CONCEPTUAL STUDY OF A SUPERSONIC LOW OBSERVABLE HEAVY GROUND ATTACK MISSILE

Ola Hamnér*, Rolf Jarlås*, Johannes Johansson**
***Swedish Defence Research Agency, FOI **Swedish Defence Materiel Administration, FMV**

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Abstract

A heavy long-range supersonic missile with high manoeuvrability has been designed. The study has focused on aeronautical and signature aspects, and is based on future turbine engine technology. Aerodynamics and structural design have been paid quite in-depth attention including substantial computations as well as extensive wind tunnel testing. Flight dynamic simulations have been performed in order to check controllability, and radar signatures are calculated with physical optics and spectral element methods.

The missile is required to carry a 500 kg warhead a distance of 400 km at low altitude and Mach 1.5. After deceleration the missile shall, without pre-climbing, turn to the ground and hit it vertically. Low signature is primarily required in the sector between the horizontal plane and 45° upwards.

The study resulted in a final design of a 7.6 m long missile weighing 1.6 tons. It has low-aspect-ratio wings that are folded on the flat upper surface of the body during cruise. The body features a ventral bump-type air intake and a T-shaped exhaust nozzle, the latter resulting in low infrared signature and enabling thrust vectoring in pitch, roll, and yaw.

1 Introduction

Significant progress in propulsion technology is foreseen within the coming decades. Within aeronautics gas turbine engines will still be dominant and here the US programs IHPTET (Integrated High Performance Turbine Engine Technology) and VAATE (Versatile Affordable Advanced Turbine Engines) show the way to tremendous improvements.

In order to illustrate the impact of this evolution on long-range missile development, a conceptual study was performed investigating what could be achieved with a small future turbojet engine in a heavy ground attack missile. Stealth characteristics and high manoeuvrability were added to the long-range requirement, making this study more interesting for the armed forces.

The work, under contract from Swedish Defence Materiel Administration, FMV, should focus on aeronautical and signature aspects.

2 Requirements and Frame of Study

2.1 Requirements

Payload: 500 kg warhead of length 2.2 m and diameter 0.35 m.

Mission: 400 km cruise with Mach 1.5 at ≤200 m altitude. After deceleration the missile shall turn directly (without pre-climbing) to the ground at hit it vertically, Fig.1.

Signature: Low signature is primarily required during cruise in the sector between the horizontal plane and 45° upwards. There are no signature requirements during the terminal phase.

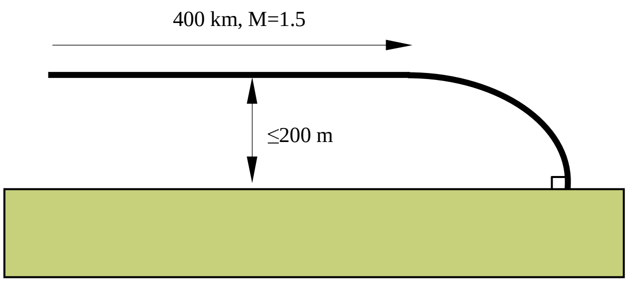


Fig. 1. Flight performance requirements.

2.2 Conditions

An engine with a thrust-to-weight ratio of 20 and specific fuel consumption of 14 mg/Ns was provided together with some relations for sizing with respect to max thrust. Standard JP-8 jet engine fuel is used.

The missile has no target seeker (advanced future accurate navigation is assumed). Antennas are flush mounted.

The missile is primarily thought of as being launched from the ground or a ship although also air launch might be of interest. Structures should be designed to withstand the loads from an adequate booster (launch lugs must however not be considered).

2.3 Frame of Work

The study intends to concentrate on structural design, aerodynamics, and flight mechanics together with the low-observable requirement. Propulsion is more focused on the integration part, since engine data is provided.

Detailed mechanical design, fuel and electrical systems, etc is treated superficial, mostly only as weight and volume estimations. Development and manufacturing costs are not quantified but only judged and used in comparison between different design options together with technical risk judgements.

3 Optional Concepts

3.1 Early Concept Ideas

The design process started with brain-storming sessions with many interested people invited. From these several widely different concept ideas appeared, a few shown in Fig. 2.

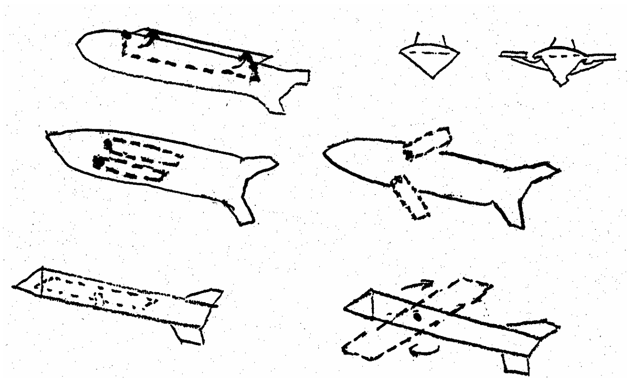


Fig. 2. Some early concept ideas.

After screening - with for different concepts more or less cursory analysis - the most vigorous concepts were brought to further investigation.

Different location of engine and warhead were studied briefly. A “normal” layout (warhead in front and one engine at rear) was early decided. Primarily for radar signature reasons a, in very broad terms, triangular body cross section was chosen with one triangle corner oriented downwards. For the same reasons one single air intake in ventral location was early decided.

Initial sizing and weight estimation was done by use of empirical data together with rough flight performance and structures calculations. Experience from earlier investigations of missiles with triangular cross-section bodies [1],[2] was useful for the aerodynamic analysis at this stage, together with semi-empirical methods and a panel method [3]. An about 7 m long missile weighing at least 1.5 tons came out as a result of these early estimations. Thrust vectoring was thought of as a means of providing necessary control forces during both cruise and the terminal phase. This was included, to more or less extent, in all the studied configurations. A T-shaped nozzle was rather early chosen, giving low infrared (IR) signature and also enabling thrust vectoring in pitch, roll, and yaw.

The most pronounced problem was to find a configuration which could manage the highly demanding terminal turn without the large drag and signature penalty associated with a wing during cruise, where it in fact is not needed. Folding wings of different types were obvious options, and some of these were further investigated. A concept with special rocket(s) for creating the necessary cross-force (instead of a wing) was found very interesting and quite extensively examined.

This investigation [4] found it rather problematic to - without any wing - decelerate acceptably before the onset of the manoeuvre. The most serious problem, however, was the difficulty to control the missile accurately enough. Tremendous requirements on rocket thrust alignment together with the judgements of the technical risks lead to the decision to abandon this concept. After this only "aerodynamic" concepts remained.

3.2 Aerodynamic Concepts

The aerodynamic solutions that survived the first screening were divided into three categories:

- High-aspect-ratio wings which are stored along the body. Swing-wing and pivoted single wing.
- Low-aspect-ratio wings which are hinged alongside the missile.
- Fixed low-aspect-ratio wings.

The first category is associated with attached flow and the other two with leeside vortex flow.

The high-aspect-ratio wings were designed for high lift conditions. They suffered, however, from practical implications with incorporating e.g. slotted flaps. It was also difficult to take full advantage of the body lift contribution (i.e. to adjust the wings to high body incidences). Since the wings must have a rather big span and be folded along the missile body, these concepts had an unwanted influence on the missile length, and meant serious problems for achieving adequate longitudinal stability. They were also found to imply a higher weight than the low-aspect-ratio folding wings. Problems when using these wings (designed for high lift) during the deceleration phase were also recognized.

Perhaps these high-aspect-ratio wing options should have earned some more efforts, but it was, however, decided to go further only with low-aspect-ratio wing configurations. A high wing location was obvious, and subsequently a solution with the wings stowed on the upper side of the fuselage was chosen. This, together with the fixed wing concept, was decided for the further, comprehensive, design investigation. So, at this stage two main candidates remained. Both of them were studied for different body and wing configurations.

4 Configuration Development

4.1 Body Shape and Wing Planforms

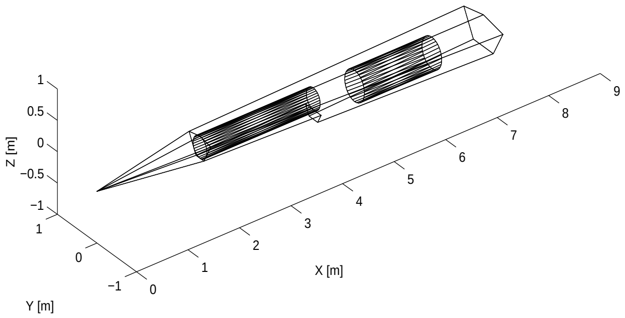


Fig. 3. 3D view of missile fuselage.

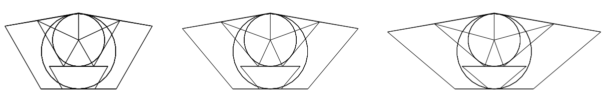


Fig. 4. Side slopes of 30º, 40º and 50º.

After the initial rough estimations of drag, engine size, weight, volume, etc a parametric aerodynamic study of the missile body geometry was performed with the constraint of housing primarily the warhead, the engine, and the air intake. Zero-lift drag and induced drag were determined by Computational Fluid Dynamics (CFD) Euler calculations for several body geometry parameters such as nose slenderness, nose camber, fuselage height and width, Fig. 3 and 4. Since small radar cross-section (RCS) is primarily required from above, all sharp corners on the lower part of the body have, after this optimization, therefore been well rounded.

The body width has a practical influence on maximum possible span for the folding wing concept. This limitation means a significant drawback (for achieving the needed lift during the terminal manoeuvre) as compared to the fixed wing alternative, which also admits freedom to design an optimal planform with respect to longitudinal stability. On the other hand the latter has to be a compromise between the supersonic cruise and the subsonic high angle of attack manoeuvre demands.

A large number of different wing planforms were investigated, both with and without the constraints of folding. Wing study efforts were focused on highest possible lift during the terminal manoeuvre and a nearly neutral longitudinal stability. They were investigated with extensive Euler calculations and wind tunnel tests. Some of the investigated planforms are shown in Fig. 5 with calculated pressure distribution. Symmetric as well as sideslip cases were calculated and measured.

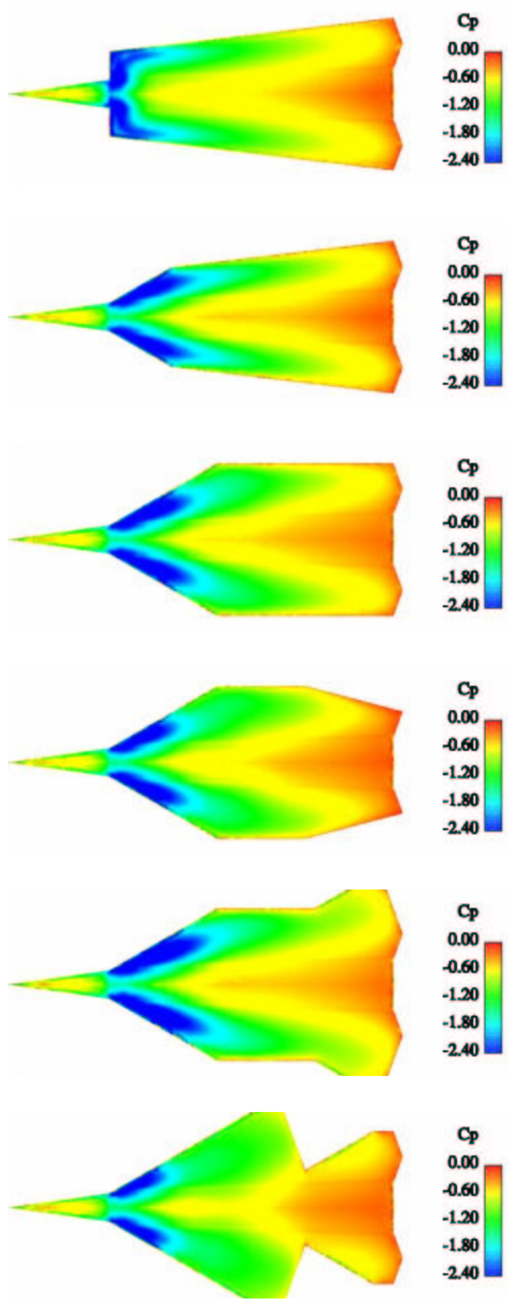


Fig. 5. Leeside pressure distribution at Mach 0.5, 30° angle of attack, for some of the investigated wings.

4.2 Fixed vs. Folding Wings

In order to make a fair comparison between the fixed and folding wing concepts they were subjects for rather in-depth analysis. This included weight, aerodynamics (total drag during cruise and ability to perform the end turn, including controllability), aeroelasticity (static and flutter analysis), RCS, and IR signature (only qualitatively). The wind tunnel test results were a solid base for the aerodynamic considerations.

Folding wings mean weight penalty due to the hinge, lock, and ejection mechanisms. Although also wings that are folded contribute to wave drag due to increased cross-section area, the fixed wing missile experienced higher total cruise drag. Because of the need for a larger engine and more fuel the latter was found to be heavier at launch but – interesting enough – somewhat lighter in the terminal phase, where weight has a tremendous influence on performance (whilst it has marginal influence during cruise).

RCS calculations showed (not surprisingly) a significant penalty for the fixed wing option. Since the two candidates came out with rather similar flight performance, the RCS aspect became decisive, resulting in the choice of folding wings. Storage and logistics considerations also helped this decision.

4.3 Geometry Modifications of the Configuration with Folding Wings

The upper side of the body was modified into one flat surface together with an increased body width. This made space for a larger wing span, which was possible without significant increase in cross-section area for the cruise configuration (i.e. folded wings) as a result of a neat storage. This also allowed greater freedom in fine-tuning the wing planform, since adjustment to the body ridge is not needed. Furthermore, this modification enabled a higher span for the horizontal part of the nozzle outlet (providing better thrust vectoring roll control, and also meaning reduction of the IR signature.

The increased body span reduces the cruise induced drag (which is rather marginal) and angle of attack (favourable for RCS contribution from the intake). One large flat body upper surface means also, in most illumination situations for our special signature priorities, an overall better radar signature than a body with a ridge.

Unacceptable RCS contribution from the intake and its diverter lead to the adoption of a bump-type intake (which also suits the rounded lower body), replacing the initial design with rectangular section. The rectangular intake was designed to work reliably at supersonic cruise as well as subsonic high incidence conditions. More uncertainties are associated with the efficiency of the bump intake, especially during the high incidence turn. However, it was found likely that the engine would stay alive during this very short timeframe, although with somewhat reduced thrust (and hence control force).

The ventral fin was subject for substantial efforts trying to minimize the RCS contribution, among them an aerodynamic investigation [5]. However, no acceptable solution was found and this conventional fin was abandoned. Sufficient directional stability is instead provided by a downwards prolonged rear part of the body, thus enabling a higher aspect ratio of the vertical part of the nozzle outlet, meaning better directional thrust vectoring and also still lower IR signature. An aerodynamic rudder had earlier been found unnecessary.

Computer aided design (CAD) was used from a rather early stage in the design process. This has been very helpful for the interior configuration, for the folding wing geometry, and for locating fuel such that centre of gravity position can be kept practically constant. CAD was used all the way to the last fine-tuning and a summary of this work with different configurations is found in [6].

The elaborative wing and body design work resulted in a missile being almost perfectly aerodynamically balanced for the configuration with wings folded as well as for wings deployed (the latter at subsonic speeds).

5 Structural Design Considerations

5.1 Discussion and Preliminary Calculations

The responsibility for a structural designer is normally defined as to determine the structural design with the lowest structural weight, satisfying stress and stiffness constraints and possible to squeeze into the volume bounded by the external and internal surfaces required for aerodynamic and other reasons. Allowable stress levels depend primarily on the chosen materials and the environmental conditions such as temperature and number of load cycles, whereas the stiffness constraints are often an effect of aeroelastic considerations, such as elevon efficiency and flutter speed. However, initially in this project the most important activity was to argue for designs giving a short load path (distance between aerodynamic lifting surfaces and heavy loads) and sufficient internal volume for fuselage frames, to reach the structural target weight. It was also important to argue for a short distance between the warhead and the engine to avoid flutter. The discussions, considerations, analyses and decisions leading to the proposed preliminary structural design were reported as a part of the project [7].

In the first calculation a slender fuselage represented by a beam was considered and the lowest bending frequency was determined. The fuselage considered was a thin-walled tube with the thicknesses for seven lengthwise positions as parameters. After a few iterations the lowest eigenfrequency had increased from 31 to 38 Hz, while the mass for the fuselage tube was reduced by 12%. To achieve this bending stiffness the thickness in the middle of the fuselage was much increased, and the distance between the warhead and engine containing air intake and duct was reduced as much as possible.

A high-aspect-ratio wing with advanced high lift devices was considered next. With this design the total wing area required for the subsonic attack manoeuvre could be reduced considerably as compared to a large chord length design, for which efficient high lift devices are difficult to achieve. Obviously the spanwise load path is long between the centre of each high-aspect-ratio wing and the heavy warhead, and calculations showed that a very thick wing or a thin but very heavy wing structure was needed.

A low-aspect-ratio wing design with a chord almost as long as the fuselage length was also considered. Ideally, from a structural point of view, the mass- and lift- distribution from nose to tail should agree. The lift produced by the wings should be transferred into the fuselage at the chordwise position were it was generated. To be efficient this design should have a large number of wing attachment points. With a small span a very thin wing can be designed without violating the weight requirements. A folding wing with a relatively small diameter hinge-and-lock mechanism along almost the whole fuselage side might also be possible.

Early in this project the structural designer had very limited information. The total weight of the missile was expected to be very approximate since the engine and other specifications and requirements were quite unique. To house the warhead, intake duct, engine, and nozzle into the fuselage behind each other, the length must be in the order of *seven* meters. With the high load factor expected for the attack manoeuvre a low-aspect-ratio wing would probably finally be chosen. Such wings may have a maximum lift coefficient of about 1.5 rather than about 3 for advanced high-lift configurations. For a manoeuvre following a circular path with given radius, the wing area required is independent of the speed. Assuming a chordlength of *five* meters, a fuselage width of *one* meter at the horizontal upper surface and *one* meter semispan for one wing gives a total projected area of 15 m^2 for wings plus body. At this point some numerical work can start, but many important decisions from a structural point of view have already been taken.

To investigate if the wing thickness was likely to be governed by aeroelastic constraints (flutter), or by the stresses caused by the attack manoeuvre the following assumptions were made. The wing planform was assumed to be rectangular 7x0.7 m and a reasonable speed for the attack manoeuvre was assumed in order to get an estimate of the maximum wing-loading. The calculations indicated that a homogenous wing was at least three times as heavy and aeroelastic properties were poor compared to a sandwich construction. A sandwich wing would be about twice as thick but still very thin.

A sandwich wing with root thickness of 28 mm was chosen. Weight of one wing pair is 110 kg.

The material properties were assumed to be those for an aluminium-alloy available today. Duralumin has been used in aeronautical applications for more than 80 years, and aluminium alloys are often chosen for aeronautical applications. Other materials should be considered for further optimization of the missile. Carbon fibre reinforced plastic may be one alternative, but a standard epoxy will not work in the high-temperature environment and standard prepreg plies cannot be used for manufacturing of the very thin skin in the sandwich wing. The choice of material does *not* however reflect an opinion about the expected future development of new materials. On the contrary there are interesting developments going on, such as nano-materials, indicating the potential for new materials with much improved properties.

With the motivation that regions with too high stresses can be identified, as more reliable and accurate programs for stress analysis can be used, it was assumed that the allowable average stress level in the structure could be somewhat increased as compared to the levels used in practice today.

5.2 Hinge-and-Lock Design

The size of the hinge-and-lock mechanism was calculated for the folding wing. A very compact generic design was developed. Using standard stress calculation handbook formulas put into a computer program some 6000 design alternatives were analyzed and the most compact one satisfying all strength criteria was identified. The best design makes it possible to store two folding wings, each with a semispan being almost 90% of the fuselage width, Fig. 6.

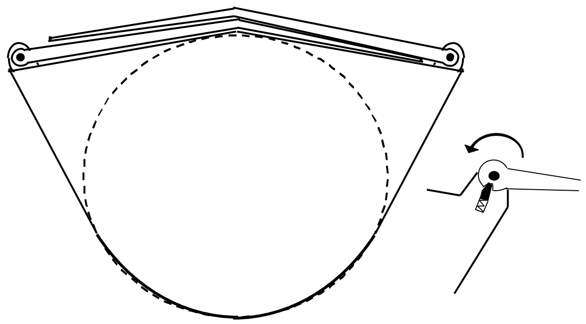
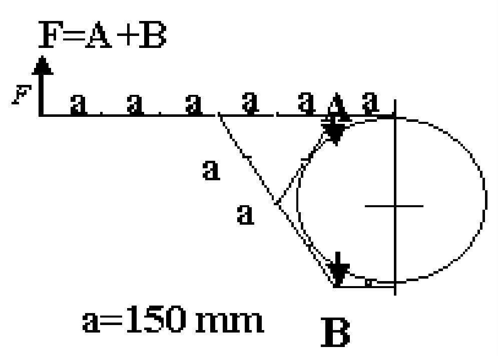


Fig. 6. Cross-section showing folded wings with hinge-and-lock mechanism.

5.3 Finite Element Analysis

To determine approximately the size and weight of the fuselage frames a finite element (FE) model of a section including two frames and the skin between them was created using 117 shell elements and 20 beam elements. The skin in the fuselage was divided into ten regions with one thickness parameter for each region, and similarly the frame consisted of ten beams with individual cross-sections, Fig. 7.

An optimization was made, where the direction of steepest descent was calculated manually from the difference quotas obtained by performing two subsequent finite element analyses with different skin thicknesses or beam cross sections. Allowable stresses in the fuselage, and in the part of the wing being modelled, were constraints during this optimization and the height of the frames directly above and below the engine/warhead was considered since an increase here reduces the slenderness of the missile. The skin thickness determined was used in an analysis of the fuselage bending frequencies similar to those described above. It was observed that the skin thickness in the upper part of the fuselage was sufficient but a keelson underneath a part of the fuselage had to be introduced.



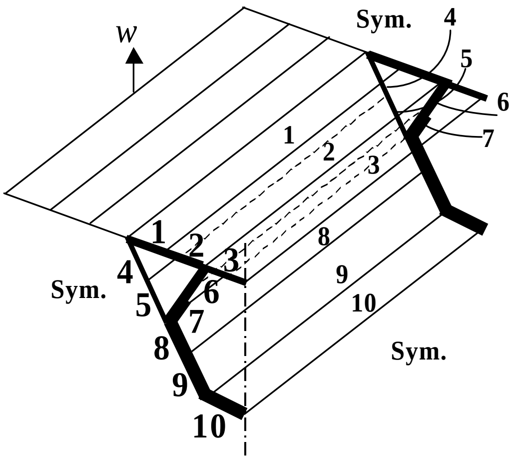


Fig. 7. Dimensions for a typical section of the missile, assumed for calculation of "optimal" frame and panel dimensions.

The structural mass and the stresses obtained with the simple shell-and-beam type finite element model were compared with results of a much more detailed 3D solid-element analysis. In this analysis the in-house developed FE code STRIPE [8],[9] was used. The model consisted of 922 elements and the equation-system had 270000 unknowns. As this 3D model contains a more realistic description of a number of details it is 17% heavier. The wing deflection was only 86% of that for the simple model whereas the highest stress in the frames was 16% higher in the 3D-model. The increases in calculated stresses for wing and fuselage skins were 6% and 15% respectively. The very high stresses calculated near some of the sharp corners in the present model, shown in Fig. 8, were not considered, as these corners should be rounded.

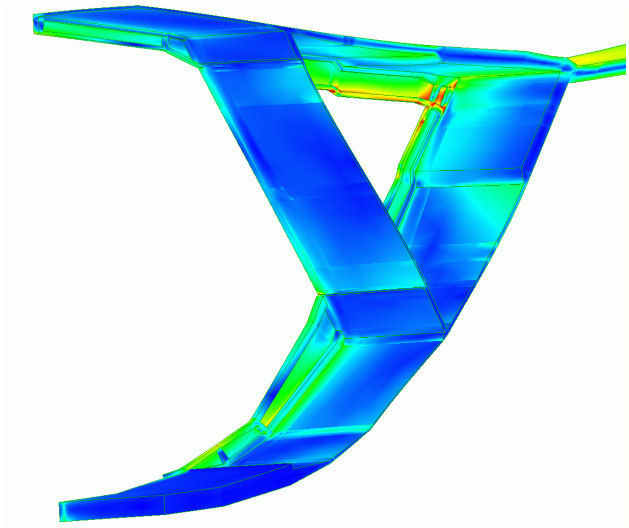


Fig. 8. FE model of one half of a fuselage section containing wing root, fuselage panels, and one frame with I-shaped cross-section.

5.4 Aeroelasticity

Aeroelastic analysis [10] was initially performed for the rectangular 7x0.7 m^2 flat plate with stiffness and mass properties for a sandwich construction. Flutter speed for this wing and static aeroelastic efficiency factor for a rectangular trailing edge flap were investigated. Then a more realistic model of the fuselage with dimension obtained as described above and a sandwich construction wing with optimized skin and honeycomb thicknesses was introduced. Mass distribution as well as the fuselage and wing designs was similar to the final design. To improve the aeroelastic behaviour of the missile a very stiff frame was introduced around the air intake. It was observed that if the total mass for the missile was increased without changes of the mass distribution, then the bending stiffness of the fuselage must be increased to avoid flutter. The associated mode is shown in Fig. 9.

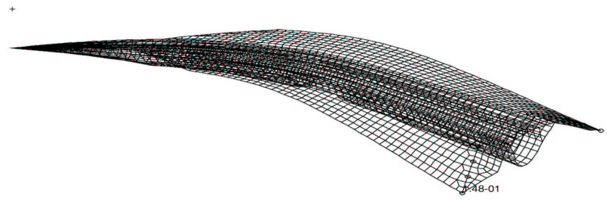


Fig. 9. Mode shown is causing flutter when the mass is increased.

5.5 Structural Dynamics

The maximum bending stresses in the wing, and the maximum stresses in the hinge-and-lock mechanism, were predicted in a dynamic simulation. The analysis indicated that the wings and the hinge-and-lock mechanism were sufficiently robust and the wing was sufficiently low-weight to allow unfolding of a wing in approximately 0.1 seconds.

A study of vibration environment concerns for electronic equipment was also undertaken [11].

6 Aerodynamics

A number of different aerodynamic investigations were performed, ranging from handbook and panel methods to CFD and wind-tunnel tests.

6.1 Calculations

Initially a parametric investigation was performed in order to see the influence of the body shape, nose length etc (as shown in Fig. 3 and 4) on the aerodynamic behaviour of the missile. These were done using Euler calculations with an in-house code EDGE [12]. Earlier calculations on configurations with similar body types [2] had shown good agreement between Euler calculations and wind tunnel tests, also at higher angles of attack. It was therefore natural to use CFD for preliminary design of the missile.

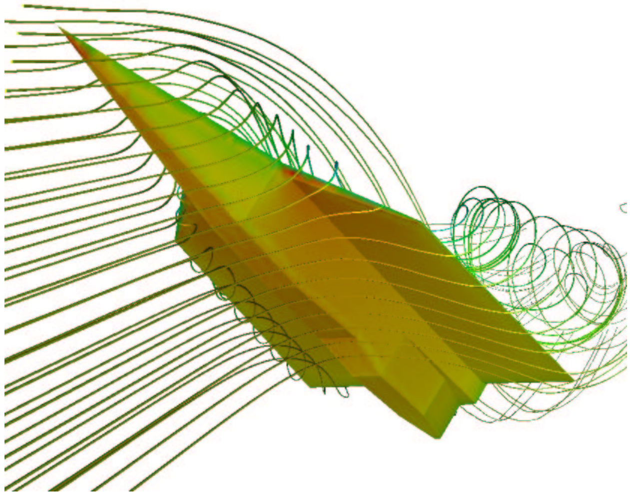


Fig. 10. Example of CFD calculation of wing configuration at high angle of attack.

After the body shape optimization, CFD was used to study a large amount of wing planforms (a few of them shown in Fig. 5). This was mainly done as a parametric investigation. Many of these wings were later investigated in wind tunnel.

CFD was also used as a tool when designing the air intake. In early models the intake was replaced by a ramp as shown in Fig. 10. Later the intake was investigated in more detail. At first a highly staggered intake was used in order to obtain good flow quality both at supersonic cruise speed with small angles of attack, as well as subsonic speed with high angles of attack during the end manoeuvre. A conventional means of boundary layer control was used.

Later the body was rounded around the air intake which made a “bump” intake more suitable, Fig. 11.

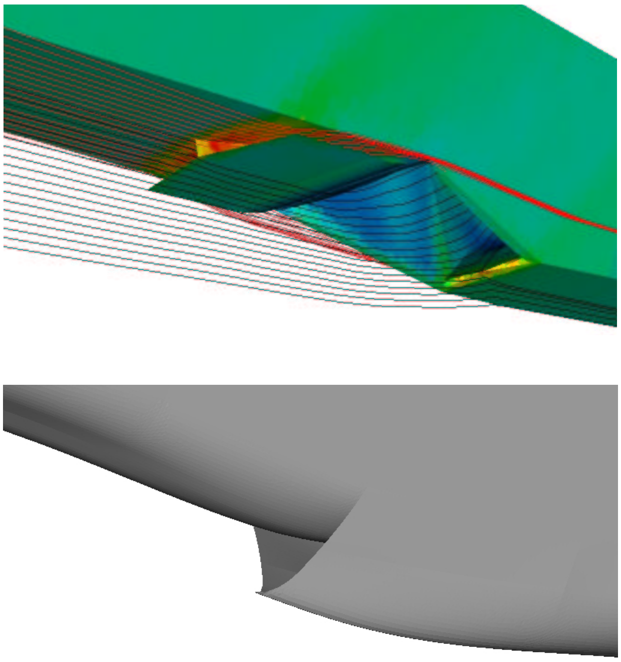


Fig. 11. Staggered intake with conventional boundary layer control (top) and "bump" intake (bottom).

The inviting *fluidic* thrust vectoring systems were subject for special studies. No reliable system able to provide the necessary control forces could, however, be found. A conventional concept with two-dimensional deflectors on both sides of each outlet leg was therefore chosen. The optimal deflector chord (with respect to control force related to hinge moment) was found to be 0.025 m in a special CFD investigation [13]. The flow at the outlet is supersonic.

6.2 Wind Tunnel Test

The terminal manoeuvre will be performed at very high angles of attack, around 35º, and hence the missile behaviour and controllability is very crucial in this aerodynamically complicated region. Phenomena such as stochastic yaw and roll moments, vortex breakdown, and strong secondary effects of control deflections make a wind tunnel investigation motivated as a complement to the CFD calculations.

Wind tunnel tests were performed with about 15 different wings and two different ventral fins/rudder for cruise (Mach 1.5) and manoeuvre (subsonic up to 45º angle of attack and 25º angle of sideslip) conditions. Wings were of both fixed and folding type [14]. The campaign [15] was focused on stability and control studies but was also used for drag determination (model without flow-through, however). Complementary Mach number sweeps were also performed. As expected for these very slender configurations the Mach number dependence is moderate.

Fig. 12. Example of configuration in wind tunnel S4.

The most important outcome of the test campaign was:

- Test results were in good agreement with the Euler calculations, and thus making the latter acceptable for further configuration modifications and fine-tuning.
- Stochastic asymmetric forces and moments at zero sideslip were not pronounced and quite manageable within the angle of attack range of interest.

6.3 Analysis

An example of stochastic forces at zero sideslip and a comparison with CFD is shown in Fig.13. Here it can be seen that CFD fails to capture some aerodynamic behaviour at high angles of attack. These forces are however small, as the side force coefficient only reaches about 2 at zero sideslip while the lift coefficient reaches a value of about 150 (the reference area is uncommonly small).

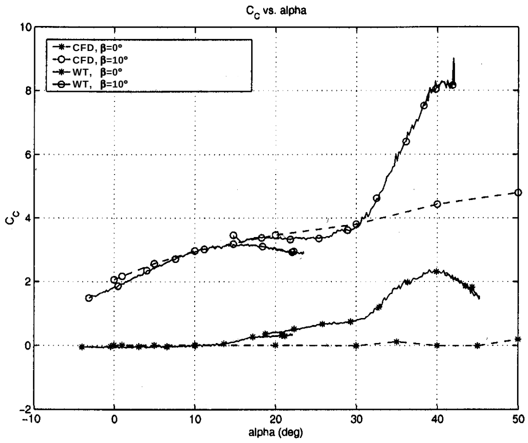


Fig. 13. Comparison of CFD and wind tunnel test results. Side force coefficient vs. angle of attack at Mach 0.5, with and without sideslip.

In order to further investigate the very important characteristics at high incidences, an oil-flow visualization test was also carried out [16]. (As a result of the experienced good behaviour an ongoing investigation of possible nose vortex flow control was not continued.)

Fine-tuning of the missile configuration included design of the fuselage rear part such that adequate directional stability was provided without increasing drag. Two alternative geometries, investigated using CFD, are shown in Fig. 14. The upper configuration was chosen. This one resulted in higher directional stability and no significant difference in drag was found.

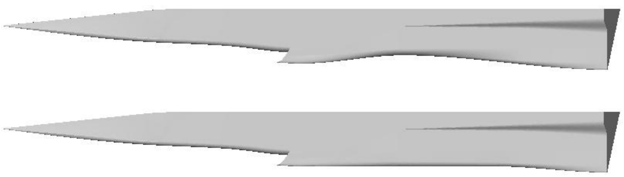


Fig. 14. Two studied alternatives for rear fuselage geometry.

The missile is - with wings unfolded or not - well stable in roll and around neutrally stable in pitch and yaw.

Zero-lift drag for the supersonic cruise condition is determined from wind tunnel measurements (forebody pressure drag is in very good agreement with Euler calculations) and adjusted for configuration differences (including flow-through and spillover) using Euler calculations. Friction drag is determined with a 4% reduction due to riblets, and the body base drag is predicted with a reasonable mean base pressure coefficient of -0.24 (in presence of the jet). Induced drag during cruise is just a few percent of total drag. Zero-lift drag with deployed wings (at subsonic speeds) has been less accurately determined than cruise drag.

Dynamic derivatives were calculated using a combination of handbook methods and a panel method [3].

All aerodynamic data of the final configuration have been determined from wind tunnel test data in combination with results from calculations. Using a special developed computer program [17] data has been compiled into a comprehensive aerodynamic database suited for the nonlinear flight dynamic model.

7 RCS Calculations

Two different kinds of RCS calculations were performed. First a physical optics method, FOPOL [18], was used for a broader survey and to investigate a number of different configurations. This method is suitable for high frequencies and it requires relatively little computation time. It could early be seen that the wing would significantly increase the RCS of the missile compared to a configuration without a wing, Fig. 15. This helped in the decision of using the folding wing instead of the fixed wing.

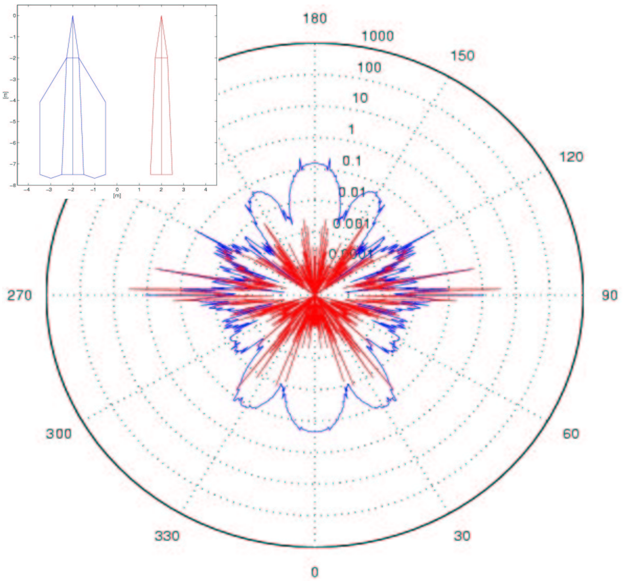


Fig. 15. Monostatic RCS [m2] of configuration with (blue) and without (red) a wing using physical optics. Horizontally polarised plane wave at 10 GHz coming in at 30° from above.

A few calculations were also performed using a spectral element method [19],[20] which is a more advanced method requiring heavy calculations. This was therefore only used to calculate the RCS on a late design in order to see the effects of edge diffraction which is not captured by the physical optics method and to investigate the lower frequency behaviour, Fig 16.

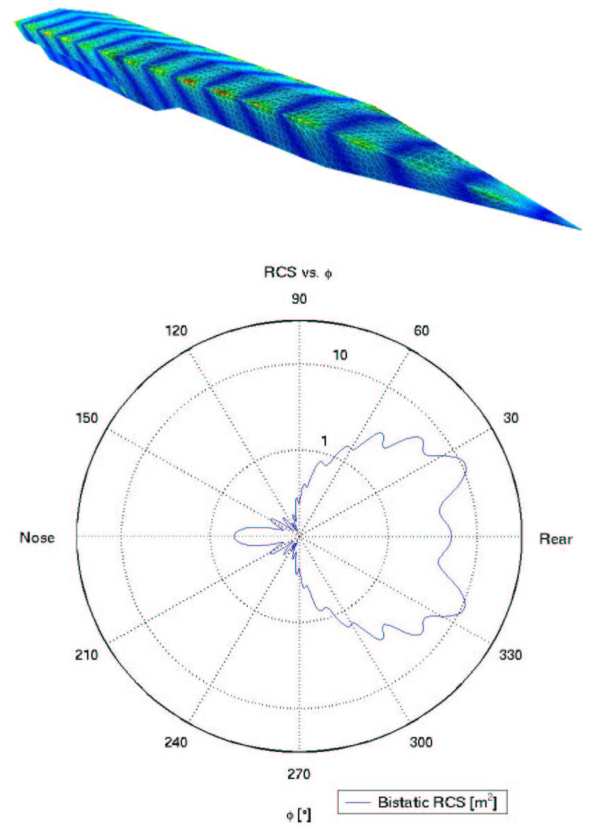


Fig. 16 Electric field on missile surface (top). Bistatic RCS [m2] with horizontally polarised plane wave incoming from nose direction at 0.3 GHz using spectral element method (bottom).

8 Final Design

This conceptual study has resulted in a final design of a 7.6 m long missile weighing 1 635 kg. For carrying the 500 kg warhead to the target a 125 kg engine, 280 kg fuel, and a structure weight (including subsystems) of 730 kg is needed.

Main features can be seen from Figs.17-19. The seekerless homing concept allows for a very slender pointed nose. Span with deployed wings is 2.7 m and with folded wings (i.e. body max width) is 1.1 m. From Fig. 18 locations of warhead (red), fuel (yellow), and engine (brown) can be seen. Most subsystems are located in the rearmost part of the fuselage.

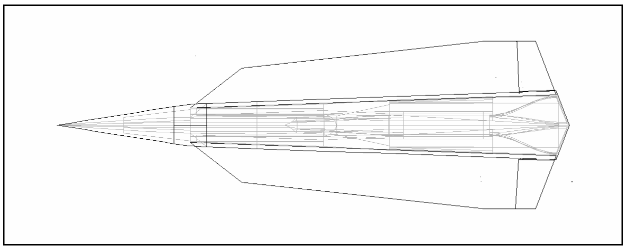


Fig. 17. Final design. View from above.

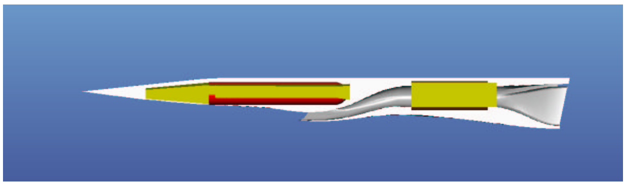


Fig. 18. Cutaway side view.

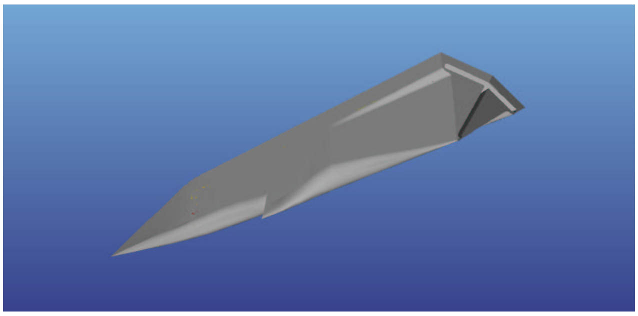


Fig. 19. Missile in cruise configuration (folded wings) showing T-shaped thrust vectoring nozzle.

For aerodynamic and stealth reasons a thin cover is placed over the folded wings, resulting in a continuous outer surface. Most wetted surfaces are covered with riblets in order to reduce friction drag, at some locations combined with radar absorbent material. The body base is slanted in different parts to reduce RCS, primarily from tail-on direction.

The trailing edge elevons are actuated with an especially designed mechanism that allows folding of the elevons together with the wings. This mechanism is totally housed inside the fuselage. Wings are deployed with pneumatic actuators which are aligned with the hinge lines. A more detailed description of the mechanical design can be found in [6].

As a consequence of the special intake and nozzle design the installed thrust is set to 80% of the provided engine data. This was done as an extrapolation of known data for other bump-type intakes and less pronounced 2D nozzles.

9 Flight Dynamics Simulation

A prescribed flight path was defined for the missile to follow during its approach to the target area. After deceleration to subsonic speed and unfolding of wings, the missile rolls to an upside-down position so that the air intake faces the wind during the following turn. The angle of attack is about 35° during most part of the turn where both thrust vectoring and elevons are used. Finally, it is directed vertically (forcing angle of attack to zero) just before hitting the ground. The simulation model was used for confirmation of the missiles ability to manage this under total control.

The simulation that was carried out for the validation of the missiles ability to perform this prescribed manoeuvre was based upon the developed nonlinear flight dynamic model. The aerodynamic database, thrust, mass, and inertia data were used together with a simple flight control law in order to improve the stability characteristics of the pitch and yaw modes of the missile. The orientation of the missile in the air was conducted by controlling the rotational velocities around the principal axis of inertia on the missile.

The nonlinear flight dynamic model was implemented to simulations using the simulation package MATLAB/Simulink.

10 Concluding Discussion

The missile has been shown to meet the requirements, marginally as it should be.

Signature requirements were never quantified. This could perhaps have resulted in a rather different configuration. Here the design just aimed at lowest possible signature without strong such constraints.

IR signature was not calculated, partly because of some unknown jet characteristics for this futuristic engine. An acceptable device for shielding (from above) of the hot jet and its deflectors could not be found. Problems encountered were primarily how to avoid degrading of the thrust vectoring effectiveness without serious implications on RCS and aerodynamics.

A well-working fluidic thrust vectoring system is highly desired, thus avoiding the significant RCS contribution from the deflectors.

Body base drag represents around half of the cruise drag. It should be possible to reduce drag by means of base area reduction. This would, however, lead to a still longer missile if the gain would not be eaten by a fatter missile.

In order to shorten the deceleration time, air brakes could be introduced. Such things were studied for the "cross-rocket" concept before it was abandoned. However, since short deceleration time was not a requirement, it was decided – in concurrence with FMV – not to introduce them.

One idea from the brain-storming was to approach the target at an offset distance, thus making the turn less sharp (and maybe fooling the enemy). This might result in a smaller wing area, but the more difficult final erection makes the winning doubtful. It was therefore decided to keep to the straight-on approach, and the possible benefit from this idea was not investigated.

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