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EFFICIENCY OF A TOP-MOUNTED INLET SYSTEM AT TRANSONIC/ SUPERSONIC SPEEDS

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Abstract

At transonic and supersonic speeds an experimental investigation has been carried out to increase the knowledge of the combined effects of the favourable wing flow and the disturbances from the forebody and especially the canopy on the inlet performance of an aircraft concept characterized by its inlet position on top of the fuselage. Wind tunnel tests were performed with a 1:23.5 model in one of FFA's transonic/supersonic wind tunnels. At the engine face station steady state pressure measurements were carried out to establish the inlet performance and the canopy was equipped with pressure taps giving the static pressure distribution. Results are presented which show that in many respects the inlet performances are very good but they also indicate some problem areas caused by the complex flow in front of the inlet.

Introduction

An extensive project work was carried out in Sweden in the late 70's concerning a subsonic aircraft with combined attack and trainer capacity. This work covered a number of different configurations of which some to various degrees were unconventional, for example, characterized by the position of the inlet on top of the fuselage. For one of these configurations a wind tunnel investigation was carried out with the objective of finding the inlet flow quality at take off and and landing conditions. The results showed that a top-mounted inlet configuration could be designed to give good inlet performance at low-speed but also, especially by flow studies, that the design of the forebody is very important and must be made with great care to match the chosen inlet position. (The results were presented at ICAS 12th Congress, 1980.)

Later, interest in a faster aircraft became greater and the project studies were now directed towards a light-weight fighter. Again a number of configurations were considered of which several were equipped with top-mounted inlet. One of these concepts is shown in Figure 1.

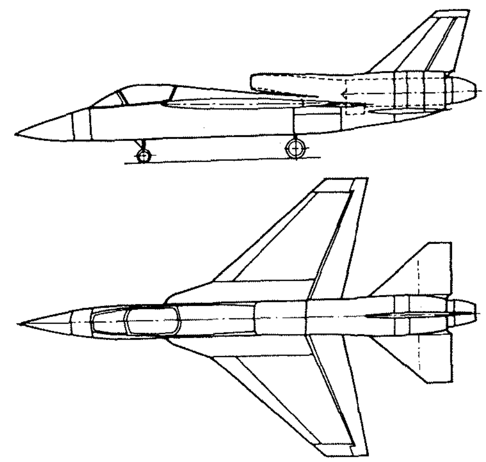

Figure 1. A light-weight fighter concept with top-mounted inlet.

Compared to more conventional engine inlet locations mounting the inlet above the fuselage gives a variety of potential advantages including: Reduced aircraft structural weight due to a single, short duct; Eliminating problems with foreign object damage during take off and landing; Reudced radar cross section area due to shielding the inlet system from especially low-altitude and ground-based radar; Eliminating engine-inlet compatibility problems during weapons delivery.

From the aerodynamic point of view the basic idea is to reduce the effect of angle of attack by placing the inlet in the flow over the wing's upper surface where favourable effects could be expected, at least as long as the wing flow is not largely separated. On the other hand, the flow over the forebody, canopy, strakes, etc., is of course not at all independent of angle of attack (or sideslip) and the primary aim with this investigation is, therefore, to increase the knowledge of the combined effects of the favourable wing flow and the disturbances from the forebody on inlet performance.

The wind tunnel program carried out with a model based roughly on the above concept covered a Mach number range from 0.5 to 1.52 with various angles of attack and sideslip.

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Model Description

With reference to the aircraft concept shown in Figure 1 a scale 1:23.5 model was built and the external contours of the model including its sting support are shown in Figure 2. The aft part of the fuselage with stabilizer etc. was considered unimportant for inlet performance and was therefore excluded. The 50° swept wing has a symmetric profile with a thickness of 6%. It is equipped with movable leading and trailing edge flaps and with slightly drooped strakes in front of the wing. To insure turbulent boundary layer on the forebody transition strips were fixed to the models nose.

The strakes are intended to prevent part of the fuselage side boundary layer reaching the area in front of the inlet. The vortices formed by the strakes remove this boundary layer flow. They are also intended to suck in some of the low energy air from the flow over the canopy and from the area at the top of the fuselage just in front of the inlet. The vortices pass outside the inlet and the geometry must be carefully chosen to prevent ingestion of a vortex into the inlet also at large angles of side-slip. Earlier low speed tests have shown that strakes are essential for top-mounted inlet configurations of this type.

The height of the canopy and the slope of the surface just ahead of the inlet is kept low to avoid too much overexpansion at high subsonic cruise Mach number. This area will then be supersonic and the intention is to minimize the risk of shock-induced boundary layer separation. On the other hand it is of course very important to try to avoid screening the pilot's visibility.

The inlet/duct arrangement is shown in Figure 3. A vertical wall located in the middle of the inlet is integrated in the duct design. The purpose is to facilitate the change of flow direction and to reduce the aerodynamic loading of the windward side of the inlet. This central wall is also intended somewhat to attenuate the unavoidable disturbances in the flow originating from the forebody.

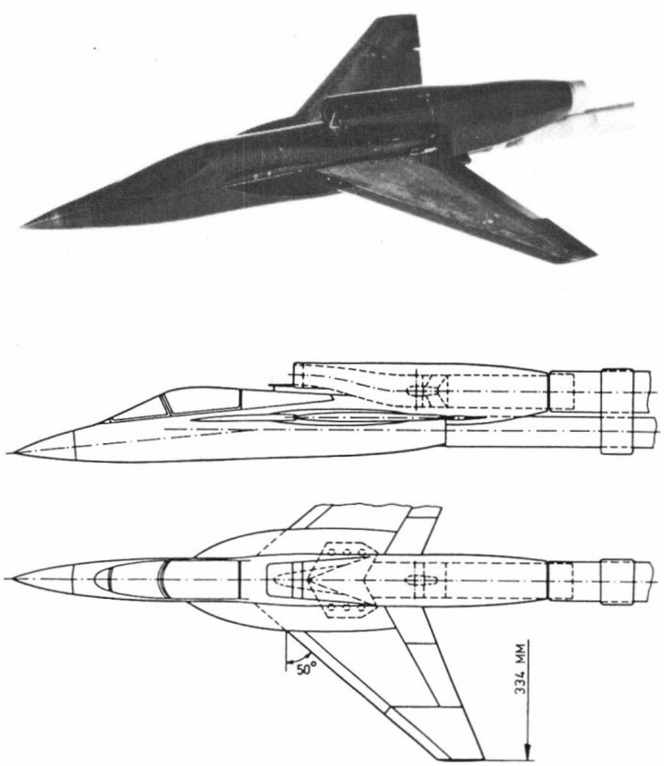

Figure 2. The Wind Tunnel Model. Scale 1:23.5.

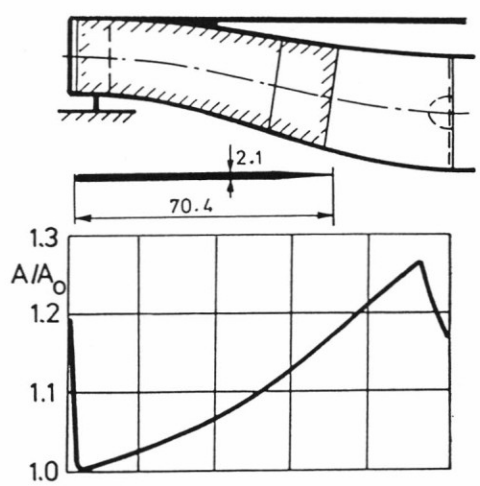

Figure 3. The duct with the central wall. Area distribution is without the effects of the wall.

From an aerodynamic point of view the central wall is probably unnecessarily long. The reason for the chosen length is to keep the possibility open for designing the wall of radar-absorbent material and as the duct often causes a large part of the radar cross-section area considerable reduction of this area may be achieved. It is also probable that the structural design of the duct could take advantage of this, giving weight reduction.

The duct is S-shaped, has constant width and sections built up of straight lines and rounded corners. The corner radius is continuously increasing towards the simulated engine face. The entire available length has been used to form an S shape with as little curvature as possible. Figure 3 also gives the general arrangement and the area distribution.

The inlet, Figure 4, is of a quite ordinary pitot-type. It is designed firstly for good manoeuvre performance at relative low speed and since only moderate supersonic speeds are of interest a multi-shock inlet system has been excluded. The lips are therefore rather thick. Both the upper and lower lips are slightly drooped giving a continuous twist along the side walls.

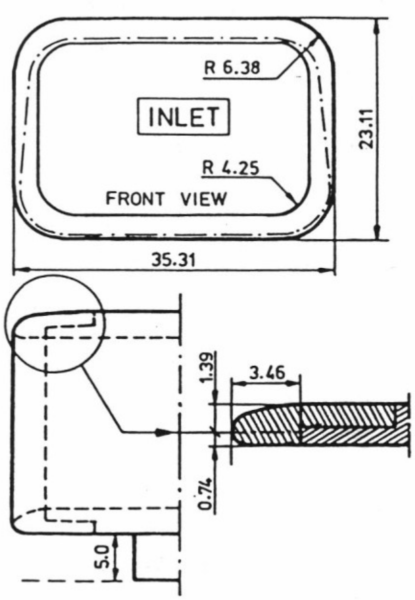

Figure 4. The geometry of the inlet.

Model Instrumentation

The model was intrumented to obtain steady state pressures at the simulated engine face station. Despite the importance of knowing the turbulence levels in this station no high-response transducers were introduced. In this early stage of investigation it was considered good enough to get a steady state survey and it should be both costly and difficult to equip the model with a number of high-response pressure transducers. The six-armed rake arrangement is shown in Figure 5 and the position of the probes in Figure 6. Each arm has five probes positioned in the centre of equal, annular, areas.

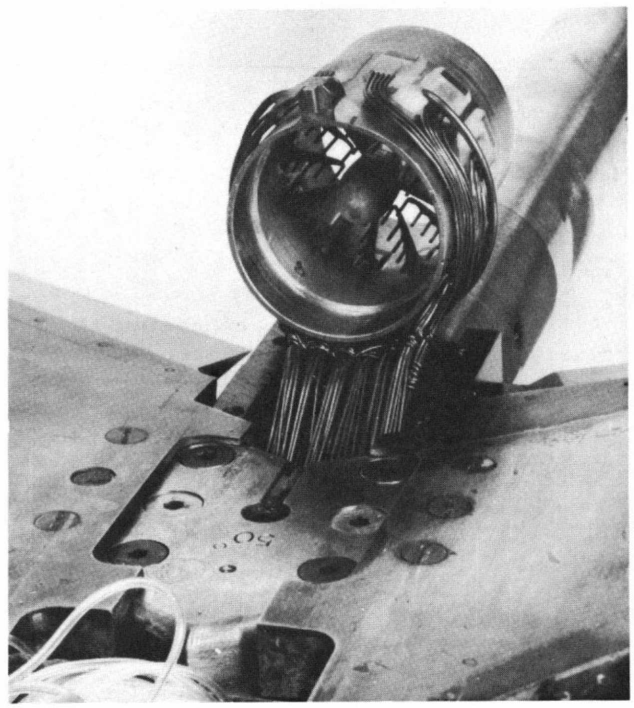

Figure 5. The six-armed rake arrangement.

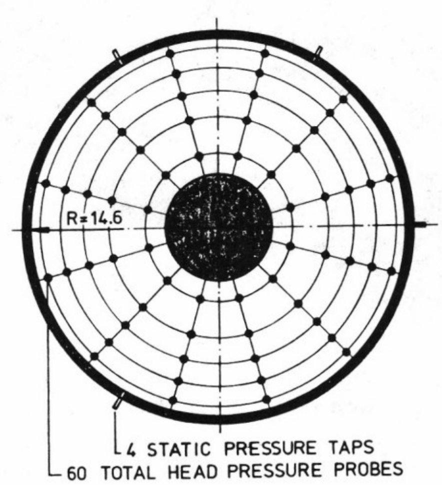

Figure 6. Engine compressor-face instrumentation.

This station was also equipped with four static pressure taps used for calculating average dynamic pressure and face Mach number. The flow condition ahead of the inlet is of course with this kind of layout of most interest, which is, why the canopy was equipped with nine static pressure taps, according to Figure 7.

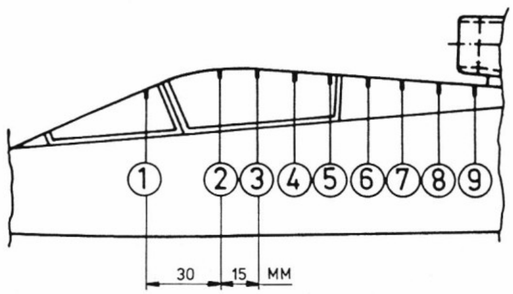

Figure 7. Instrumentation for canopy static pressures.

All pressure tubes were connected to an ordinary computer-controlled scanivalve system. The signals were filtered through a 18 dB/octave LP filter with an upper cut off frequency of 7 Hz prior to sampling. The two scanivalves, Figure 8, located under the removable canopy, were by consideration of space closely packed and driven by a single step-motor.

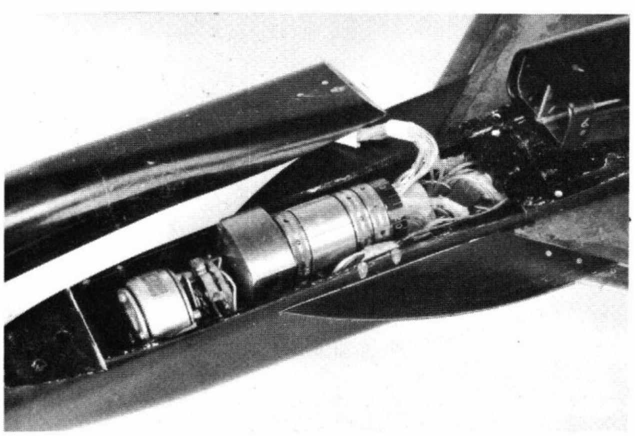

Figure 8. The two parallel coupled scanivalve installation.

Test Facility

The model was tested in one of the FFA's transonic-supersonic wind tunnels, S4. The tunnel is of suck-down type, sucking atmospheric air via a driver into a vacuum storage. The size of the transonic test section is 0.92×0.90 m and that of the supersonic is 0.92×1.15 m. To generate the air flow through the inlet the model is connected by a pipe-line to the mentioned vacuum storage. The pipe was equipped with a nozzle system to measure the inlet mass flow.

Inlet Test Program

Testing in the transonic test section was conducted at Mach numbers over the range 0.50 to 1.18. Maximum angle of attack was 32^{o} (at M=0.5) and the sideslip angles were 0, 6 and 10 degrees. The supersonic tests was conducted with the M=1.52 nozzle. The angle of attack range was 0 to 12^{o} and the sideslip range was 0 to 6^{o}. The Reynolds number varies from 11×10^{6}/m at M=0.5 to 27×10^{6}/m at M=1.52.

At both the transonic and supersonic tests the inlet air flow was varied. Most of the results presented are for the maximum air flow condtion. The conditions at the simulated engine face are in general described by total pressure losses (f) and steady state distortion indices (DC_{60} and IDCL).

Subsonic-Transonic Inlet Test Results

The subsonic and transonic inlet performance at 0^{o} sideslip and 100% air flow at M=0.5, 0.8 and 1.18 is presented in Figure 9 in terms of total pressure losses and distortion indices. For both M=0.5 and 0.8 the losses are practically unaffected by angle of attack. This good quality corresponds well with the experience from earlier mentioned tests at take off and landing speeds (M=0.18). The pressure loss level, 2.5%, is high due to the relatively large central wall in the duct, which gives an estimated increase of close to one percent unit. The distortion indices should in fact be compared to maximum allowable values for a typical low-bypass fighter engine. These values have not been quoted due to their proprietary nature but the indices obtained at M=0.5 and 0.8 are considered low.

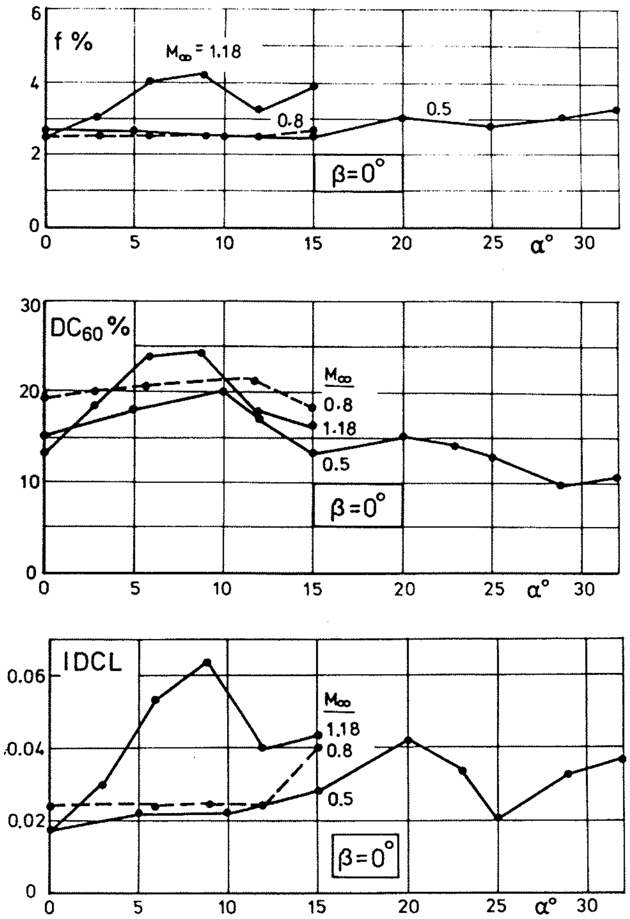

Figure 9. Total pressure losses and distortion at $\beta=0^{\circ}$ and 100% mass flow.

The canopy pressure distribution, Figure 10 show an almost continuously decreasing pressure level in front of the inlet with very small changes concerning the pressure gradients. The figures also included the local Mach number, M_L, which is based on local static pressure and freestream total pressure. At M=0.8 the maximum local Mach number is close to unity.

At M=1.18 a moderate incease of angle of attack gives a marked performance deterioration; both losses and distortion increases. Figure 10 shows that the Mach number at the top of the canopy is about 1.5 and the schlieren picture taken at α=3 deg., Figure 11, shows a rather complicated flow with a severe boundary layer separation in front of the inlet.

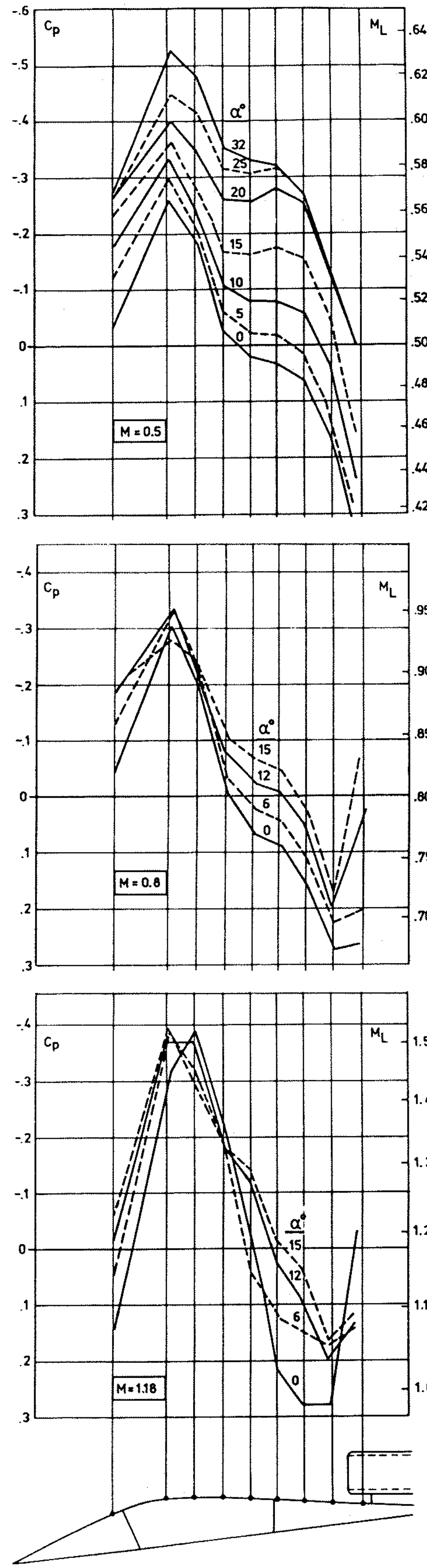

Figure 10. Canopy pressure distribution at $\beta=0^{\circ}$ and 100% mass flow.

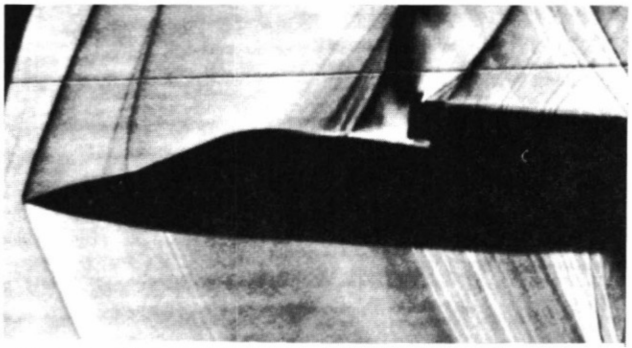

Figure 11. Schlieren picture at M=1.18, $\alpha=3^{\circ}$, $\beta=0^{\circ}$ and mass flow 100%.

The effects of slideslip are shown for M=0.5 and 0.8 and at α=15 deg. in Figure 12. Figure 13 shows that the flow in front of the inlet is very sensitive to angles of sideslip. Downstreams the expected suction peak at the top of the canopy another sideslip dependent suction peak develops that increases the adverse pressure gradient in front of the inlet.

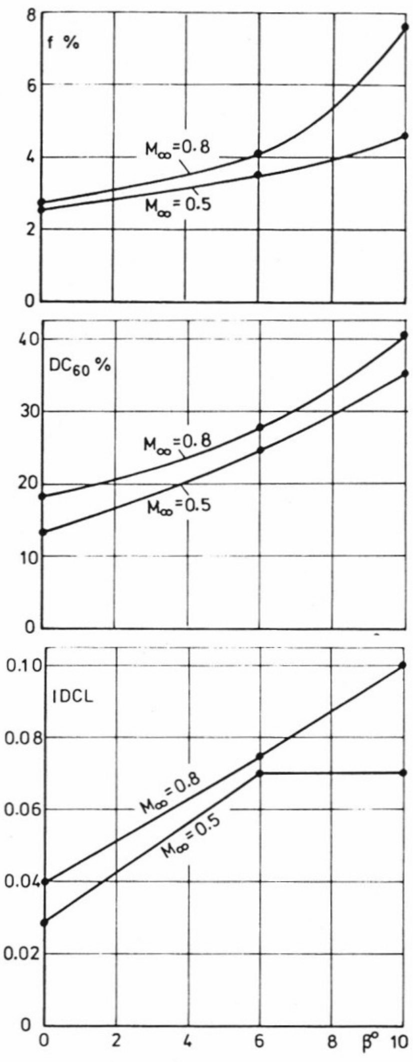

Figure 12. Effecs of sideslip at $\alpha=15^{\circ}$.

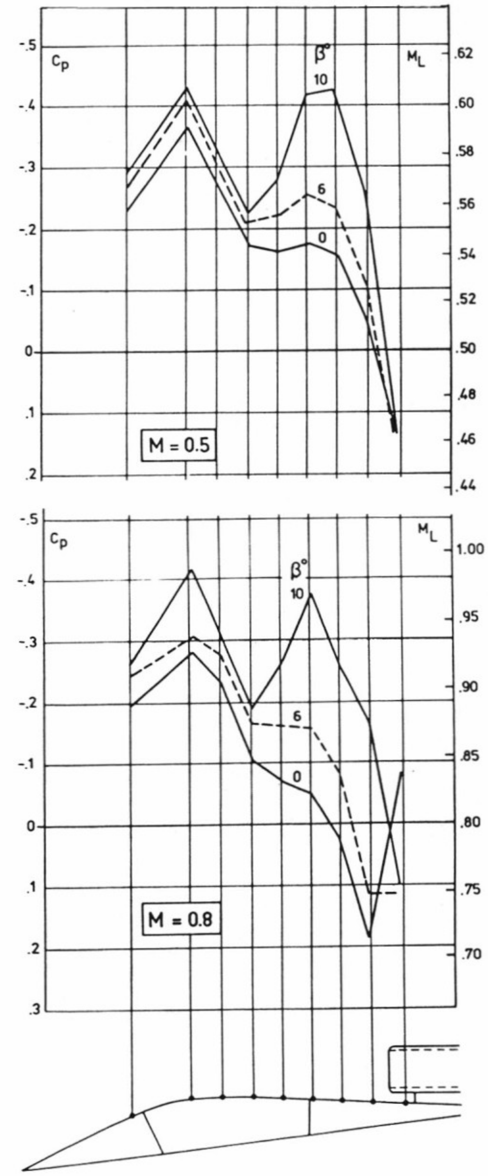

Figure 13. Effects of sideslip on canopy pressure distribution at $\alpha=15^{\circ}$.

The results discussed up to now have all been with maximum air flow through the inlet, but some tests were also done with reduced air flow. As shown in Figure 14 reducing air flow at M=0.5 gives no problems but at M=0.8 and 1.18 there are marked limits below which the flow separates at the top of the canopy. The schlieren picture, Figure 15, shows this separated flow conditions at M=1.18 and α=3 deg. At inlet air flows close to the limits the flow over the canopy is unstable and a low frequency pumping situation easily arizes. When pumping the flow is gradually changed between the two flow conditions shown in Figures 11 and 15. The effects of inlet air flow reduction on canopy pressure distribution are summarized in Figure 16.

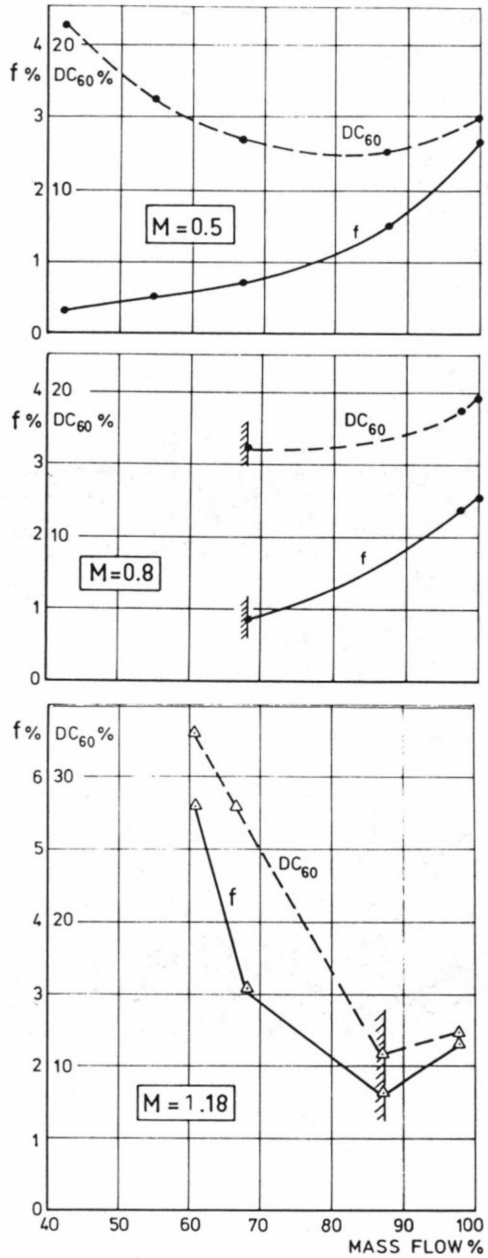

Figure 14. Inlet performance at reduced mass flow. $\alpha=0^{\circ}$, $\beta=0^{\circ}$.

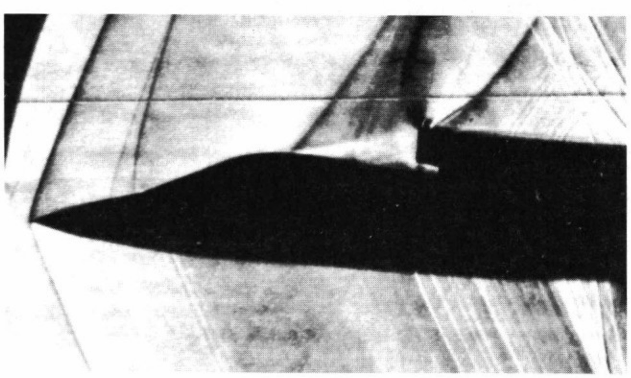

Figure 15. Mass flow reduced to about 70% at M=1.18, $\alpha=3^{\circ}$ and $\beta=0^{\circ}$.

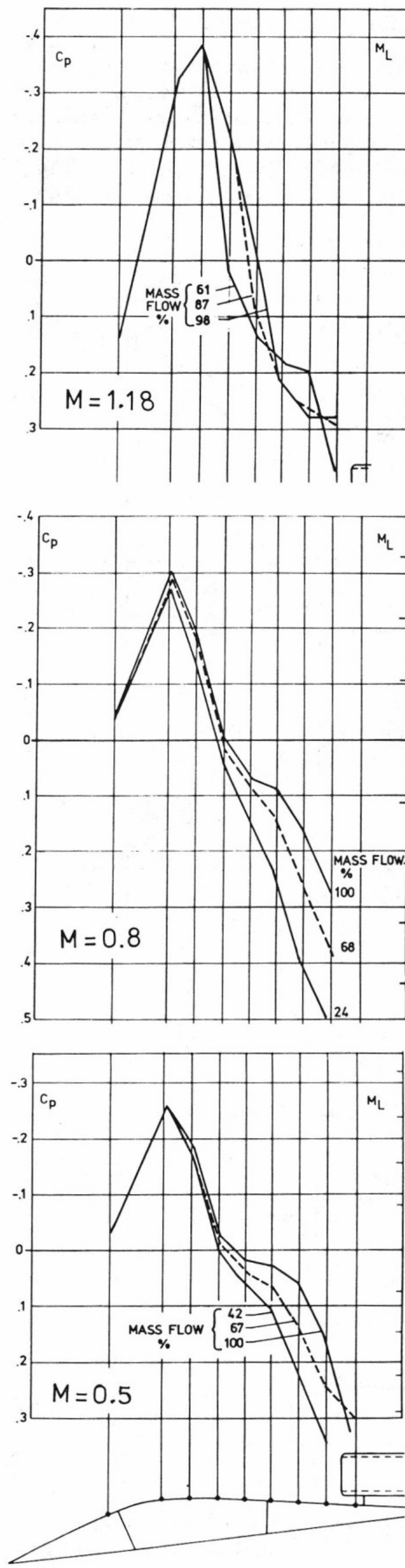

Figure 16. Canopy pressure distribution at $\alpha=0^{\circ}$, $\beta=0^{\circ}$ and reduced mass flow.

Supersonic Inlet Test Results

The inlet performance at M=1.52 and maximum inlet airflow is shown in Figure 17. At β=0 deg. the pressure losses increase with angle of attack while the distortion index IDCL is practically unaffected and the DC_{60} shows a high value at $\alpha=6^{\circ}$.

The series of schlieren picture, Figure 18, also show only minor changes in flow conditions when increasing angle of attack from 0 to 12 degrees. In front of the normal shock there is a low separated region and it can be seen that it grows slightly with angle of attack but also that most of the separated air passes below the inlet.

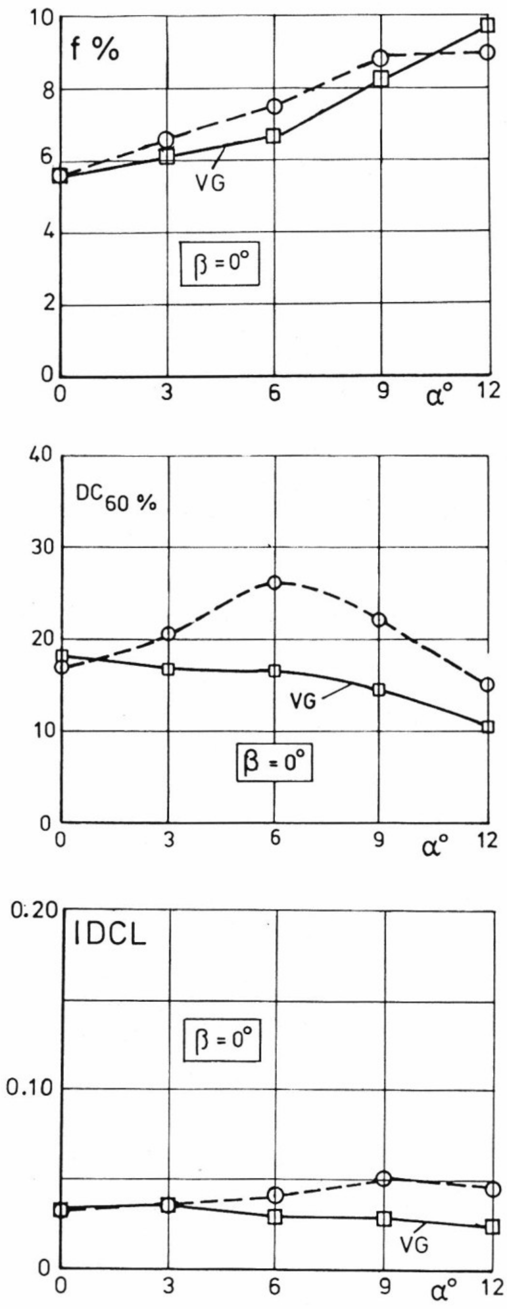

Figure 17. Inlet performance at M=1.52, β= 0° and 100% mass flow.

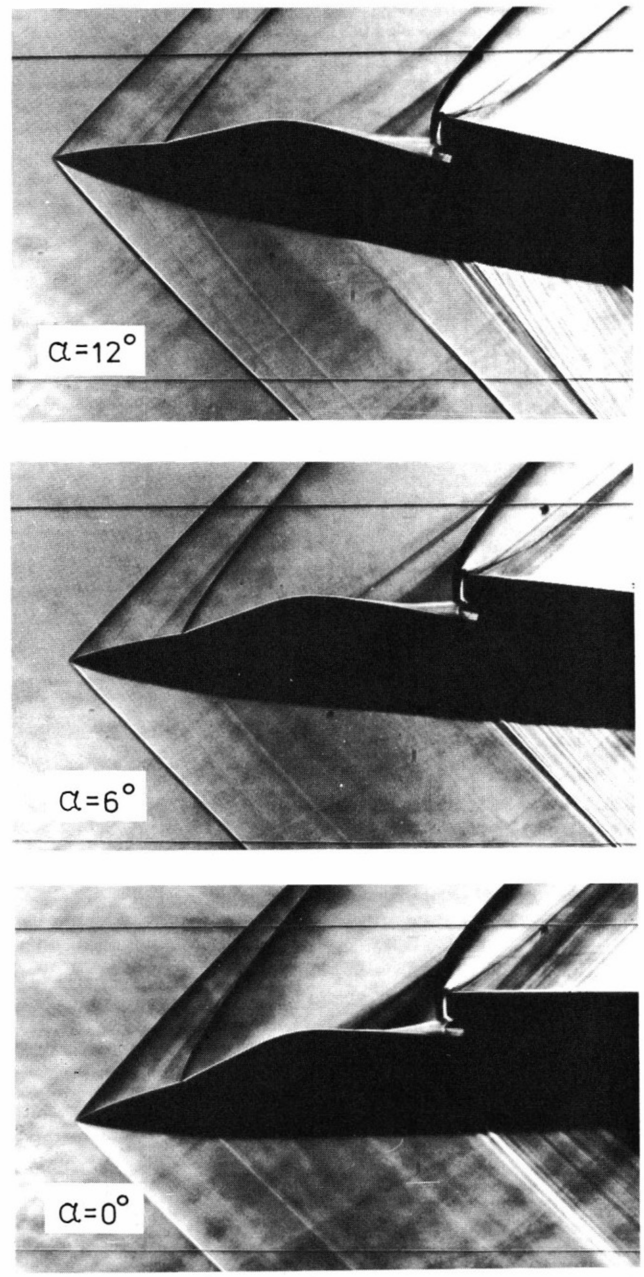

Figure 18. Schlieren pictures at M=1.52, β= 0° and 100% mass flow.

To study this separated region in more detail, oil flow tests were carried out and very complicated flow conditions were found, Figure 19. Two boundaries or fronts can be seen and the flow in the areas behind these fronts are both in forward direction but the details are not clear.

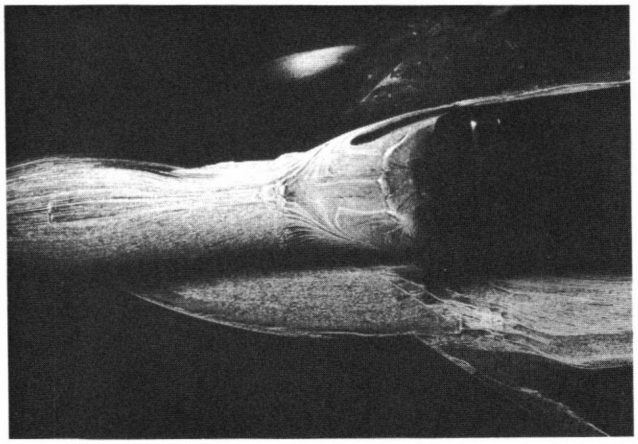

Figure 19. Oil flow picture at M=1.52, $\alpha=6^{\circ}$, $\beta=0^{\circ}$ and 100% mass flow.

The canopy pressure distribution, Figure 20, shows that there is no significiant increase in static pressure caused by the normal shock as could be expected. Furthermore, despite the maximum local Mach number at the top of the canopy being close to 2 the Mach number at the entrance to the inlet is below the freestream Mach number.

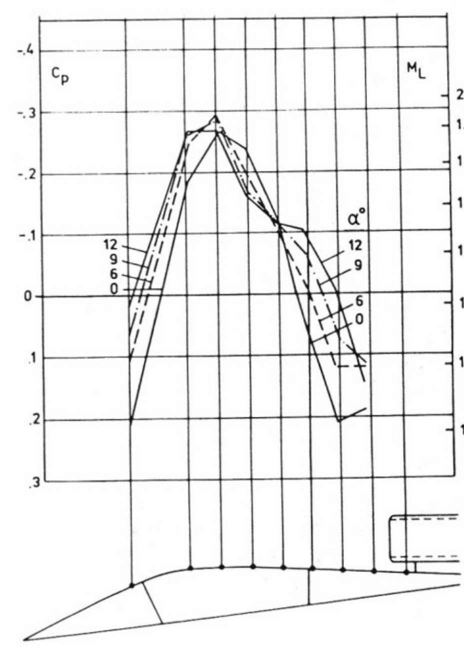

Figure 20. Canopy pressure distribution at M=1.52 $\beta=0^{\circ}$ and 100% mass flow.

At sideslip condition, Figure 21 shows that the inlet performance deteriorates. To try to get improvements in this respect a number of vortex generator configurations were tested. The best results were obtained with a configuration according to Figure 22, consisting of four vortex generators only one millimeter high and placed just forward of the earlier mentioned "fronts".

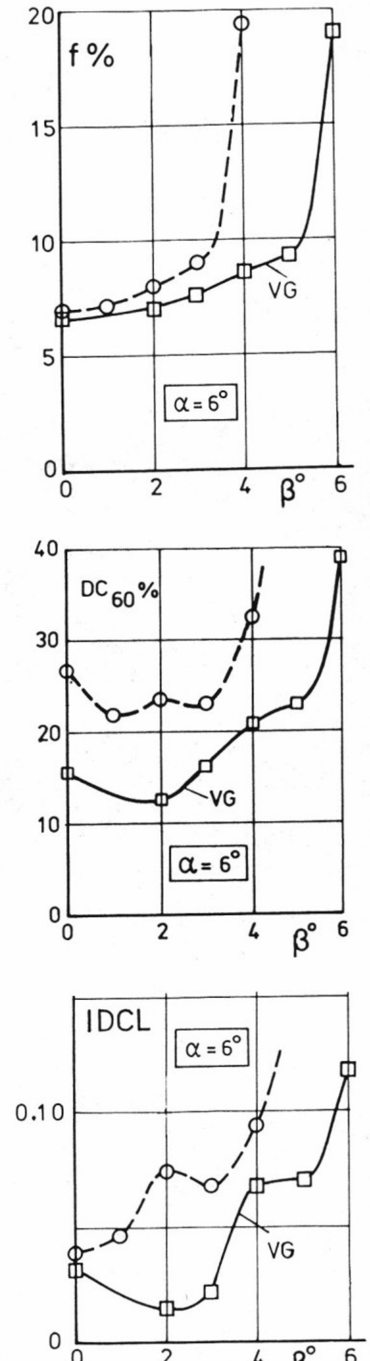

Figure 21. Effect of sideslip at M=1.52, $\alpha=6^{\circ}$, $\beta=0^{\circ}$ with and without vortex generators.

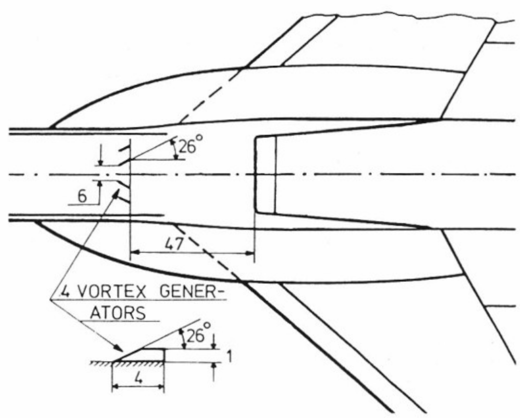

Figure 22. The vortex generator configuration.

The oil flow picture, Figure 23, shows that the generators affect the forward front effectively but also that the second separation front is almost unchanged.

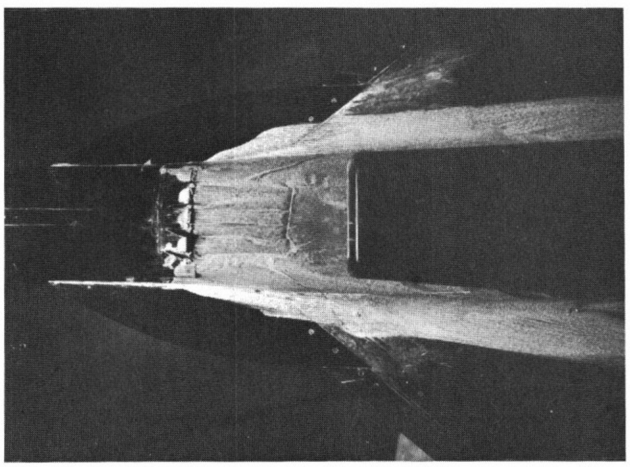

Figure 23. Oil flow picture with vortex generators in front of the inlet at M=1.52 $\alpha=6^{o}$ $\beta=0^{o}$ and 100% mass flow.

At zero sideslip this vortex generator configuration gives some improvements concerning the inlet performance, Figure 17. At sideslip, however, Figure 21 shows a significant improvement both in losses and distortion and probably more extensive work in this direction should give even better results.

One technique for improving inlet flow condition compared to an ordinary side-mounted installation is to shield the inlet by the fuselage as on the F-16 fighter aircraft. Total pressure losses for F-16 (found in open literature) are compared with losses obtained by this top-mounted inlet configuration in Figure 24.

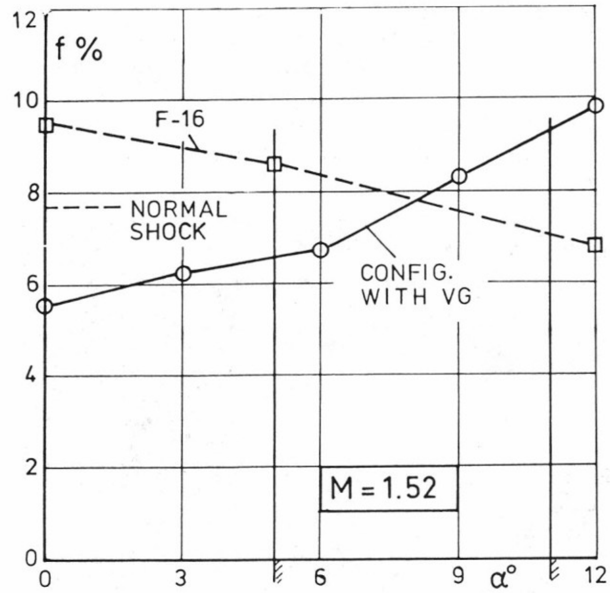

Figure 24. A comparison of total pressure losses for the top-mounted configuration studied and an under-the-fuselage installation (F-16).

The top-mounted inlet has lower pressure losses up to about α=8 deg. The losses are also lower than those for a normal shock up to the same angle. The main reason for this is probably the fact that the local Mach number in front of the inlet is lower than the freestream Mach number giving lower shock losses.

Conclusions

The study described in this paper indicates that an aircraft concept characterized by the inlet mounted on top of the fuselage has the potential of being designed to give good inlet performance at subsonic, transonic and supersonic speeds. The results at large angles of attack and at moderate sideslip angles are promising.

But there are certain problem areas. The most important one is the unstable flow condition achieved when reducing air inlet mass flow at higher Mach number. Also the performance at higher sideslip angles especially at M=1.52 ought to be improved. Both these problems are related to the adverse pressure gradient over the aft part of the canopy and the very complex flow conditions just in front of the inlet. It is supposed that the flow in this area is sensitive to Reynolds number and maybe it should be pointed out that these tests are conducted at relatively low Reynolds number which could affect the results obtained.

Nomenclature

f Total pressure losses $(p_{o_\infty} - p_{TAV})/p_{o_\infty}$

IDCL Engine fan circumferential distortion index

DC_{60} Engine fan distortion index based on the worst 60° section

M Freestream Mach number

M_L Local Mach number based on local static pressure and freestream total pressure

p_{o_∞} Freestream total pressure

p_{TAV} Engine face average total pressure

α Angle of attack

β Angle of sideslip

Acknowledgement

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