the above coefficient estimates, showing their calculations of the overall drag of the M.52 at 36,000ft. These are brought together in Fig. 26.

Also shown in this Figure is a curve labelled 'McColl', which had been plotted with the Miles results in PE.52[109]. There is no reference to it in the text of the report, and its authority is uncertain. But the contribution made by Maccoll during the conception of the M.52 could have been very significant, so it would seem unwise to overlook it. When Power Jets representatives visited him at the

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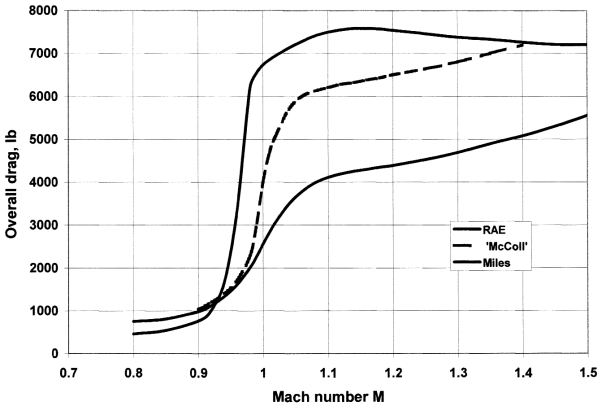



Figure 26. Overall drag of Miles M.52 in level flight at 36,000ft, as estimated by RAE (upper curve)[108] and Miles Aircraft Ltd[109]; dotted curve as labelled 'McColl' in Ref. 109.

beginning of the project, he offered to make an estimate of the drag of the whole aircraft, including 'the wing fuselage interference effects which he thought it was possible he could do'[69]. Then, in June 1944, with two colleagues from ARD, he met personnel from Miles and Power Jets at Whetstone, specifically to discuss the estimation of drag[115]. On that occasion, his estimate of the fuselage drag for the preliminary form of the aircraft was around 4,000lb, at M = 1·5 at 36,000ft, corresponding to a fuselage drag coefficient (gross wing area basis) of about 0·037. The total drag for the whole aircraft would be about 6,000lb. Estimates made by Miles and Power Jets were stated to be very similar to these figures at that time.

A presentation from Miles at a meeting in February 1945 refers to estimates from Maccoll as being the only data on the subject considered to be reliable. Results obtained by calculation would require use of the 'method of characteristics', that Maccoll had employed with Taylor[55]. Involving much numerical work, this would entail 'approximately three months' work to predict the drag of any given shape'[116]. Nevertheless, it appears that he had undertaken the task, as in the account of the wartime contributions made by Miles, it is reported that 'Dr MacColl has estimated the pressure distribution round the fuselage supersonically but so far this information has not been received'[25]. That report is undated, but as it also refers to information obtained from Germany, it was probably written during the second half of 1945. It would be conceivable then that the drag curve labelled 'McColl', in the final performance report PE.52 of the spring of 1946, could represent this final estimate, having by then become available. However, it must be added that in discussion with the author, Dennis Bancroft has been unable to confirm this provenance from his recollections of events at the time.

It is seen above that the estimates of wing drag were similar by both parties, though on a basis that would be likely to make them both too low, but there are greater concerns about the processes for estimating the drag of the fuselage. Since there were reasons for supposing the RAE estimate to be too large and the Miles estimate too small, the location of the 'McColl' curve, coming between them, might indicate that perhaps it had a more secure foundation. It seemed then that it could be of interest to see if this curve would be reconstructed, if a further estimate of the drag coefficients was made, without the uncertainties that have been noted, though still based solely on the state of knowledge at the time of the project.

### 9.5.2 Fuselage drag

A new estimate of the drag of the fuselage is therefore obtained from the data of the 1940 Ballistic Tables for a projectile, having the same nose shape as that adopted for the M.52[59]. The sharp cut-off at the base of the projectile would be expected to give a broad wake and high base drag, which would not be present on the fuselage. The only work published at the time, by which the base drag might be estimated, was that of von Kármán and Moore[57]. In this, the flow is assumed to make an isentropic expansion from the rim of the projectile's (full diameter) base, giving a pressure drop leading to a base drag contribution expressible as

$$C_{Db} = C_{Do}\,(1 - C_{D}oM^2/4) \qquad \ldots\ (14)$$

where $C_{Do}$ is the profile drag coefficient at low speeds, after subtraction of the skin friction component. For the case of the Standard Projectile, this is about 0·18. The implication that there is some continuity of base drag through the turmoil of the transonic region might be questionable, but there was nothing else to go on at the time.

For the M.52 fuselage, a large part of the base area is occupied by the exhaust nozzles, which would contribute nothing equivalent to base drag, since the pressure in the exhaust flow was to be included when calculating the thrust of the power plant. For the versions of the machine fitted with the thrust augmenter shown in the Miles drawings, the exhaust areas range from about 25% to 38% of the presented area of the body, so that a significant part of the base drag contribution for the projectile should be subtracted on this account. Another difference from the Standard Projectile is the shape of the rear fuselage. Its curvature more closely resembled the boat-tail form that was used for some shells, which reduced the base drag by not having the sharp cut-off, giving them distinctly longer range in flight[61]. On the other hand, the flow expanding over the afterbody of the aircraft had to accommodate a complex pattern of shock waves from the fin and tailplane. It is apparent from post-war publications that much work had been in progress at ARD on the estimation of the flow at the base of projectiles[50], and this would have been known to Maccoll, but it seems likely that only a rough judgment could be made at the time about how much to include in the fuselage drag estimate. For present purposes, it is supposed that the base drag would be assumed to vary with Mach number as in Equation (14), with the value for $C_{Do}$ of 0·18. Then, that half of this base drag should be subtracted before applying the tabulated data to the fuselage.

For skin friction, Prandtl's boundary layer equation is again used to correct the drag coefficients for the difference in Reynolds number, though the adjustment from a projectile at sea level to the aircraft at altitude is not great.

The peak value of the fuselage drag coefficient, referred to gross wing area, is then estimated to be about 0·050, coming at around M = 1·07. There would follow a gradual decrease to about 0·042 at M = 1·5.

### 9.5.3 Wing and empennage drag

For the subsonic region, the values are not recalculated, the Miles evaluations being utilised, as they were thoroughly argued.

In the supersonic region, both drag estimates relied mainly on thin-wing theory, but a fresh set of drag coefficient values for the wing and empennage is calculated, using the Ackeret equations of Section 5.4.2, including the lift-dependent contributions. As before, the skin friction components are evaluated using Prandtl's boundary-layer equation for flow on a flat plate.

No data had been cited by either party for the transonic region, between Mach numbers of say 0·95 to 1·2. Sketches show that various plausible curves could be fitted between these limits, depending on how 'peaky' they are chosen to be. Thus, if any estimate could be found by some means, in the vicinity of M = 1, this ambiguity could be reduced. Given the impossibility of obtaining any directly-measured values for wings, the only other sources from which some guidance might have been sought at the time seem to be the tests made in the 1920s on aerofoils for propeller blades, and the open-jet results reported in 1938 by Ferri.

It is noted that, at the meeting of the Supersonics Committee of 10

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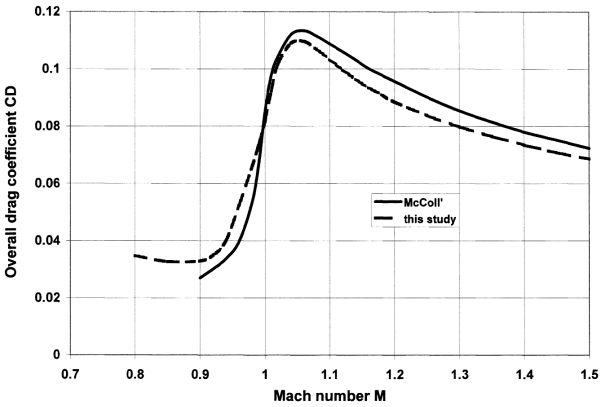



Figure 27. Overall drag coefficient of Miles M.52, as derived from 'McColl' curve (full line)[109] and estimate from this study (dotted).

August 1943, Ronald McKinnon Wood recalled the 'old RAE work on propeller blades with bi-convex sections' and thought that for a 10% thick blade the drag coefficient at sonic speed was 'of the order of 0·08'[7]. This would have been a reference to the results of Douglas, Perring and others, which were mentioned in Section 5. He was asked to prepare a note about this, but there is no record of it at later meetings, and it does not appear to have been followed up, so what might have been a significant opportunity was missed.

As the results had been few in number and rather scattered, It was perhaps thought that they would not allow a trend to be assigned, that would enable the drag coefficients of thinner sections to be estimated. However, guidance on this could have been obtained from the firm trend with thickness ratio shown by the results of Briggs and Dryden, as in Fig. 6. Due to the different experimental methods employed, it was expected that the results of Douglas, Perring and others would be larger than would be given by this curve, and they are so, by an average factor of 1·57. If that ratio applied at lower values of thickness ratio, it would suggest that, for the thinner sections, where the relation is linear, the peak drag coefficient would be approximately $1{\cdot}0(t/c)$. For the mean thickness ratio of the M.52 wing, this would be about 0·062. This is supported by the results of Ferri for a section of the same thickness ratio, from which the best estimate at $C_L = 0{\cdot}15$ at M = 1 would be about 0·059[46].

In both the Miles and RAE estimates, drag was assumed to be generated only by the nett (exposed) area of the wing, and if that is also assumed here, the value would be about 0·048. Together with values for the empennage, and after adjusting for the change in skin friction from tests with small blades at sea level to the lifting surfaces at operational height, the peak for the wings and empennage becomes about 0·061. The little available evidence, such as that shown in Fig. 5, would indicate that the peak would occur at a Mach number of around 1·05.

#### *9.5.4 The overall drag coefficient*

The estimated contributions to the overall drag coefficient of the fuselage, wings and empennage obtained in the above manner would, at this distance, appear to represent a fair estimate of the drag coefficients for the M.52 that could have been obtained on the basis of contemporary knowledge. The corresponding total drag coefficient is shown in Fig. 27, and compared there with that derived from the 'McColl' curve. It is seen that, although the agreement is only fair in the subsonic region, they become close beyond sonic speed, where the 'McColl' curve has a similar form, though rising above that obtained by the methods given above, by about 7% overall.

A small difference could result from a multitude of factors. One possiblility is that, from his better knowledge, Maccoll might have calculated a base drag contribution that would be more than has been assumed here. Another possibility is that he had included a contribution from interference effects. Maccoll had told representatives of Power Jets in 1943 that he might be able to estimate these, but no record has been found that would indicate what method he could have used. Immediately after the war, Maccoll reported some of his wartime work (on the nose drag of projectiles in the transonic region), and he would have been aware of other relevant work at ARD, such as the modelling of the base flow[50]. Work reported on interference at the junctions of projectile bodies and fins must have been started at this time also[117].

Another possibility is that the drag of the M.52 nose could now be considered in two parts. As illustrated in Fig. 11, the shape of the nose, although based on the same ogival form, is not identical with that of a projectile, being interrupted behind the pilot's capsule, to form the annular intake, surrounded by the intake shroud. Estimation of the drag of the nose in two parts, the capsule and the shroud, had been considered in a report of Power Jets Ltd in 1944[118]. This is the only known contribution by participants in the project to the theory of external supersonic flow. In it, they introduced a modification to the approximate method of von Kármán and Moore, which gave results in good agreement with the calculations for an ogival nose, and also with those of Maccoll for cones. They went on to apply this to a body having a nose with an annular intake, which resembled that of the M.52, but differed in the shape of the afterbody. The Power Jets report makes clear that they were in discussion with Maccoll about their procedures, so it is possible that the latter might have made a final calculation in which the nose form was also treated in two parts. The discontinuity in the shape and the formation of a weak secondary shock at the intake lip might then have added an increment to the drag, relative to that of a plain ogival nose.

And so, in relation to the other leading figures in the matter, Maccoll would clearly have had the most comprehensive knowledge of the relevant factors, and to have been involved with the problem throughout the duration of the project. However, the only available drag curve associated with his name is the one attributed to 'McColl' in the Miles final report. Although uncertainties must remain, with the further support from the re-evaluation of the available information in the present study, it could reasonably be conjectured that this might represent the most robust estimate for the M.52 that was made at the time.

## 10.0 THRUST AND SPEED

### 10.1 The available thrust

In the Whittle-inspired power plant devised for the M.52, Power Jets offered an innovative concept to match the novelty of the Miles airframe. Employing bypass flow and secondary combustion, it was the most promising development of the new technology of turbojet propulsion that could be foreseen in 1943. The performance expected in the early stages of the project is set out in detail in a Power Jets report of January 1944[119]. This shows clearly the enhancement expected to be obtained from the addition of the augmenter. The W2/700 core engine had at that time a static thrust of 2,000lb at sea level. If increasing thrust was required in the climb and acceleration, the bypass flow could be provided with a sufficient pressure rise to give reasonable cycle efficiency by the outboard compressor blades on the augmenter rotor. At the higher speeds, the pressure rise due to ram compression in the intake would become the dominant component, bringing the plant closer to the concept of the propulsive duct, around which the possibility of supersonic flight had first been envisaged. Within less than three years of the first flight of a Whittle engine in the Gloster E.28, the thrust expected at 36,000ft was as shown by the lower curve in Fig. 28, rising to over 4,000lb nett at 1,000mph.

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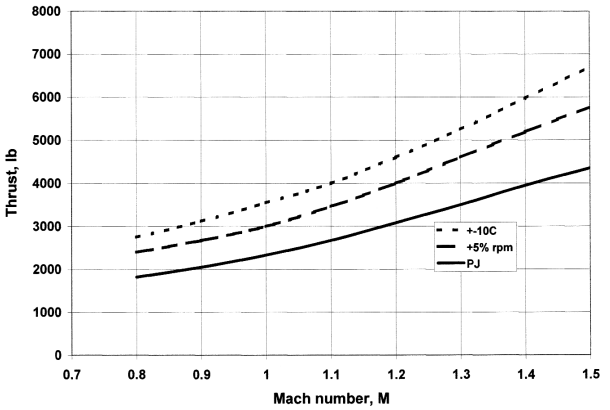



Figure 28. Thrust of W2/700 engine with No 4 Augmenter at 36,000ft. Lower – ICAN Standard Atmosphere[119]; (Power Jets Ltd) middle and upper – 5% rpm increase and 10°C lower ambient temp[109] (Miles Aircraft Ltd).

However, at the meeting between representatives of Power Jets, Miles and ARD in June 1944, there was general agreement that the drag to be overcome at this speed would be around 6,000lb, and that without further measures 'level flight at M = 1·5 will not be attained'[115]. Some increase in thrust would be possible if the outer duct diameter could be enlarged, and Miles shortly agreed that an increase of 3in was possible at the limiting point in the region of the rear fuselage near the tailplane. The lower curve in Fig. 28, which was given in the Miles paper PE.52/1 of 1946[109], shows an increase of about 8% above the thrust being predicted by Power Jets in 1944, perhaps reflecting the effect of the increased duct diameter.

Power Jets went on to consider developments of the power plant, aiming to 'provide a thrust that would definitely exceed the present estimated requirements'[115]. Possible drag reduction measures were considered also, such as 'increasing the fineness of the nose', but Walter Capley of Miles insisted that it was not then possible to 'envisage any alterations that would delay the completion of the airframe or engine'. Here, he was probably expressing anxiety about the growing time-scale for the project. Although it must have been obvious from the beginning that the original requirement for delivery of the aircraft within nine months was absurdly optimistic, it was perhaps already being perceived that the longer its completion was delayed, the less secure the project would become.

In the final RAE report, TN1470, the estimated drag was so great that there was concluded to be 'no hope' of the M.52 reaching sonic speed in level flight. Although it might just do so in a steep dive, the thrust curve would not cross the predicted drag curve thereafter, so that level flight could not be obtained in the supersonic region. A US visitor, having viewed the project at Miles, and discussed it at RAE with Smelt, is reported to have understood that its performance had been 'greatly handicapped by the power plant specified'[120]. This was unjustified, as no other power plant existed at the time that was within reasonable development distance of providing enough thrust to sustain flight over the full performance envelope. A similarly uncompromising conclusion about the prospects of the project seems to have arisen in the minds of the Supersonics Committee also, despite the RAE report having taken a more cautious line elsewhere. It goes on to state, for instance, that 'No great accuracy is claimed for the results, for there is still doubt about the drag and the engine performance', and even that 'The results are qualitative only. . '. Finally, it is conceded that 'The speeds likely to be reached by the Miles E.24/43 are sufficiently high to be interesting, and valuable results could be obtained from the design'[108].

The drag values agreed by Miles and others in 1944, rising to about 6,000lb at M = 1·5, are reflected in the curve in the Miles final report[109]. Being much lower than those of RAE, they were certain to lead to very different conclusions about the achievable performance. At this time, these calculations were of what they expected to be achieved when flight trials with the second prototype were likely to begin. For this, they could take into account the increases in thrust that would result from measures which had been identified earlier by Power Jets[115]. These included an increase of 5% in engine rpm, made possible by the use of the new Nimonic 90 alloy then becoming available for the turbine blades. Miles also suggested choosing occasions for flight when the ambient temperature at altitude was 10°C lower than for the ICAN (International Commission on Air Navigation) Standard Atmosphere, which had previously been assumed. Variation due to changing the area of the exhaust nozzle for the augmenter flow was also investigated. With the optimum choice of nozzle area, the effects of the increased rotational speed and reduced ambient temperature are shown by the middle and upper curves in Fig. 28[109]. Relative to the original thrust values, these represent increases of 30% and 15% respectively. In the text of the report, the increases are attributed to the thrust varying as the fourth power of the rotational speed and inversely as the square of the ambient temperature, though the values plotted are in fact about a third greater than those rules would have given. The actual increases would be determined by the gradient of the equilibrium running line of the power plant.

Miles estimated the performance of the aircraft from the relation between this enhanced engine performance and their final drag estimates. With the thrust and drag as given, the maximum level speed was found to be about M = 1·05. However, the thrust would again exceed the drag from about M = 1·15 onwards. The intermediate region could be traversed in a very shallow dive from near the aircraft's ceiling, after which level supersonic flight would again become possible, and the speed could then be taken up to the maximum specified value of 1,000mph (M =1·5).

The drag curve attributed to Maccoll was steeper in the transonic region than the estimate given by Miles. If that had been used instead, it may be seen by superimposing Figs 26 and 28 that the maximum speed predicted in level flight would have been very close to M = 1. But as the thrust curve would not cross the 'McColl' drag curve after this point, level supersonic flight would not be attainable. It is however relevant to consider the potential performance in a dive with this drag estimate, since that was in any case an element in the flight test schedule considered by Miles.

### 10.2 Supersonic excursions

The procedure used by Miles is given in the final report PE.52[109]. The dive is entered at 50,000ft at a high subsonic speed, and it is assumed that a constant dive angle is maintained until the beginning of a pull-out, at the point required to return to level flight at 36,000ft. This ending to the dive was necessary, since the Specification required the performance to be demonstrated at this altitude. In corresponding calculations made by RAE, the dive is modelled from the beginning, also in level flight at 50,000ft, by assuming that the aircraft is flown along the curved path resulting from a condition of zero lift (described as a '0g' dive)[106]. The dive is continued down to about 36,000ft, and followed by a pull-out at 3g, in the denser air of the troposphere. This manoeuvre, from a realistic beginning to the dive, in a condition that would be fairly comfortable for the pilot, is also adopted here. An all-up weight of 7,500lb is assumed to apply throughout. The full thrust of the power plant at 36,000ft, as estimated by Miles, is used, and factored for other altitudes in a region in which the ambient temperature is 10ºC below that of the ICAN Standard Atmosphere. The equilibrium running diagram of a turbine power plant and the drag equation for an airframe show that, at a given Mach number, both quantities are proportional to the ambient pressure. It is a characteristic of the Standard Atmosphere that the pressure at a given altitude in the stratosphere does not change with a change in temperature, so the ICAN values are applicable. Minor discrepancies in the modelling, through the neglect of the reduction of weight due to consumption of fuel and the retention in the drag data of a contribution due to lift, have effects of opposite sign, and are neglected here.

With the chosen form of the manoeuvre, the progress of a dive would be as shown in Fig. 29 if the thrust and drag had been as estimated above. The speed obtained on passing through the 36,000ft level would be M = 1·26, with the dive angle having reached 46º at this point. The total

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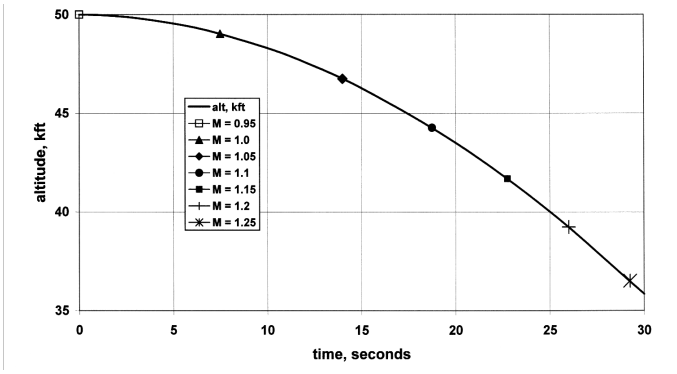



Figure 29. Estimated dive performance of Miles M.52, thrust as Fig. 28, drag as Fig. 27.

elapsed time in the dive is around 30 seconds, so that, with the pilot having to concentrate on maintaining the specified flight condition, the need for the 'automatic observer' to record the flight data is evident. Though the dive is becoming steep, there would be no operational need to terminate it at this level, but the estimation is not continued into the tropospheric region, because of likely inaccuracy in revising the available thrust estimates for an atmosphere of varying temperature. In a practical experimental programme, it would be expected that the aircraft would be flown in a sequence of gradually steepening dives, beginning with very shallow angles. With experience, these might well be taken down into the troposphere before beginning the pull-out, with some further increase in maximum speed.

The above summary is as far as the possible flight characteristics of the M.52 can be given within the framework set for this study, that is, as they could have been estimated within the state of knowledge at the time. The true potential of the machine could become clearer only when the power plant had been fully developed and tested, and understanding of aerodynamics in the transonic region had matured and could be applied with more confidence. Successful flight trials with the M.52 would themselves have contributed to advances in both of these areas. Aerodynamics research could not resume its historic progression until the transonic wind tunnel was developed, but meanwhile, flight trials could have provided both an essential input to the modelling of this most complex flow regime, and much-needed data for validation of tunnel corrections as possible arrangements were investigated. Proposals for continuing development of the M.52 airframe and power plant were considered, but not put into effect, as the project was terminated before these could be worked up.

## 11.0 POSTSCRIPT

Prior to the ending of the M.52 project, teams following the advance of the Allied forces in Europe were able to examine surviving hardware and the work of research institutes and other centres in Germany as they were overrun. It would now become possible to establish the veracity of the prisoner-of-war's report that had brought the E.24/43 project into being.

It was soon found that extensive work had indeed been in progress on propulsive ducts. In accordance with Hitler's policy of 'divide and rule', work was carried out in different institutes, responsible to different ministries, and without coordination. It seems however that the most work towards applications was that carried out at the DFS (Institute for Gliding Research), Ainring, under the direction of Eugen Sänger[121]. This established the basic proportions required for ducts to have high stage efficiencies, supported by testing with a variety of fuels, including powdered coal and coke, reflecting the looming shortage of liquid hydrocarbon fuels. Tests with ducts mounted on ground vehicles were followed by others with diameters up to 1·5m (4·9ft) in flight, at true speeds up to about 200ms$^{-1}$ (450mph) and altitudes up to 7km (23,000ft). These trials were made with ducts mounted above the fuselage on

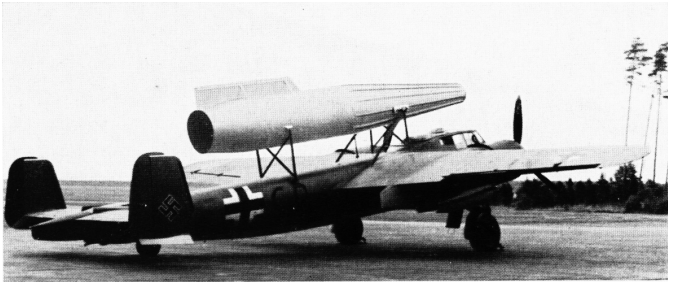


Figure 30. Dornier 217 E.2 test aircraft, with propulsive duct 1,500mm (4·9ft) diameter x 10·6m (34·8ft) length[121].

Dornier Do17 and Do217 aircraft, and so were the most likely origin of the prisoner's reference to the 'Do. Geräte' (see Fig. 30).

Although Dornier seems not to have proceeded to any aircraft projects with propulsive ducts, others were considered by Focke-Wulf, Heinkel and Messerschmitt at least[122]. These were studies for fighter-type aircraft, with maximum speeds in the high-subsonic range.

It became clear however that no supersonic aircraft existed in Germany or had been planned at the time of the report in 1943. Only in the last days of the war was a contract placed on the Siebel company at Halle to produce an aircraft based on a design study from another section of DFS. This, the 8-346, was to be air-launched and powered by Walter rocket engines. With a duration of about 2mins at full thrust, it was expected to reach 1,250mph (M = 1·9) at 66,000ft[123]. Like the M.52, it was intended to have a jettisonable pressure capsule to safeguard the pilot in an emergency, but in most other respects the 8-346 differed significantly from the Miles design. The cabin, in which the pilot would have to lie prone, was to have a rounded nose, and the relatively thick wing was of conventional section, also with a rounded leading edge, though with 45° sweepback. Pitch control was to be by conventional elevators. If the Miles team could have known about this design at some point, it seems unlikely that they would have thought that any of its features would be preferable to those that they had contrived long before. But in any case, it would not have conformed to their vision of a research aircraft that could be flown extensively, requiring it to be able to take off and land normally.

The range of testing facilities found in Germany impressed the inspection teams greatly. Several supersonic tunnels were in use, some of large size, mainly in support of the A4 (V2) ballistic missile development. But the problems of tunnel choking around M = 1 had not been resolved effectively, and there was no facility that could have provided the reliable data on drag throughout the transonic region, that had been needed to refine the estimations of performance of the M.52. However, in discussions with the staff of the Focke-Wulf company, Smelt learnt that reliable results up to sonic speed were believed to be obtainable with an open-jet wind tunnel[124], as had been pioneered before the war by Ferri[46]. Studies of the corrections for the presence of the model had shown them to be smaller for the constant-pressure boundary of the jet than for the fixed boundary of an enclosed tunnel. This suggests that, if the early work on propeller sections, and the later results of Ferri had been reconsidered in relation to the M.52 project, a further investigation in that direction could have been of great value.

The Focke-Wulf company had also received results from dropped-body tests of the type that had been tried, but had not been fruitful in Britain. Though supersonic speed had been obtained with the body alone, with test wings in place the limit had been about M = 0·92. It was estimated from the results that the value for the peak drag coefficient would be about 1·2($t/c$). Details of the wing sections tested were not given, but this value would be consistent with the results reviewed above, if they had a larger thickness ratio than that chosen for the M.52.

Both Miles and Power Jets would have been interested to learn of the multiple-shock annular intake developed for supersonic ramjet projectiles by Karl Oswatitsch and others at Göttingen, Brunswick and elsewhere[121]. A similar form had originally been intended for the nose of the M.52 (see Fig. 4), but there would have been little reason for them to regret having abandoned it, given the reported difficulties in obtaining

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Figure 31. Vickers rocket-propelled model version of the Miles M.52, showing faired-in intake and modified fin shape (FAST Museum).

good flow stability and energy recovery when this form of intake had to operate over a wide range of Mach number.

In the months before the Miles project was cancelled, a contract had been placed with Vickers to produce air-launched, rocket-propelled model aircraft, with which transonic and supersonic aerodynamics were to be explored without risk to pilots. The single successful flight that resulted was with a 3/10 scale model of the M.52, accurate except for the fairing-in of the air intake and a modification to the rear fuselage and fin, but this was not achieved until 1948 (see Fig. 31)[125]. The wing loading was comparable with that of the M.52 only in the final part of the flight, after the exhaustion of the fuel, when the aircraft was decelerating. In the report, warnings are given of potential inaccuracies in the drag results from this trial, both in the measurements themselves, and through corruption during telemetry to the ground station. The best estimate obtained for the overall drag coefficient in the supersonic region was similar to that derived in the estimation of Section 9.5 above, being 0·08 at M = 1, with a local peak value of about 0.10 at around M = 1·08. However, beyond this, the derived drag curve had begun to rise again, rather than continuing to fall as expected on theoretical grounds. Perhaps of greater significance was that the model was stable and controllable by a simple autopilot throughout the high subsonic, transonic and supersonic regions, up to a maximum Mach number of 1·38. Although this provided some validation of the aerodynamic design of the M.52 on which the model was based, it was concluded that the test method could not yield data of the reliability required for a research programme, and it was abandoned after the single successful flight.

It had been intended originally that the model trials would go on to explore the potential for drag reduction by sweepback, that had been revealed in the surveys of German work. Bancroft came to feel that the personnel at RAE had been unduly impressed by this. He retained his opinion that a sufficiently-thin straight wing had the best combination of properties for aircraft that had to proceed beyond the transonic region into fully-supersonic flight[126]. In due time, this view was put to the test when another aircraft having a thin, straight wing, of biconvex section and with sharp edges, made its appearance. This was the Lockheed F-104 Starfighter, that came into service more than a decade later[127]. With a wing of only 3·4% thickness ratio and wing loading of around 100lb/ft$^2$, it was fitted with both leading and trailing edge flaps, as had been proposed to counter the early separation and high profile drag found with the M.52 wing (see Fig. 18), though in this case blowing was used on the rear flap for greater effectiveness. Engine development had by that time enabled the transonic drag rise to be overcome, and the aircraft was flown to more than M = 2·2, including setting a world speed record of 1,404mph.

## 12.0 CONCLUSIONS

In the M.52 of 1943-46, Miles Aircraft Ltd designed the first aircraft intended to fly supersonically. At the time, relevant knowledge was fragmented, and divided between the disciplines of aerodynamics and ballistics. But brought together, and supplemented by the new theoretical work prompted by the project, it was a sufficient basis for the conception of a machine with the potential to explore the transonic and supersonic flight regimes required by its Specification.

Concerning the particular aerodynamic design, the following further conclusions are drawn:

- Miles made an astute appraisal of the available information, and conceived a forward-looking machine, that was well fitted for its intended purpose. The aerodynamic design, complemented in the structure and systems, was well supported by a considerable programme of testing, in wind tunnels, in ground test rigs and in low-speed flight at full scale. Novel approaches were adopted to problems that were being met for the first time.
- Aerodynamic data for the transonic region were sparse and uncollated, but no satisfactory method of obtaining new information was devised at the time. Despite remaining uncertainties, it was sufficiently plausible to suppose that the aircraft would be stable and controllable through this region to justify entering on a flight test programme.
- Prediction of the attainable performance was obscured by misconceptions about the basis for determining the drag of the fuselage, which caused estimates made by RAE to be too high and by Miles to be too low. An appraisal attributable to Dr J.W. Maccoll of ARD was possibly sounder, but did not appear to have been widely recognised.
- The Whittle-inspired power plant for the M.52 was as innovative a concept as the airframe. This was to enable the aircraft to take off and land conventionally, as necessary for the repeated test flying required of a research vehicle. For high-speed flight, the novel use of bypass flow with combustion would augment the thrust, to reach levels not foreseeable with any other engine development at the time.
- At the termination of the project, the most plausible estimates of available thrust and drag would indicate that sonic speed would not be exceeded in level flight, though speeds well into the supersonic range would be obtained in a dive. An opportunity to acquire a promising and most timely research tool was lost in its cancellation.

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## REFERENCES

1. WOOD, D. *Project Cancelled*, 1975, Macdonald & Janes, London.
2. The supply of military aircraft, 1955, White Paper, Cmd 9388, HMSO, London, UK.
3. KERSHAW, T. *Jet Pioneers: Gloster and the Birth of the Jet Age*, 2004, Sutton Publishing, Stroud, UK.

https://doi.org/10.1017/S0001924000003602 Published online by Cambridge University Press

4. GRIERSON, J. *Jet Flight*, 1946, Sampson Low, Marston & Co Ltd, London, UK.
5. BRINKWORTH, B.J. On the aerodynamics of the Gloster E28/39 – a historical perspective, *Aeronaut J*, 2008, **112**, pp 307-326
6. LOCK, C.N.H. Problems of high speed flight as affected by compressibility, *Aeronaut J*, 1938, **42**, pp 193-228.
7. Research on aircraft flying at supersonic speeds, 1943-1945, AVIA15/1908, continued in Athodyd propulsion for aircraft flying at supersonic speeds 15/1909, National Archives, Kew, UK.
8. HALL, A.A. and CONSTANT, H. Note on the development of the additional power from aircraft by jet reaction, May 1940, RAE Report BA1604, FAST Museum, Farnborough, UK.
9. Staff of Engine Dept. A brief discussion of the use and performance of propulsive ducts, June 1943, RAE Tech Note Eng 159, FAST Museum, Farnborough, UK.
10. RELF, E.F. Some simple considerations relating to flight at supersonic speeds, 28 May 1943, Ae2242, Aero Sub-Cttee, ARC, AIR 62/800, National Archives, Kew, UK.
11. DURAND, W.F. (Ed). *Aerodynamic Theory*, Vol III, 1935, Springer, Berlin.
12. SMELT, R., FOUGÈRE, C.M. and HOWELL, A.R. Note on supersonic flight, June 1943, RAE Tech Note Aero 1216, FAST Museum, Farnborough, UK.
13. GOLLEY, J. *Genesis of the Jet*, 1996, Airlife Publications, Marlborough, UK.
14. BROWN, D.L. *Miles Aircraft since 1925*, 1970, Putnam, London, UK.
15. TEMPLE, J.C. *Wings over Woodley*, 1987, Aston Publications.
16. SERBY, J.E., MORGAN, M.B. and COOPER, E.R. Flight test on the profile drag of 14% and 25% thick wings, 1937, ARC R&M 1826.
17. Staff of Aerodynamics Dept, RAE. Flight and model rests of the 'Hawcon' with wings 30% thick at the root, 1939, ARC R&M 1937.
18. COHEN, J. and S/L FRASER, H.P. Flight tests of a Falcon fitted with an Irving flap, 1938, ARC R&M 1863.
19. HILLS, R., BROWN, E.C., MORRISON, M.A. and BECKER, H.V. Wind-tunnel test on the experimental high lift M.18, 1942, ARC R&M 2453.
20. BROWN, D.L. Visit to Messrs Power Jets Ltd, Rugby, 5 November, 1942 and Memorandum on visit to Power Jets Ltd, Rugby, 16 June 1943, PRATT, J., private communication.
21. Miscellaneous papers relating to visits of and telephone calls with Miles Aircraft Ltd (formerly Phillips and Powis), September/October 1943, Power Jets Ltd, AIR 62/804, National Archives, Kew, UK.
22. BANCROFT, D.S. The Miles M.52 supersonic aircraft, *Space Chronicle*, *J Brit Interplanetary Soc*, 2002, **55**, Suppl 2, pp 77 -85
23. WHITTLE, F. *Jet – the Story of a Pioneer*, 1953, Frederick Mueller, London, UK.
24. Miles E24-43 with large wing, January 1944. AIR 30/1628, National Archives, Kew, UK.
25. Some notes on the firm's activities during the war, October 1947, for inclusion in MAP Scientific War Records, Miles Aircraft Ltd, AVIA 44/603, National Archives, Kew, UK.
26. COWLEY, W.L. 1927 Schneider Trophy Contest (Intro), January 1931, Collected Reports on British High Speed Aircraft. ARC R&M 1300.
27. DIEHL, W.S. *Engineering Aerodynamics*, Revised edition, 1936, The Ronald Press Co, New York, USA.
28. WARNER, E.P. *Airplane Design – Performance*, Second edition, 1936, McGraw-Hill, London, UK.
29. WILKINS, E.W.C. *Airplane Design*, 1938, Charles Griffin, London, UK.
30. PIERCY, N.A.V. *Aerodynamics*, 1937, EUP Ltd, London, UK.
31. ACKROYD, J.A.D. The United Kingdom's contributions to the development of aeronautics, Part 4. The origins of the jet age, *Aeronaut J*, 2003, **107**, pp 1-47.
32. REED, S.A. Air reactions to objects moving at rates above the velocity of sound, with applications to the air propeller, 1922, NACA Tech Memo 168.
33. BRIGGS, L.J. and DRYDEN, H.L. Aerodynamic characteristics of twenty-four airfoils at high speed, 1929, NACA Rep 319.
34. BRIGGS, L.J. and DRYDEN, H.L. Aerodynamic characteristics of circular-arc airfoils at high speed, 1930, NACA Rep 365.
35. DOUGLAS, G.P. and PERRING, W.G.A. Wind tunnel tests with high tip speed airscrews, The characteristics of the aerofoil RAF 31a at high speeds, 1927, ARC R&M 1086.
36. DOUGLAS, G.P. and PERRING, W.G.A. Wind tunnel tests with high tip speed airscrews, The characteristics of a bi-convex aerofoil at high speeds, 1927, ARC R&M 1091.
37 DOUGLAS, G.P. and PERRING, W.G.A. Wind tunnel tests with high tip speed airscrews, The characteristics of bi-convex No2 aerofoil at high speeds, 1927, ARC R&M 1123.
38. DOUGLAS, G.P. and PERRING, W.G.A. Wind tunnel tests with high tip speed airscrews, The characteristics of a conventional airscrew section, aerofoil R&M 322 No3 at high speeds, 1927, ARC R&M 1124.
39. PERRING, W.G.A. Wind tunnel tests with high tip speed airscrews, The characteristics of airscrew section R&M 322 No4 at high speeds, 1927, ARC R&M 1134.
40. DOUGLAS, G.P. and PERRING, W.G.A. Wind tunnel tests with high tip speed airscrews, Some experiments upon an airscrew of conventional blade section, aerofoil R&M 322 No. 3 at high speed, 1928, ARC R&M 1174.
41. DOUGLAS, G.P. and PERRING, W.G.A. Wind tunnel tests with high tip speed airscrews, Some experiments upon an airscrew of conventional blade section, 0·082c thick, and of RAF 27 and RAF 28 sections, 1928, ARC R&M 1198.
42. GLAUERT, H. The effect of compressibility on the lift of an aerofoil, 1927, ARC R&M 1135.
43. ANDERSON, J.D. *A History of Aerodynamics and its Impact on Flying Machines*, 1998, Cambridge University Press, Cambridge, UK.
44. THOM, A. Blockage corrections and choking in a high-speed tunnel, 1943, ARC R&M 2033.
45. MAIR, W.A. (Ed). Research on high speed aerodynamics at the Royal Aircraft Establishment from 1942 to 1945, 1950, ARC R&M 2222.
46. FERRI, A. (trans VANIER, J.). Investigations and experiments in the Guidonia supersonic wind tunnel, July 1939, NACA Tech Memo 901.
47. SMELT, R. Proposed programme of work on drag of aerofoils near the speed of sound, by measurement on falling bodies, March 1944, RAE Tech Note Aero 1410, FAST Museum, Farnborough, UK.
48. HALLION, R. *Supersonic Flight*, 1972, Collier-Macmillan Ltd, London, UK.
49. Tallboy: ballistic trials, May 1944, Orfordness Research Station Rep BT61, AVIA6/13312, National Archives, Kew, UK.
50. HOWARTH, L. *Modern Developments in Fluid Dynamics*, Vol 2, *High Speed Flow*, 1953, Clarendon Press, Oxford, UK.
51. HILTON, W.F. British aeronautical research facilities, *Aeronaut J*, 1960, Centenary edition, **70**, pp 103-107.
52. PRITCHARD, J.L. The dawn of aerodynamics, *Aeronaut J*, 1957, **61**, pp 149-180.
53. HUNT, F.R.W. The reaction of the air to artillery projectiles, *The Mechanical Properties of Fluids*, 2nd Edition, 1936, Blackie & Son, London.
54. STANTON, T.E. The development of a high speed wind channel for research in external ballistics, 1931, Proc Royal Society London, Ser A, **131**, pp 122-132.
55. TAYLOR, G.I. and MACCOLL, J.W. The air pressure on a cone moving at high speed, 1933, Proc Royal Society London, Ser A, **139**, pp 278-311.
56. MACCOLL, J.W. The conical shock wave formed by a cone moving at high speed, 1937, Proc Royal Society London, Ser A, **143**, pp 459-472.
57. VON KÁRMÁN, T.H. and MOORE, N.B. Resistance of slender bodies moving with supersonic velocities, with special reference to projectiles, Trans ASME, *J Appl Mechs*, 1932, **54**, pp 303-310.
58. TSIEN, H.S. Supersonic flow over an inclined body of revolution, *Aeronaut J*, 1938, **5**, pp 480-483.
59. Ballistic Tables 1940, Cmd 726, War Office, UK, WO279/166, National Archives, Kew, UK.
60. LIGHTHILL, M.J. Supersonic flow past bodies of revolution, 1945, ARC R&M 2003.
61. NELSON, W.C. (Ed), *Selected Topics on Ballistics*, 1959, Pergamon Press for AGARD.
62. STANTON, T.E. A high speed wind channel for tests on aerofoils, 1928, ARC R&M 1130.
63. TAYLOR, G.I. Applications to aeronautics of Ackeret's Theory of aerofoils moving at speeds greater than that of sound, 1932, ARC R&M 1487.
64. HOOKER, S.G. The pressure distribution and forces on thin aerofoil sections having sharp leading edges, and moving with speeds greater than that of sound, 1936, ARC R&M 1721.
65. RELF, E.F. Some simple considerations relating to flight at supersonic speeds, ARC Current Paper Ae2242, AIR62/801, National Archives, Kew, UK.
66. FERRI, A. Experimental research in the Guidonia Supersonic Wind Tunnel, *Aeronaut J*, 1939, **43**, pp 803-832.
67. FERRI, A. The Guidonia High Speed Tunnel, A description of the Italian wind tunnel built for super-sonic research. *Aircraft Engineering*, 1940, **12**, pp 302-305.
68. FERRI, A. (trans FLINT, M). Some experimental results obtained on wing profiles in the Guidonia Supersonic Wind Tunnel, November 1940, Current Paper FM476, ARC (MAP Trans 1115), DSIR 23/10617, National Archives, Kew, UK.
69. CHESHIRE, L.J. Note on visit to Dr J.W. Maccoll, Armament Research Dept, Sevenoaks, 30 December 1943, AIR62/815, National Archives, Kew, UK.
70. M.52 Construction, The Miles Aircraft Collection, undated, AMOS, P., private communication.


https://doi.org/10.1017/S0001924000003602 Published online by Cambridge University Press

71. DAVIDSON, I.M. and UMNEY, L.E. On the preliminary tests of an annular intake at supersonic speed, 1943, NGTE Memo M4, AVIA28/13198, National Archives, Kew, UK.
72. Miles Aircraft Company – M-52 Project, July 1946, US Military Attache London Report No R-2420-46, PRATT, J., private communication.
73. BANCROFT, D.S. The Miles E.24/43 cabin stability and velocity distribution above pilot's emergency exit, Miles Aircraft Ltd report WT52/4, undated, BANCROFT, D.S., private communication.
74. MARSHALL, W.S.D. Model tests in the Blower Tunnel on the emergency escape of a pilot from a high speed aircraft (Miles E.24/43) and hood jettisoning tests, November 1945, RAE Tech Note Aero 1727, FAST Museum, Farnborough, UK.
75. BANCROFT, D.S. The Miles E. 24/43 take-off and landing performance, Miles Aircraft Ltd Report PE.52/2, undated, BANCROFT, D.S., private communication.
76. BANCROFT, D.S. The Miles E.24/43, September 1945, Miles Aircraft Ltd report 52/0, BANCROFT, D.S., private communication.
77. STURTIVANT, R. *British Research and Development Aircraft*, 1990, Foulis/Haynes Publishing Group.
78. The Miles Supersonic, *The Aeroplane* 13 September 1946, **71**, (1843), pp 295-296.
79. BROWN, E. Miles M.52 – The supersonic dream, *Air Enthusiast*, August/November 1980, pp 35-42.
80. BANCROFT, D.S. Wind tunnel tests on a biconvex wing with 25% plain flap, Miles Aircraft Ltd report WT52/3, undated, BANCROFT, D.S., private communication.
81. LEVACIC, I., MARSHALL, W.S.D. and YOUNG, A.D. No.2 11 1/2 foot wind tunnel tests on a ¼ scale model wing of the Miles E 24/43 aircraft, June 1945, RAE Report Aero 2044, FAST Museum, Farnborough, UK.
82. MARSHALL, W.S.D., LEVACIC, I. and YOUNG, A.D. No.2 11 1/2 foot wind tunnel tests of a high lift supersonic wing, May 1945, RAE Report Aero 2045, FAST Museum, Farnborough, UK.
83. SCHMITZ, F.W. (trans FLINT, M.). Aerodynamik des Flugmodels, 1942, Air Min RTP Trans 2460.
84. BANCROFT, D.S. Flight tests of the Falcon with biconvex wings, Part I, Miles Aircraft Ltd report M.A.3/1, undated, BANCROFT, D.S., private communication.
85. BANCROFT, D.S. E.24/43. Clothed wing, (1), Miles Aircraft Ltd report WT.52/6 undated, BANCROFT, D.S., private communication.
86. MARSHALL, W.S.D, LEVACIC, I., YOUNG, A.D. and POWTER, G.J. Low-speed wind tunnel tests of a high lift supersonic wing, 1946, ARC R&M2365 (comprising RAE Reports Aero 2045, May 1945, Aero 2157, Dec 1946 and Addendum to Aero 2157, March 1947).
87. WINTER, K.G. and MORGAN, M.B. Wind tunnel tests on the Miles E24/43, July 1944, Part I, RAE Report Aero 1959, FAST Museum, Farnborough, UK.
88. HUTTON, S.P. and GAMBLE, H.E. High speed wind tunnel tests on an aircraft designed for supersonic speeds (Miles E24/43), July 1945, RAE Report Aero 2057, FAST Museum, Farnborough, UK.
89. BANCROFT, D.S. Preliminary wind tunnel tests on the Miles E.24/43, August 1945, Miles Aircraft Ltd Report WT.52/1, BANCROFT, D.S. private communication.
90. HILTON, W.F. and PRUDEN, F.W. Subsonic and supersonic high speed tunnel tests of a faired double wedge aerofoil, 1943, ARC R&M 2057.
91. LIGHTHILL, M.J. The conditions behind the trailing edge of the supersonic aerofoil, 1944, ARC R&M 1930.
92. LIGHTHILL, M.J. The supersonic theory of wings of finite span, 1944, ARC R&M 2001.
93. BANCROFT, D.S. Wind tunnel rests of a biconvex tailplane and elevator, Miles Aircraft Ltd Report WT.52/2, undated, BANCROFT, D.S., private communication.
94. BANCROFT, D.S. Flight tests of the Falcon with biconvex wings, Part II, Biconvex tailplane and elevator, Miles Aircraft Ltd Report M.A.3/1a, undated, BANCROFT, D.S., private communication.
95. BANCROFT, D.S. Flight tests of the Falcon with biconvex wings, Part III, (all-moving biconvex tailplane), Miles Aircraft Ltd Report M.A.3/1b, undated, BANCROFT, D.S., private communication.
96. KLECKNER, H.F. Preliminary flight research on an all-movable horizontal tail as a longitudinal control for flight at high Mach numbers, March 1945, NACA Report L5CO8.
97. MITCHELL, K., THORPE, A.W. and FRAYN, E.M. A theoretical investigation of the response of a high-speed aeroplane to the application of ailerons and rudder, May 1945, ARC R&M 2294.
98. COLLAR, A.R. Theoretical forces and moments on a thin aerofoil with hinged flap at supersonic speeds, 1943, ARC R&M 2004.
99. MCDONNELL, P. Beaten to the barrier, *Aero Monthly*, January 1998, pp 20-25.
100. SMELT, R. Note on longitudinal stability and trim changes at speeds near the speed of sound, February 1944, RAE Report Aero 1911, FAST Museum, Farnborough, UK.
101. BURNET, C. *Three Centuries to Concorde*, 1979, Mechanical Engineering Publications Ltd, London, UK.
102. COLLAR, A.R. Resistance derivatives of flutter theory, Part II: Results for supersonic speeds, 1944, ARC R&M 2139.
103. TEMPLE G. and JAHN, H.A. Flutter at supersonic speeds : derivative coefficients for a thin aerofoil at zero incidence, 1945, ARC R&M2140.
104. AIRCRAFT: Design and Development: Bombers (Code 6/2), Miles fighter-bomber jet propulsion aircraft to specification E.24/43: development policy, AVIA15/1968, National Archives, Kew, UK.
105. BANCROFT, D.S. Definitive list of M.52 models to various scales, March 2000, AMOS, P., private communication.
106. FOUGÈRE, C.M. and SMELT, R. Note on the Miles supersonic aircraft (preliminary version), December 1943, RAE Tech Note Aero1347, FAST Museum, Farnborough, UK.
107. HODGE, J. Comments on Minutes of 8th Meeting of Supersonic Flight Committee, 14 December 1943, January 1944, AIR62/813, National Archives, Kew, UK.
108. FOUGÈRE, C.M. Further note on the Miles supersonic aircraft, July 1944, RAE Tech Note Aero 1470, FAST Museum, Farnborough, UK.
109. BANCROFT, D.S. The Miles E.24/43, fitted with W2/700 and No.4 Augmenter, Estimated performance, March 1946, Miles Aircraft Ltd, report PE.52/1, BANCROFT, D.S., private communication.
110. SMELT, R. Experimental supersonic aircraft employing rocket propulsion, February 1945, RAE Report Aero 2016, FAST Museum, Farnborough, UK.
111. NONWEILER, T. The design of a supersonic aircraft utilising the German HWK-509 A-1 bi-fuel rocket unit, June 1945, RAE Tech Note Aero 1655, FAST Museum, Farnborough, UK.
112. BROTHERTON, J. Further consideration of an experimental supersonic aircraft, June 1945, RAE Tech Note Aero 1648, FAST Museum, Farnborough, UK.
113. HILTON, W. Subsonic and supersonic tests on a 7½% bi-convex aerofoil, May 1944, ARC R&M 2196.
114. THOM, A.W., SMITH, F. and BROTHERTON, J. Flight tests at high Mach number on E28/39 W4041 (single-engined jet-propelled aircraft), October 1945, ARC R&M 2264.
115. CHESHIRE, L.J. Visit of Dr Maccoll, Dr Hayworth, and Mr Illingworth of the Armament Research Depot; and Mr Capley and Mr Bancroft of Miles Aircraft Ltd, 19 June 1944, 16-6-44, Power Jets Ltd, AIR62/ 821, National Archives, Kew, UK.
116. Some notes on compressibility effects in relation to the design of high speed aircraft, 9 February 1945, Miles Aircraft Ltd, AVIA15/2233, National Archives, Kew, UK.
117. MOSS, G.M., LEEMING, D.W. and FARRAR, C.L. *Military Ballistics*, 1995, Brassey's, London, UK.
118. PLACE, E.L. and SMITH, A.G. Note on the estimation of head drag of nacelles at supersonic velocities, 28 June 1944, Aerodynamics Test Report No. 106, Power Jets Ltd, AIR62/823 National Archives, Kew, UK.
119. HODGE, J. Estimated performance of W2/700 with No.4 design augmenter in M.52, January 1944, Report A.122, Power Jets Ltd, AVIA28/2970, National Archives, Kew, UK.
120. MILLIKAN, C.B. Draft Technical Report of Visit, British M.A.P. Project E24/43 Miles 52 Transonic Research Airplane, June 1946, Clark Millikan papers, Box 6, File 6.2, NavTechMisEu, California Institute of Technology.PRATT, J., private communication.
121. BENECKE, T. and QUICK, A.W. (Ed). History of German guided missiles development, *AGARDograph*, 1957, **20**, Verlag E. Appelhans, Brunswick, Germany.
122. NONWEILER, T. German high-speed aircraft and guided missiles, Part 1 – aircraft, August 1945, RAE Report Aero 2070, AVIA 6/9900, National Archives, Kew, UK.
123. The 8-346 supersonic research aircraft, A.I.2(g) Report 2382, AIR 40/142, National Archives, Kew, UK.
124. SMELT, R. Notes on discussion with the staff of Focke-Wulf Ltd, May 1945, RAE Tech Note Aero 1644, FAST Museum, Farnborough, UK.
125. Staff of Supersonics Division, Flight Section, Flight trials of a rocket-propelled transonic research model: the RAE-Vickers Rocket Model, March 1950, ARC R&M 2835.
126. BANCROFT, D.S. Note on the practicability of wing section and plan forms etc, for various flight speeds, Miles Aircraft Ltd Report no 52/1, undated, BANCROFT, D.S., private communication.
127. BOWMAN, M.W. *The Lockheed F-104 Starfighter*, 2000, Crowood Press Ltd, Marlborough, UK.


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