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Delft University Research in space launcher technologies

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In this paper the (Advanced) Space Launcher Research program of the Delft University of Technology, Faculty of Aerospace Engineering, will briefly be described, including some of the major results obtained sofar. The research program was initiated early 1992 and is supported, in part, by the Netherlands Agency for Aerospace Programs (NIVR). The main goal of the research program is to perform (long-term) research necessary to develop advanced technologies for future space launchers and (re)entry capsules and to allow improvement of the Faculty's curriculum.

The main activities in the program are coordinated with Dutch industry and research institutes as part of a joint research program on advanced technologies for space launchers, referred to as AEOLUS.

The Faculty's research program includes system analysis, aerothermo-dynamics, flight dynamics, and structures and materials. The investigations concerning systems analysis and flight dynamics concentrate on fully integrated design and design optimization, 6 degrees of freedom flight simulation and joint state- and (aerodynamic) parameter estimation. The studies on aerothermodynamics aim to improve the numerical simulation of (high) super- and hypersonic airflow, including the accompanying heat transfer and chemical reactions. The studies on structures and materials concentrate on high thermally-loaded structures in super- and hypersonic airflows and on the use of fibre-metal laminates.

1. Introduction

The Faculty of Aerospace Engineering (faculty LR) is one of the 13 faculties of Delft University of Technology (TU-Delft). It was set up in 1945 as a national academic institute to stimulate education and research in aeronautical engineering. The faculty has a teaching and research staff of 120 persons and a support staff of 90 persons divided over 5 disciplines. These disciplines include aerodynamics, structures, production and materials, aerospace design and flight mechanics, stability and control.

At regular time intervals, the faculty plans long-term programmes (typically 5 years) for interdisciplinary faculty research in close cooperation with the Dutch aerospace industry, among which are Fokker B.V., National Aerospace laboratory (NLR), KLM, etc. These programmes are especially intended to allow students to improve upon their understanding of aeronautics and their aeronautical engineering skills in close relation and interaction with Dutch industrial companies and research institutes.

One such, long-term, faculty programme is the Space Launcher Technology Research Programme, which aims at the development of technology and design principles for space launchers with aeroplane-like features. This programme, sometimes also referred to as the Aerospace Plane Programme, has been initiated in 1993 and is carried out in close contact with the Netherlands industrial space plane platform (AEOLUS) including Fokker Space & Systems B.V. (FSS), NLR, Stork Product Engineering B.V. (SPE) and TNO-Prins Maurits Laboratory (PML).

The reasons for selecting space launchers with aeroplane-like features as a topic for faculty research are fourfold. Firstly, the transportation potential of space planes cannot be ignored, at least not within the present situation of tight (space) budgets and the present high cost of space transportation. For Europe, this is an estimated US$ 1 billion per year (excluding launcher development costs).

Secondly, the design and development of space launchers poses a formidable task to future aerospace engineers. This is, not only,

because it requires major technology advances in almost all fields of aerospace technology, with possible spin-off to super- and hypersonic airplanes, conventional air transportation, and other high technology branches of industry, but also presents a wide range of possible design solutions, each with its specific (dis)advantages, which must be assessed.

Thirdly, advanced launchers have a declared Dutch industrial interest; In 1993, FSS, representing the principal industrial space capability in The Netherlands, together with SPE, NLR, PML and the faculty LR started AEOLUS, with the aim to secure the continuity of The Netherlands' contributions to space launcher development and manufacturing.

Fourthly, the faculty LR is of the opinion, that it is well equipped to prepare tomorrow's engineers for the formidable task of designing and developing airplane-like space launchers. For this, the faculty offers:

- education in space technology as an integral part of the study for aerospace engineer with amongst others courses on rocket propulsion, jet propulsion, hypersonic aerodynamics, orbital mechanics, flight mechanics of space launchers, space system aspects, spacecraft attitude control and manned spaceflight to chose from.
- a dedicated student-group exercise on the design of advanced launchers.
- a possibility for student exchange under the EEC Erasmus student-exchange program.

The faculty's advanced launcher research programme aims at a basic understanding of the principal problems associated with the design, development and production of space planes. It is concentrated, presently, on five topics, including design principles, flight dynamics, fibre metal laminates, passive thermal protection systems and aero(thermo-)dynamics. For all these topics, programme coordination occurs through regular meetings of the principal investigators. To date, approximately 9 graduate students, 2 Ph.D. student and 2 post-MSc. students contributed to the programme. In addition, a group design exercise has been performed on a two-stage-to-orbit (TSTO) space plane, involving approximately 25 graduate students [HARALD, 1993]. Hereafter, a detailed overview is given of the research carried out sofar.

2. Space launcher design

In the space launcher design field, LR is aiming at the conceptual design of (advanced) space launchers, in order to allow for the ranking/selection of space launcher concepts and for technology ranking (in terms of mostly performance and cost benefits) and for the determination of the criticality of the technologies needed.

Compared to conventional rocket launchers and airplanes, airplane-like space launchers are characterized by very strong relations between subsystem performance and overall system performance [Furniss and Walters; 1990]. To solve for this problem, a truly multi-disciplinary approach seems necessary, capable of not only handling thermal protection, materials, structures, propulsion (including precompression and single expansion ramp nozzle), fuels/propellants, and flight simulation to a sufficient level of detail, but which also allows for extensive design analysis, parametric analysis and design optimization.

The approach chosen at the start of the programme in 1992 is based on the experience gained with the in-house developed, conceptual and preliminary Aircraft Design and Analysis System (ADAS) for subsonic aeroplanes [Bil; 1988]. This computer tool already provides extensive support for design definition and evaluation, including parametric optimization as well as multivariate optimization (based on Standard Quadratic Programming; SQP). Planned developments and extensions are the inclusion of methods capable of determining the propulsion, aerodynamic and thermal protection characteristics of the vehicle at hypersonic flight speeds, the use of optimization methods for space launcher design and mission analysis including flight path optimization and the prediction of the environmental load as well as the costs of the launcher.

Development efforts, sofar, have led to the design tool SPADES (SPAce plane DESign) [Oving; 1994] as a first step in the further development of ADAS. SPADES already allows for a fair degree of conceptual design of multiple staged and/or fuelled, winged, air-breathing launchers using either rocket or air-breathing jet propulsion and which can take-off and land horizontally. For SPADES, an efficient design process (i.e. tasks of the tools and the order in which they are addressed) is chosen, so that a minimum of internal iterations is required to arrive at a valid design, see figure 1. It basically includes vehicle definition, vehicle analysis, mission analysis, vehicle scaling, costing, and

design optimization.

Vehicle definition

The first step in the design process is to define a (baseline) geometry of the (staged) vehicle. Hereto, the module implemented in SPADES allows for each stage to be defined separately in a way that allows for the generation of a 3D-model of each stage as well as of the total vehicle, see figure 2. Such a model can then be used for both evaluation purposes and in e.g. aerodynamic panel codes. Main items considered for each stage are fuselage, wing, tail surfaces, canard (if any), powerplant (airbreathing and/or rocket), and tanks. Most of these items can be subdivided into sections, which then allow for different materials and Thermal Protection Systems (TPS's) to be selected for each of these sections. In addition, the geometry module also allows for the calculation of several important geometry-dependent parameters, like wetted area, wing area, and maximum available volume.

Vehicle analysis

Presently, three modules are available for the calculation of the vehicle characteristics. These modules include an aerodynamics, a propulsion, and a structural (mass) module.

Aerodynamic module

The aerodynamic module facillitates the use of two-dimensional data tables for aerodynamic data, like the lift and drag coefficients. Also aerodynamic corrections for the take-off and landing configuration of the space plane are taken into account.

In addition, this module allows for the calculation of the heating of the vehicle based on a simple relationship, which can be determined for the average heat flux over a flat plate of 1m length and the 'actual' heat flux distribution over the vehicle at a chosen reference condition (heat flux distribution is assumed constant).

Aerodynamic data can be generated using two separate aerodynamic models, not yet incorporated in SPADES, including:

- an analytical model based on the well known USAF DATCOM method, and
- panel codes, like the NLR developed NLRAERO for sub- and supersonic flow and the in-house developed PEACH-code (Veldman; 1994) for hypersonic flow (M > 4 á 5).

The 'actual' heat flux distribution of the vehicle, at a chosen reference condition, is calculated using PEACH, see also the section on thermal protection structures.

Propulsion module

Like the aerodynamic module, the propulsion module facillitates the use of two-dimensional data tables for the different engines used. This allows for providing the thrust and (fuel) specific impulse as a function of dynamic pressure and flight Mach number. The module also allows for the effect of the angle of attack on precompression to be taken into account using relationships derived from the ESA WLC-studies [An.; 1991]. Powerplant performance has to be given for one fuel only. The effect of different fuels on the specific impulse (when considering dual-fuel engines) is taken into account by comparison of the heating value of the different fuels considered.

Propulsion data can be calculated using the in-house developed $(SP)^2S$ tool [Margadant; 1994] and [van Doorn; 1993]. This tool enables an estimation of the propulsive performance (thrust and specific impulse) for rocket engines, as well as for turbo- and ramjet engines. For the turbojet, off-design calculations must be implemented yet.

Structural module

The structural (mass) module allows for the estimation of the mass of each (sub-)system. It is based on simple statistical-analytical methods. Approximately 35 different (sub-) systems are taken into account including the thermal protection system (TPS) and some miscellaneous items like a pressurization system and an auxiliary power unit. A typical mass estimation is shown in table 1.

Mission simulation

The mission module of SPADES, basically, allows for the calculation of the required fuel or propellant mass and or separation conditions (in terms of range, Mach number, altitude and flight-path angle), field performance, maximum dynamic pressure, cruise speed and altitude, and fuel reserve (including boil-off). It features two methods of mission analysis.

The first method uses a pre-defined 'optimum' trajectory in terms of altitude, velocity and lift-to-weight ratio, e.g. determined using ASCENT, see the section on flight dynamics. A disadvantage of this approach is that it does not enable fast design iterations, since for each design iteration,

a new optimum trajectory should be investigated upon. However, as a first step, when considering only small design changes, the optimum trajectory could be considered as being constant.

In the second method, a simple built-in trajectory generation tool is used. Internal iterations guarantee that the mission specifications are met. Besides the calculation of the field performance, this method allows for an optimum climb phase optimized for minimum fuel consumption divided by the gain in energy height or:

$$\min \quad \int \frac{dW}{dh_e} dh_e$$

According to [van Buren and Mease; 1991], this approach leads to a completely optimal trajectory on the slow manifold (i.e. large time scale) when the so-called minimum principle is applied.

For winged vehicles, a cruise phase, at a specified altitude and Mach number, can be applied to meet the range requirement. The duration of this phase is iterated for. The well-known Brequet equation is used to determine the change in weight during this phase.

A pull-up phase is also available, allowing the separation conditions to be met, again through an iterative procedure.

Vehicle scaling

From the required fuel or propellant mass, calculated in the previous step, the required fuel/propellant volume for each stage is calculated and compared with the available fuel/propellant volume. In order to get the required volume in agreement with the available volume, the stage under consideration is scaled. For this, a scale factor λ is determined according to:

$$\lambda = \left(\frac{V_{fuel_{req}}}{V_{fuel_{av}}}\right)^{\frac{1}{3}}$$

Once this scale factor is determined, all stage dimensions are multiplied with the scale factor and vehicle analysis is repeated.

At this moment, because of the use of data tables, it is assumed as a first approximation, that the thrust scales quadratically with the scale parameter. The specific impulse is assumed constant. With respect to the drag, the effect of the constant size of the upper stage(s) on the total drag can be modeled in the following way:

$$C_D = \left(1 + \frac{C_{D_{2nd}}}{C_{D_{basic}}} \frac{1}{\lambda^2}\right) C_{D_{basic}}$$

For this, the relative influence of the upper stage(s) on the total drag of the baseline vehicle is known.

Hereafter, also the mission analysis is repeated. The scaling process continues until the required tankage volume equals the volume available in the vehicle (i.e. $\lambda = 1$), thereby taking into account limitations with respect to the size of the vehicle and with respect to field performance.

Costing

To determine the operating or life cycle cost of the vehicle considered, use can be made of the TRANSCOST model [Koelle; 1991]. This model, however, has not been implemented in the tool yet.

Optimization

SPADES allows for an optimization of the first stage scale size of a Sänger-like vehicle in relation to the powerplant scale size. Hence allowing for a first estimation of the level of interaction that exists between the powerplant and the space plane. Hereto, the scaled air-breathing powerplant drag is modelled in a way comparable to the second-stage drag, whereas for each change in powerplant size, the earlier mentioned scaling of the vehicle is repeated, i.e. a bi-dimensional optimization. The results found, show that the empty mass of the first stage (excluding powerplant) changes about 6% for a change in scale size of the powerplant from 1.0 to 0.8. It should be noted, however, that these values are likely to change, when for each vehicle a really optimum trajectory is taken into account.

In figure 3 some typical results are given for the take-off mass of a typical TSTO, winged, air-breathing space launcher as a function of the powerplant-to-vehicle scale size ratio at three distinct values of the fuel-to-total vehicle volume percentage, being respectively 65%, 55%, and 43.5%. Also indicated in the figure is the required field performance. Only the design solutions to the right of this limit remain within the specified field performance (3000m). From the figure, it can be concluded that [Quesada; 1994]:

- for each volume percentage, there is an optimum size ratio,
- the optimum size ratio differs with varying volume percentage, and
- the introduction of field performance as an

additional requirement can significantly affect the 'optimum' solution.

To end with, the tool SPADES is a very first step at the faculty LR to perform conceptual design studies of space launchers. For the near future, plans have been made, aiming at the development of an highly integrated, interactive computer tool, which allows the user to take all technical disciplines into account, including aerothermodynamics, propulsion, structures and materials, stability & control and services & equipment, and to make full use of (multi-dimensional) optimization techniques for vehicle design and flight performance optimization.

The development of SPADES has been partly funded by NIVR and has been carried out in close cooperation with FSS, SPE, NLR and PML under the heading of the AEOLUS programme.

3. Aerothermodynamics

In the field of aerothermodynamics, the research work is mainly directed to hypersonic aerodynamics for aerospace vehicles. The main aim is the modelling of intriguing physical aspects occurring in the hypersonic flow regime using experimental and numerical methods. Key problems, whose solutions will contribute significantly to the understanding of the aerodynamic characteristics of hypersonic vehicles, are identified and studied. The following research topics/projects are defined:

- Numerical simulation of high enthalpy non-equilibrium flows,
- Validation of CFD-codes for three-dimensional flow prediction around a blunt hemisphere-cylinder-flare re-entry configuration,
- Experimental hypersonic flow research.

Numerical simulation of aero-thermal effects in hypersonic re-entry flows.

The design of future aerospace planes and space launchers relies more and more on computational methods. The computational simulation is of prime importance because it is a necessary step for a subsequent extrapolation to free flight. Ground based facilities cannot simulate all aspects of the flow physics, e.g. viscosity, compressibility, chemical kinetics etc. that are involved. Moreover, the new high-enthalpy facilities do not only partially simulate the chemistry, but this partial simulation is far from complete due to the difference in degree of dissociation in the windtunnel compared to free flight. Therefore computational simulation is used to bridge between validated experiments in laboratory environment and free flight conditions.

As the use of inviscid methods become more and more routine, the accurate and efficient computation of viscous hypersonic flow with or even without real-gas effects is still a critical point in Computational Fluid Dynamics.
Hypersonic viscous flows are mostly characterized by strong shocks producing large pressure jumps and the appearance of regions with almost vacuum conditions due to strong expansions. To cover these aspects one needs robust flow solvers featuring various dissipation rates but still allowing the accurate resolution of viscous flow phenomena. Within this respect CFD-codes for quasi one-dimensional and for two-dimensional viscous flows are under development at the faculty LR.

For quasi one-dimensional nozzle flows a code incorportating chemical and vibrational multi-temperature non-equilibrium has been developed [Walpot; 1991]. The code is based on a van Leer scheme and ENO (Essentially Non-Oscillating) interpolation for improved accuracy. The viscous effects are modelled in a classical way using boundary layer theory. The development of the boundary layer thickness is approximated by the Edenfield method [Edenfield; 1968].

As an example the flow of Nitrogen (N_2) in the HEG (High Enthalpy Götingen) nozzle is calculated for a reservoir pressure of 400 bar and a stagnation enthalpy of 20 MJ/kg. As a typical result figure 4 shows the centre-line distribution of the static and vibrational temperatures together with the growth of the boundary layer thickness and its displacement thickness. Note the rapid freezing of the vibrational temperatures, the quite thick boundary layers and the velocity which quickly reaches constant values.

For two-dimensional flows a Navier-Stokes code, equipped with a flux-difference splitting solver, is under development. For perfect gas applications the Navier-Stokes-code is now in the validation phase. Several test cases, e.g. hypersonic laminar ramp flow, the viscous flow along a flat plate and the impingement of an oblique shockwave on a boundary layer are calculated and compared with experimental data; for more details see [Walpot, Schwane and Bakker; 1994].

Recently we have investigated the impinge-

ment of an oblique shock on a bow shock of a cylinder. The flow parameters are chosen such that the interaction is classified as an Edney-interaction of type IV. As an example in figure 5 the experimental and computed interferograms displaying the density distribution in the interaction zone are shown. Although the shock locations look quite similar the interferogram does not display the supersonic embedded jet flow so typical for the type IV Edney-interaction.

Validation of three-dimensional flow modelling using a blunt hemisphere-cylinder flare re-entry configuration.

Interference effects in the high supersonic/hypersonic flow field past (re-)entry bodies are known to cause local areas of enhanced surface heat flux in the vicinity of surface discontinuities such as cockpit/body junctions etc. The enhancements, which can give a significant increase of heat flux, are attributed to shock-shock and/or shock-boundary layer interactions causing separation and reattachment of the flow. Shock interaction is frequently a stimulus for boundary layer transition bringing the boundary layer in a turbulent state and increasing the local heat flux in a considerable manner. The prediction of the complex three-dimensional flow field in such regions is a challenging task for numerical models [Horstman et al.; 1986]. In order to validate such CFD methods there is a need for good quality experimental data. Hereto an experimental program is being carried out on a blunt hemisphere-cylinder with a conical flare afterbody.

Figure 6 shows a Schlieren picture with a clear view on the model geometry and the interaction of the bow- and flare shock. The tests are directed towards the leeward side and the windward side flow properties. At the leeward side the flow is complex due to the appearance of vortical flow, large separated regions, embedded shocks and reattachment zones. The windward part is less complicated apart from an interesting shock-shock interaction of various Edney type occurring at high angles of attack.

The experimental program contains a flow visualization study (spark shadowgraph, surface oil flow, colour schlieren), detailed surface pressure measurements and a quantitave exploration of the flow field with holographic interferometry. The experiments are made in the TU-Delft/LR high speed wind tunnel (test section: 27cmx28cm). The tests are performed at free stream Mach numbers of 3 and 4 and at angles of attack from 0 to 20 deg. The Reynolds number based on the model length varies from 6 to 7.6 million. Preliminary results have already been obtained; a typical example is shown in figure 7 displaying separation zones, attachment regions and formation of vortices in the leeward side surface flow topology at Mach number M = 3.

In figure 8 some interferometer results are shown for the axisymmetric case. In this figure the measured phase map of the flow field is compared with a computed phase map resulting from numerical simulation of the holographic experiment using a 3D Euler code. The fringes in the phase map represent lines of constant integrated density. The agreement between experiment and theory is encouraging for performing similar experiments for the non-axisymmetric case. However, for non-axisymmetric flows the determination of the local density is not a routine job, since it has to be deduced by tomographic reconstruction using an appropriate set of various phase maps taken in different directions. Research in this field is taken up in close cooperation with the University of New Hampshire.

Experimental hypersonic flow research.

Flow research at Mach numbers in excess of about 4 at the faculty LR is restricted mainly to computational studies. However, there is an urgent need to also have an experimental capability, mainly with respect to validation and the development of scaling laws. Within this respect, a simple, low cost, Hypersonic Test Facility (HTF-Delft), allowing for experimental hypersonic flow research in the Mach number range from 6-11, will be set up in Delft. This facility will become operational in 1995 and provides the faculty with an unique experimental capability. In addition, because of the simple, low cost features, this facility also can be used for educational purposes.

The HTF-Delft will be based on the 'Ludwieg Tube' principle: test gas stored at a high pressure in a long supply tube is released through a supersonic/hypersonic nozzle by means of a fast operating valve in front of the nozzle. Tube pressures are maximum 200 bar, whereas part of the supply-tube is pre-heated to 1000 K to allow heating of the test gas. Downstream of the nozzle the gas flows through a diffusor into a vacuum vessel. A schematic view of the whole test-arrangement is given in figure 9. The test section has a diameter of 35 cm and a length of about 30 cm. The Mach number ranges from 6 to 11. The Reynolds number based on the test sec-

tion diameter varies from 3.1×10^6 to 8.1×10^6. The running time is about 100 msec. At this moment, the desired facility is under development at the research institute Hyperschall Technology Göttingen (HTG) in Germany.

The instrumentation of this test facility will consist of flow visualization techniques, like Schlieren, shadowgraph and interferometry, force measurement instrumentation, hot film and liquid crystal measurement techniques, high frequency pressure transducers, digital holographic interferometry, pressure sensitive paints.

Planned research for the HTF-Delft are fundamental studies on heat transfer and skin friction in flow situations such as laminar and turbulent boundary layers, transition, and viscous interactions of boundary layer flow with external hypersonic flow fields. The fundamental results obtained with the Ludwieg tube are meant to improve the physical flow modelling, which in turn can contribute to the development of better transfer models (scaling laws, computational methods) for the extrapolation of windtunnel results to free flight. Parts of this research will be carried out in cooperation with ESTEC.

4. Flight dynamics

The Flight Dynamics Group of LR is looking into the area of trajectory optimization and guidance and control for hypersonic trans-atmospheric vehicles. Basically 4 topics are being addressed, including trajectory generation and optimization, controller design and development, joint state and aerodynamic parameter estimation and manual flight handling qualities. These topics will be discussed in more detail below.

Flight simulation.

An important part of the activities is directed to the development of software tools for the generation of optimized ascent and descent trajectories of space planes and for the evaluation of the flight qualities of the vehicle under all possible flying conditions.

An important activity in this field is the development of the FORTRAN-coded simulation tool ASCENT [Korswagen; 1993]. This tool allows for the simulation of the motion of the vehicle as the motion of a point mass, under influence of aerodynamic lift- and drag forces, thrust forces and gravity forces. The propulsion and aerodynamic data are included as two-dimensional tables. The Earth is modelled as a flattened sphere, thereby taking into account the J_2 harmonic to describe the gravitational field. The atmosphere used can be either the US-76 atmosphere or the exponential atmosphere. Wind effects are not considered yet. To describe the trajectory of a trans-atmospheric vehicle, the complete trajectory is divided into flight segments. For each flight segment, simple guidance laws are available. Typical such guidance laws are a constant normal load factor, a constant flight-path angle and a constant flight path angle at constant speed. Mission constraints, like the dynamic pressure and the axial acceleration can also be applied.

To date, ASCENT has been used to compute a 'near' optimum trajectory for the TSTO space plane, which is used as the reference vehicle for the AEOLUS studies of the Dutch industrial platform on advanced launcher developments [Marée et al.; 1994]. A typical result for the velocity-altitude relationship of this vehicle using an airbreathing first stage and a rocket-propelled second stage is given in figure 10. Other outputs of the program include amongst others flight path angle, climb speed, range, thrust, angle of attack and the mass of the vehicle, all as a function of time and/or altitude.

ASCENT has also been used to perform a sensitivity analysis to get insight in the importance of trajectory optimization and in the parameters of major influence to the optimum performance of a typical space plane [Marée et al.; 1994]. In figure 11 some typical results are given, illustrating that this tool can very well be used to determine the sensitivity of space plane performance to small variations in its trajectory. It also clearly shows the necessity of trajectory optimization.

Future developments planned for ASCENT are to also investigate the effect of the drag increment associated with the use of the space plane controls, like the elevons, etc. As a first step, it will be assumed that, at each instant of time, the vehicle is in rotational equilibrium. For example, the pitching moment is set to be zero. This leads to an algebraic equation, from which the deflection of the elevons to maintain equilibrium for a given angle of attack can be calculated. Next, also the associated increments in lift and drag can be calculated as well as their influence on the performance of the space plane.

Another important item is trajectory optimization. Presently, two methods are being investigated for use with ASCENT, being the well-known collocation method and a weak Hamiltonian finite element method, developed by

Hodges and Bless [Hodges and Bless; 1992].

A further major activity taking place is the development of a six degrees-of-freedom (DOF) flight dynamics software tool [Mooij; 1994]. This tool is based on the re-entry simulation tool START, which, for instance, has been used to evaluate the re-entry of the Huygens probe in the Titan atmosphere [Mooij; 1992]. Development extensions of START are mainly related to variable mass properties, guidance and control, and propulsion including reaction control. Since also provisions will be made for control modelling, the effects of variations of controller parameters on the resulting trajectory may be investigated. No specific attention is given in START to controller optimization, because this will probably lead to a very high computer load. Therefore, the improved START will mainly be used to investigate the flying qualities of the launcher and to determine the effects of 6-DOF compared to 3-DOF.

Controller design and development.

During the flight of a space plane through the atmosphere, its dynamical characteristics tend to vary over a much wider range than for ordinary aeroplanes. The space plane guidance and control system has to be designed such, that it can cope with these wider range of variations. Here, the guidance system is associated to the large scale control of the vehicle along a predefined trajectory, whereas the control system is related to the smaller scale motions of the vehicle.

The design of a guidance and control system, generally, involves the repetitive flight simulation of the vehicle to be controlled. In the conceptual phase of the development a number of relatively simple controller types is adopted and tested using relatively simple vehicle models. Based on these tests, the most promising concepts are selected, developed further and again tested, but then with some more complete vehicle models. Finally, the resulting control system is selected and evaluated extensively using the most accurate and thus most complex vehicle model.

To facilitate the design and development of a controller for a space plane, an overview of applicable guidance strategies has been made [Marée; 1992] and [Mooij, 1994]. In addition, several mathematical models of various complexity are being developed at the faculty LR. So far, a first model, called MASPSYM, has been completed and is currently being tested. It enables the design of a controller for the symmetric motion of a winged hypersonic trans-atmospheric vehicle flying over a flat earth. For the determination of the aerodynamic coefficients use can be made of three-dimensional tables in combination with linear interpolation. Recently, though, also a method using a two-dimensional polynomial combined with one-dimensional linear interpolation has become available. Planned extensions for MASPSYM include the modelling of a spherical earth and earth rotation. In a parallel development the model will be extended to include asymmetric motions (MASP).

To enable quick changes of both the vehicle and the controller model. This model, called MASPSYM, is being developed using MATLAB and its extension Simulink as the (graphical) working environment. This working environment facilitates a good insight in the modelling and enables easy modifications. However, a major drawback might be the rather long computer time to perform a simulation run. As a solution for this problem, the earlier mentioned START model could be used, although then changing the vehicle controller will take some more time.

Aerodynamic parameter estimation.

Presently, the subject group for "Aerospace Vehicle Stability and Control" of the faculty is investigating the application to space planes of the in-house developed two-step-method for (aerodynamic) parameter estimation using (simulated) dynamic flight test data. This method has already been applied successfully in many aeroplane flight test programs, but in principle should also be applicable to space planes. The method is, basically, as follows. To estimate the aerodynamic parameters, a joint state and parameter estimation problem must be solved. This problem can be solved in two consecutive steps. In the first step the flight path is reconstructed using state estimation techniques. In this step also biases and scale factors in the instrumentation can be estimated simultaneously. As the second step, the linear and angular accelerations are used to determine the aerodynamic forces and moments and also the aerodynamic coefficients.

As a first step, [Chu et al.; 1994] have investigated the advantages of flight path reconstruction of a space plane using a nonlinear recursive identification method (adaptive filtering) compared to the use of the Extended Kalman Filtering (EKF). Data used as input consists of data provided on position and velocity and information obtained from accelerometers and rate gyros with additional constant biases and white measurement noise. Figure 12 presents some typical results for an aeroplane (a) and a space plane (b) using both the nonlinear recursive prediction method and EKF. Data used as input

consisted of air-data sensors (aero plane) or GPS provided data about position and velocity (space plane) and information obtained from accelerometers and rate gyros with additional constant biases and white measurement noise. From the figure, it is found that the adaptive filtering gives a smaller standard deviation (STD) and hence better accuracy. Also the error is found to improve for adaptive filtering. This effect is, however, larger for the aeroplane then for the space plane. This is attributed to the highly non-linear system, which results due to the transformation of the GPS-provided data to the navigation frame. Improvement is sought by taking into account the coupling, which exists between state estimation and parameter estimation and which is not yet accounted for in the adaptive filtering method [Verbaas; 1994].

Manual flight handling qualities.
An important issue, when flying a space plane manually, is the handling qualities of the space plane. To allow investigation of the handling qualities of such space planes, usually extensive flight testing or flight simulation in a hardware flight simulator are required. The subject group for "Aerospace Vehicle Stability and Control" has investigated the requirements to be fulfilled for the presently developed six degrees of motion simulator (SRS) of the SImulation, MOtion and NAvigation "SIMONA" institute of the Delft University of Technology to allow such evaluations [Advani and Kramer; 1994]. From this investigation, it has been concluded that the SRS provides a good environment for the full-flight simulation of space planes. Sofar, however, no such plans have been developed yet.

5. Structures

In the structures field, LR is examining shell design procedures aiming at improved design criteria for shell stability. This research is, amongst others, of importance of the design of the curved stiffened shells for, for example, launcher stages, the Ariane 5 Bati moteur, and space plane fuselages. In addition, numerical methods are being developed for the design of thermal protection systems of space planes.

Shell stability research
The goal of the shell stability research being carried out at the faculty LR is the development of a shell design procedure with improved design criteria in which the latest available knowledge is incorporated and which makes efficient use of the currently available interactive and (super) computing facilities [Arbocz; 1993].

It is a well known fact that there are great differences between the theoretical buckling load and the experimental buckling loads measured on axially compressed cylindrical shells. Initial imperfections in the shell geometry have been accepted as the main cause of the wide experimental scatter. Despite this recognition the incorporation of the idea of imperfection sensitivity into engineering practice has not been accomplished. Until very recently design manuals adhered to the so-called Lower Bound Design Philosophy, i.e. they recommend the use of an empirical knockdown factor, which is so chosen that when it is multiplied with the classical buckling load a Lower Bound to all available experimental data is obtained.

$$P_a \leq \frac{\lambda}{f_s} P_{cl}$$

where P_a is the allowable applied load, P_{cl} is the classical buckling load (perfect shell) given by:

$$P_{cl} = \frac{2\pi E t^3}{\sqrt{3(1-\nu)}}$$

λ is the knockdown factor:

$$\lambda = 1 - 0.902(1 - e^{-\frac{1}{16}\sqrt{R/t}})$$

f_s is the safety factor, E is Young's modulus, t is thickness, ν is Poisson ratio, and R is radius.

For isotropic shells the effect of the knockdown factor and the wide experimental scatter is shown in figure 13.

The improvements in currently recommended shell design procedures are primarily sought in a more selective approach by the definition of the knockdown factor. Thus, for instance if a company takes great care in producing its shells very accurately then under certain restrictions the use of an improved (higher) knockdown factor λ_a derived by a stochastic approach should be allowed. Therefore, it is necessary to determine a reliability function for a group of shells produced by the same process, using e.g. the First Order Second Moment Method, figure 14. Thus shells produced by a process characterized by a less damaging initial imperfection distribution are allowed to be designed for higher load intensities, see also figure 13.

The key to the success of a stochastic approach of stability analysis lies in the reliability and accuracy of the underlying deterministic buckling analysis used. To this end, at the faculty LR, several non linear computer codes have been developed and explored to determine the behaviour of imperfect shell structures. They incorporate imperfection sensitivity analysis of increasing complexity.

Some years ago the Delft Interactive Shell Design Code DISDECO was introduced facilitating by the improved shell design procedures offering access to the above mentioned analysis codes as well as the accumulated theoretical, numerical and practical knowledge in the design of imperfection sensitive thin-walled shell structures. DISDECO can be characterized as open-ended, hierarchical, modular and interactive. Today the program is fully operational and in the Structures and Materials group of the faculty subject of continuous improvements and expansions. In addition the Structures and Materials Laboratory offers full possibilities of complete surface scans of shells using modern measuring and data acquisition systems and data reduction analysis to determine the necessary reliability functions.

To conclude with, the faculty is of the opinion that DISDECO already provides a workable solution to achieve a substantial improvement in the design of shell structures of advanced space launchers.

Thermo-structural research

Recently, a research program has been defined aiming at the design of improved TPSs. Within this program attention is given to first the development of a structural model for the design of passive TPSs, thereby taking into account time dependent heat loading, and second the use of Fibre Metal Laminates (FMLs) as a fatigue-insensitive, structural material for high temperature applications (see section on materials).

Presently, a very first model for the prediction of the TPS mass of a space plane is currently available. This model uses the in-house developed PEACH-code [Veldman; 1994] to predict the surface temperature and heat flux distribution over the vehicle. Next, a database on TPSs is used to determine, for specific parts of the surface, the type of TPS to be applied and also the thickness (as a function of heat flux and surface temperature). Finally, the TPS mass is calculated.

To calculate the heat flux and the surface temperature, firstly a geometrically complex vehicle is divided into a number of plane surfaces in a manner which adequately approximates the real shape, see figure 15. Secondly, for each panel, the surface equilibrium temperature is calculated based on the steady state heat balance between the boundary layer convection to the surface and (grey body) surface radiation (view factor assumed equal to 1) to space. Convection is approximated using the modified Reynolds analogy:

$$St = \frac{c_f}{2} \cdot Pr^{-\frac{2}{3}}$$

where St is the stanton number, c_f the local skin friction coefficient and Pr the prandtl number. Skin friction is calculated based on flat plate incompressible skin friction, which is corrected for compressibility effects using standard available methods, including the reference enthalpy method. In addition to the calculation of the surface equilibrium temperature, also an assumed temperature distribution can be used or the adiabatic wall temperature can be calculated. Typical calculated wall temperatures and heatfluxes are shown in figure 16.

A second topic being investigated is the thermo-structural design of the movable ramps of the inlet of an airbreathing space plane engine (carried out jointly with Stork Product Engineering). Hereto, use is made of ANSYS, thereby taking into account the variable heat loading of the ramp, due to the changing flight conditions during the mission. In figure 17, a typical ramp design, designed for a maximum temperature of 1600K and a stiffness in excess of 2×10^7 N/m, is given. It consists of a titanium honeycomb configuration with metal facings, an insulating material and a C/SiC outer skin. For this structure, use is made of, presently being developed, ceramic adhesives (see also the section on materials) capable of handling temperatures up to 1600K. The dimensions of the ramp are approximately 2m wide and 2.5m long. The mass of the ramp is approximately 15 kg/m^2.

In addition, the faculty has plans to develop a thermal test setup for experimental research of TPSs.

6. Materials

Future materials to be used for airplane-like space launchers and high speed airplanes have to withstand not only fatigue loading, but also high temperatures. For example, the High Speed Civil Transport Aircraft (HSCT), which flies at speeds up to Mach 3, will be heated up to 580K [Drexel; 1993], whereas the first stage German Sänger space plane, flying at speeds up to about Mach 7, will be heated up to about 800-1300K [Grallert and Volmer; 1993]. These loads then require the use of fatigue insensitive, high-temperature resistant, materials.

Candidate fatigue insensitive materials, like the traditional composites, are limited due to fabrication problems, temperature limitations and damage tolerance, whereas high-temperature resistant materials, like titanium alloys, are sensitive to fatigue. To combine the advantages of titanium and composites FMLs have been developed at the faculty LR. This material originated in the 1980's when it was developed at the structures and materials laboratory of the faculty LR. FMLs, are thermally anisotropic, hybrid materials, basically consisting of alternating layers of thin metal sheets (aluminum alloys or titanium) and composites (glass, carbon or aramide), see figure 18. FMLs are considered very attractive for structural applications in space planes and high speed transportation vehicles, offering significant weight savings and a longer service life. However, presently available FMLs, like ARALL and Glare, are limited for structural applications to approximately 400K, due to the use of aluminium alloys (2024 and 7075/7475) and a 380-400K cure epoxy.

In order to use FML as a structural material at elevated temperatures, the maximum allowable temperature is extended to the higher temperature range. To this end, the following topics are studied at the structures and materials laboratory of the faculty LR [Medenblik; 1994] and [Mensink; 1994]:

- New constituents to replace the currently used aluminum alloys and adhesive systems. The new laminates can be used in the temperature range upto 500 - 600K. Typical materials studied include Al-Fe alloys, Ti-6Al-4V, commercially pure (CP) titanium, PEEK-carbon (APC-2) - AS4 pre-preg, phenolic triazine (PT) - IM600 carbon and PT-glass pre-pregs (in cooperation with Drexel University, Philadelphia for the US Navy).
- The use of ceramic bonding materials to extend the use of FMLs well in excess of 600K. Organic resins limit the maximum service temperature to 600K therefore ceramics resins have to be used at higher temperatures.
- As a first step for launcher applications attention will be given to the manufacturing of stiffened compression loaded Glare panels. Future plans are to perform these compression tests as well on titanium-carbon panels, if possible at elevated temperatures.
- Optimisation of titanium surface pretreatment for high temperature bonding of titanium alloys.

In addition to the development of FMLs for high-temperature applications, also some efforts are given to investigate the use of FMLs as a structural material for e.g. cryogenic storage tanks.

The material properties characterization of FMLs includes:

- Adhesive properties of the laminates including lap shear and peel strengths.
- Tensile and fatigue properties at both elevated and cryogenic temperatures.
- Blunt notch properties.
- Residual strength testing.
- Coefficient of thermal expansion (CTE). A theoretical model for the prediction of the CTE has been developed by [Mensink; 1994]. From this model, it follows that the CTE of a FML is lower than the CTE of the metals they consist of, in those particular directions where fibres are oriented, due to the generally lower CTE of the fibres. Tests on the CTE of GLARE and ARALL have shown good agreement with the model results indicating that the used model is valid and therefore can be used for other FML as well.
- Fire resistance. Burn through tests, performed by Boeing on GLARE specimen, have generated ideas for use of FML in Thermal Protection Systems for space vehicles. A standard burn-through test, for screening ablative materials by means of an oxy-acetylene test, is performed on GLARE and ARALL and their constituents. Due to the anisotropic thermal conduction behaviour, high melt temperature and carbonizing effects of the fibres and matrix, the burn-through resistance of FML is better than for composites and metals separately.

Some material properties of high temperature

FMLs, GLARE and ARALL are:

1. A lower density than the metals (for Ti-6-4-APC2: 3045 kg/m^3 instead of 4432 kg/m^3 for titanium) combined with superior fatigue properties, see figure 19 for Ti-APC-2 laminates (determined at a maximum-to-minimum load ratio of 0.05) and good mechanical properties, see table 2.
2. Anisotropic thermal conductivity (see also burn through properties). As a rule of thumb one can use for the conductivity through the laminate sheet twice the conductivity of the composite (if volume % of the composite is about 50%). This results in a relative low conductivity (several W/mK). In plane the thermal conductivity is about half the conductivity of the metal (70 W/mK for Al-based laminates and 5 - 10 W/mK for Ti-based laminates). An advantage of this anisotropic behaviour can be a reduction of the thermal insulation mass for a structure, see also burn through properties.
3. Damage tolerance properties (impact, residual strength, fatigue) are much better than "normal" composites and monolithic metals [Vlot and Fredell; 1993].
4. Metal like manufacturing possibilities, e.g. deformation of a cured laminate, are possible [Sinke; 1993].
5. Good burn-through properties, see table 3. An explanation of the good behaviour can be given by using the anisotropic heat conductivity of the material. When there is a hot spot on an FML sheet the metal layer at the outside conducts the heat over the surface but the insulating composite will prevent rapid heating through the thickness of the sheet. These much better burn-through properties of FML compared to the monolithic material, will allow designers to use lower safety factors on the thermal insulation. During crashes, which result in a fire, occupants of the vehicle will have about a 10 times longer escape time when compared to aluminum.
6. Tensile tests on GLARE 4 at temperatures between -150 °C (about 125K) and +150 °C (about 525K) show an increasing modulus, strength and strain (no brittle behaviour), while the yield strength remains relative constant. This behaviour can be predicted and explained by the properties of the constituents and by the internal stresses in the material. The internal stresses can be calculated using the CTE model.

In conclusion, FMLs can offer interesting properties because of the combination of metal and composite properties.

The (damage) tolerant behaviour of the material will allow designers to use lower safety factors for some applications.

At elevated temperatures FML can offer a combination of attractive metal properties combined with high specific mechanical properties of composites.

Some thermal properties (burn through and thermal conduction) are quite unique because of the extreme anisotropic thermal behaviour.

7. Closing statement

At Delft University of Technology, Faculty of Aerospace Engineering, several activities are taking place, which serve to achieve a basic understanding of the principal problems associated with the design, development and production of space planes, and hence allow the development of an appropriate training of our future engineers. Important aspects to be considered for this training seem to be the high cost of development of space planes, typically, at least a factor 2 higher than for, e.g. Ariane 5, which makes it necessary to further increase international cooperation in the already highly international space society. Another important aspect seems to be to promote/educate a truly integrated approach towards the design, development and production of such advanced launchers instead of the traditionally used system approach. Otherwise, cost overruns will be almost inevitable, with the danger of killing any future space plane development project.

For further information on the advanced space launcher research carried out at the faculty LR, please contact B.T.C. Zandbergen. Fax. (31-15)783444, The Netherlands.

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Tables and figures

Fuselage items		Fuel system	
basic shell	28.6	fuel	120.0
crew	0.48	tanks	11.4
Wing items		Equipment	
total	45.8	controls	3.73
basic wing	36.5	active cg-control	0.44
strake	1.21	electrical system	3.17
flaps	1.85	avionics	5.07
slats	1.93	pressurization	2.69
leading edge	3.41	auxiliary power	1.02
trailing edge	0.95	furnishing & paint	1.16
Vertical tail	9.62	Second stage	100.0
Powerplant items		Landing gear	
total	41.7	nose gear	2.11
ramps	6.08	main gear	10.5
duct	3.52		
turbojets	18.6	Thermal Protection System	20.6
ramjets	2.88		
nozzle	9.13		
cooling system	1.50		
Total		Mass ratios	
take-off mass	408.3	m_e/m_{to}	46.1%
empty mass	288.3	m_{fuel}/m_{to}	29.4%
first stage	308.3	m_{wing}/m_{to}	11.2%
first stage, empty	188.3	m_{pay}/m_{to}	24.5%
first stage, struct	94.0	m_{pp}/m_e	22.2%

Table 1: Typical mass estimation for the first stage of a TSTO vehicle (in tons)

Properties	dir.	Test temp.			
		-150°C	25°C	150°C	280°C
UTS	L	1185	1180	1124	747
(MPa)	LT	917	682	548	452
TYS	L	1040	907	720	615
(MPa)	LT	840	618	510	425
E	L	104	109	111	97
(GPa)	LT	87	66	72	67
Elongation	L	1.9	1.9	2	1.6
(%)	LT	1.1	0.9	0.8	1.3

Table 2: Ultimate tensile strength (UTS), tensile yield strength (TYS), Young's modulus (E) and elongation of Ti 6-4/APC-2, 3/2 UD (3 metal layers/2 layers of composite; Uni-directional) for two directions (L and LT).

type of material	burn-through tests	flame leaking (s)	burn-through (s)
FML	GLARE 2; 3/2, 0.3 (1.4 mm)	83	133
	GLARE 3; 3/2, 0.2 (1.1 mm)	80	120
	GLARE 4; 3/2, 0.3 (1.6 mm)	185	185
	ARALL 3; 3/2, 0.3 (1.3 mm)	84	125
metal	2024-T3, (1.3 mm)	8	8
composite	Glass/epoxy (1.09 mm)	38	116
	Aramid/epoxy (1.13 mm)	13	13

Table 3: Some burn-through properties of FML; flame temperature 2500˚C. heat input: 560 W/cm^2

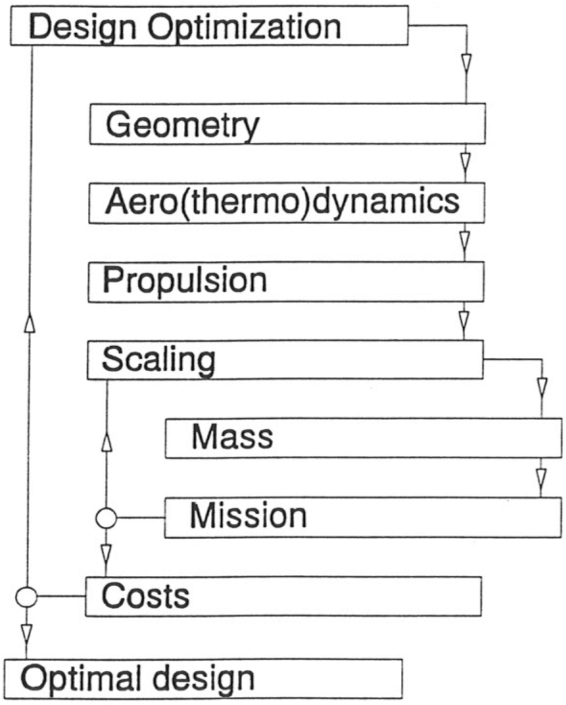


Figure 1: *SPADES design approach*

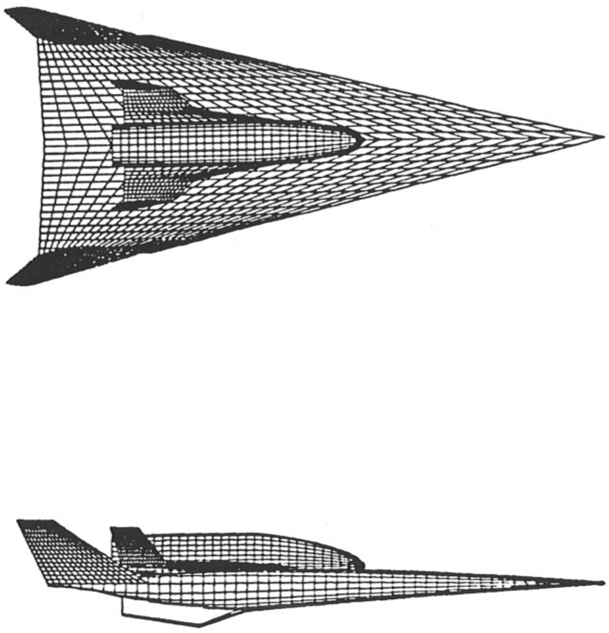


Figure 2: *Typical 3D-representation generated with SPADES*

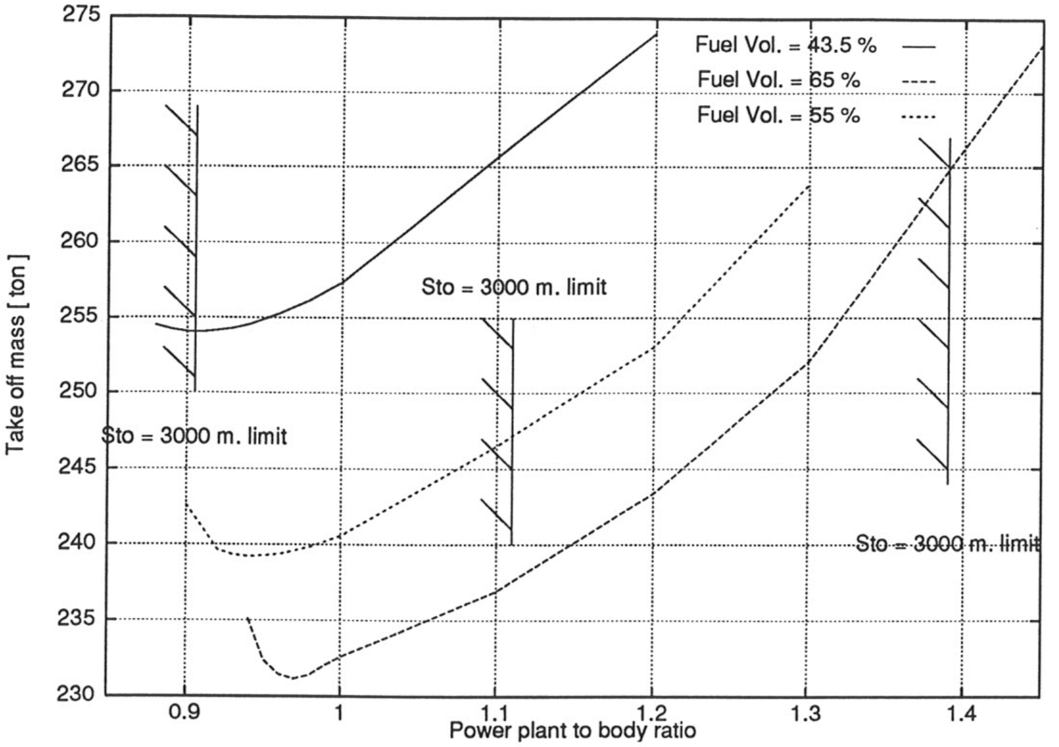


Figure 3: *Take-off mass as a function of powerplant to vehicle scale size ratio for a typical TSTO vehicle at three different values of fuel-to-total vehicle volume percentages*

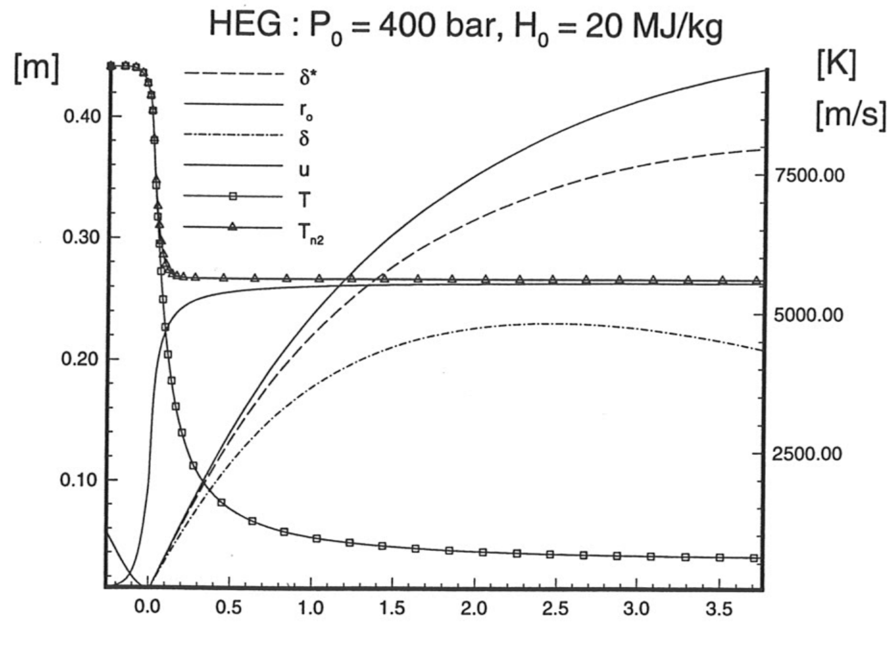
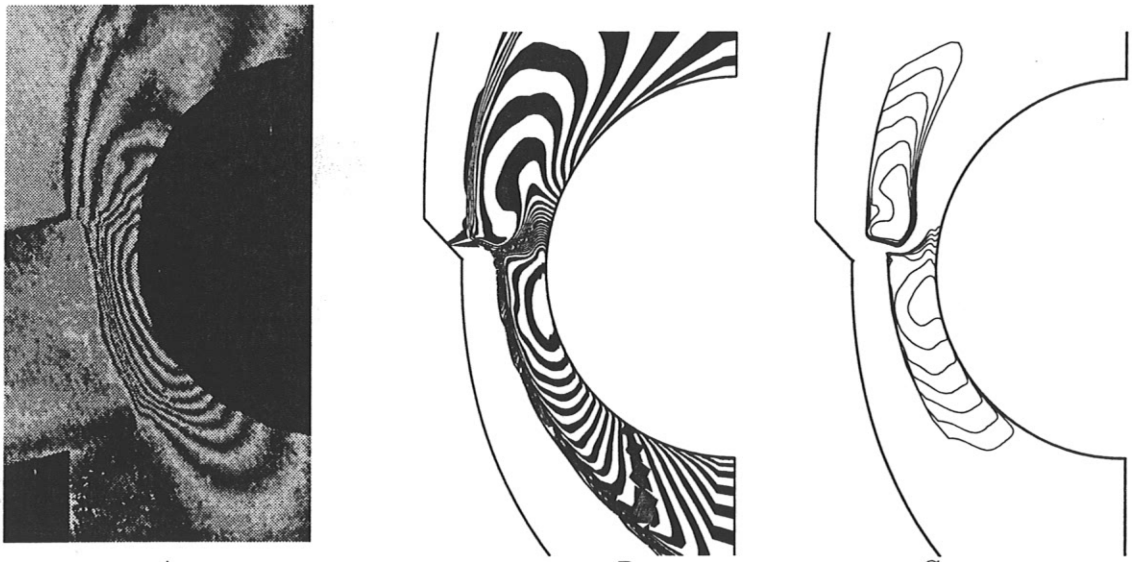


Figure 4: *1 D high enthalpy nozzle simulation of HEG*



A B C

Figure 5: *HEG simulation: experimental (A) and computed (B) interferogram, and iso-mach lines between 0 and 1*

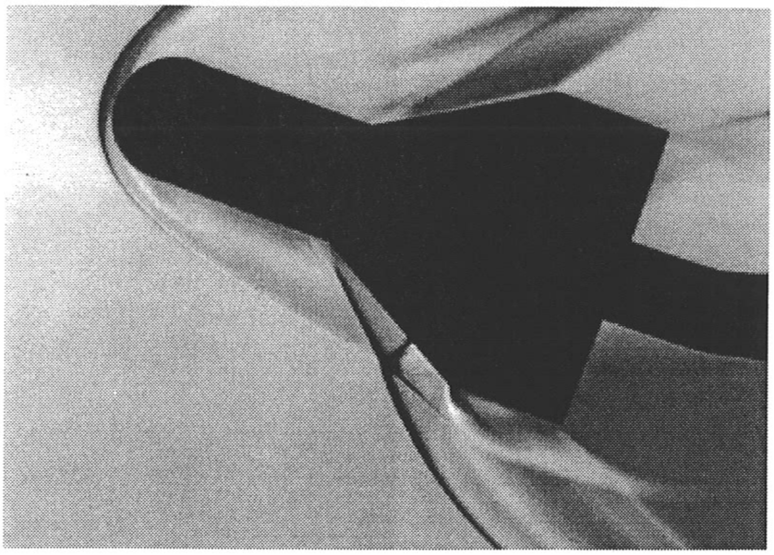


Figure 6: *Schlieren photograph showing model geometry and typical wave pattern*

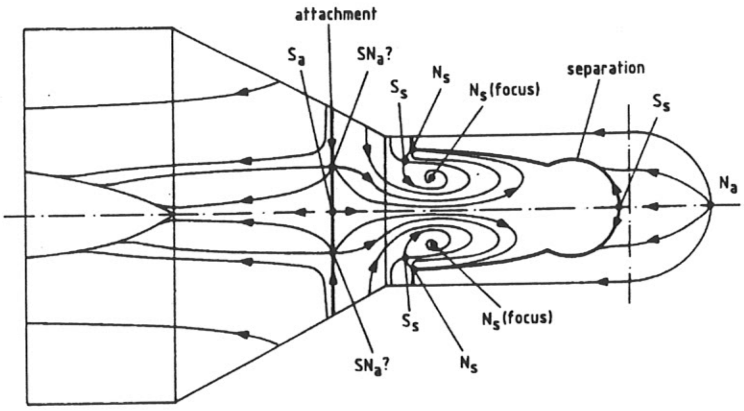
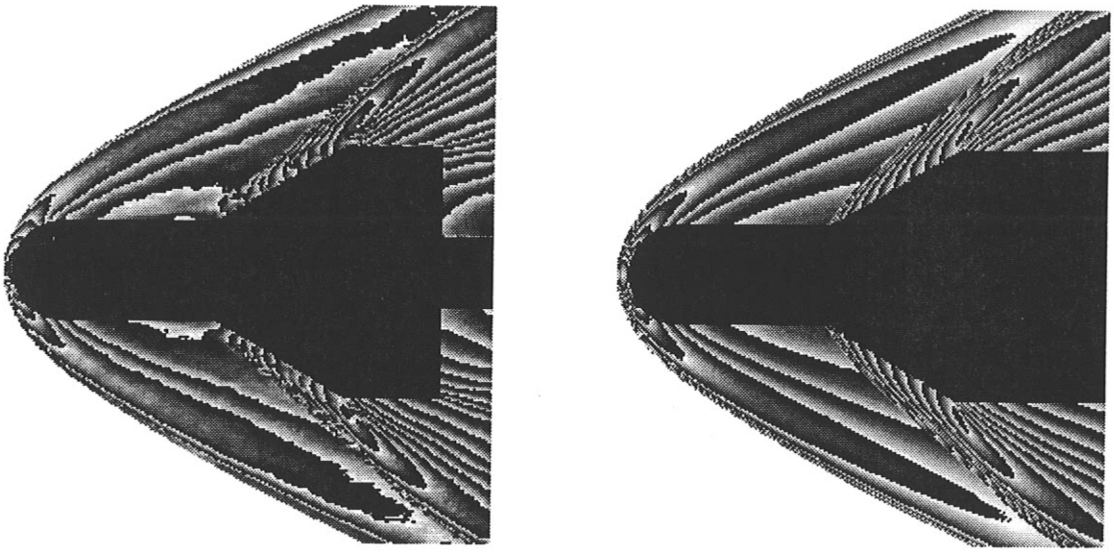


Figure 7: *Leeward side surface topology at M_∞ = 3 for α = 10°*



A B

Figure 8: *Measured (A) and computed (B) phase map at M_∞ = 3 for axisymmetric case*

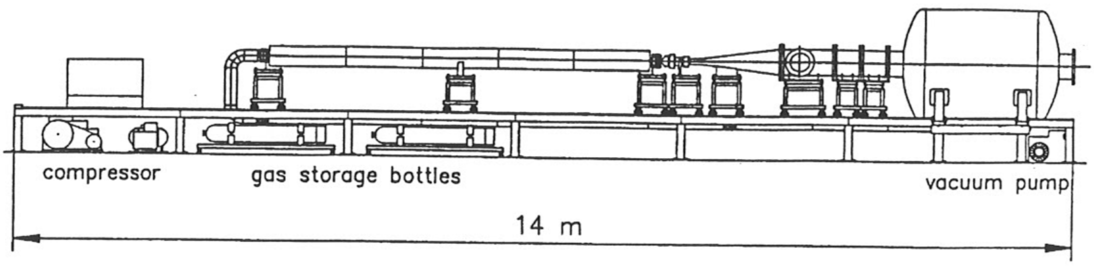


Figure 9: *Sketch of HTF-Delft*

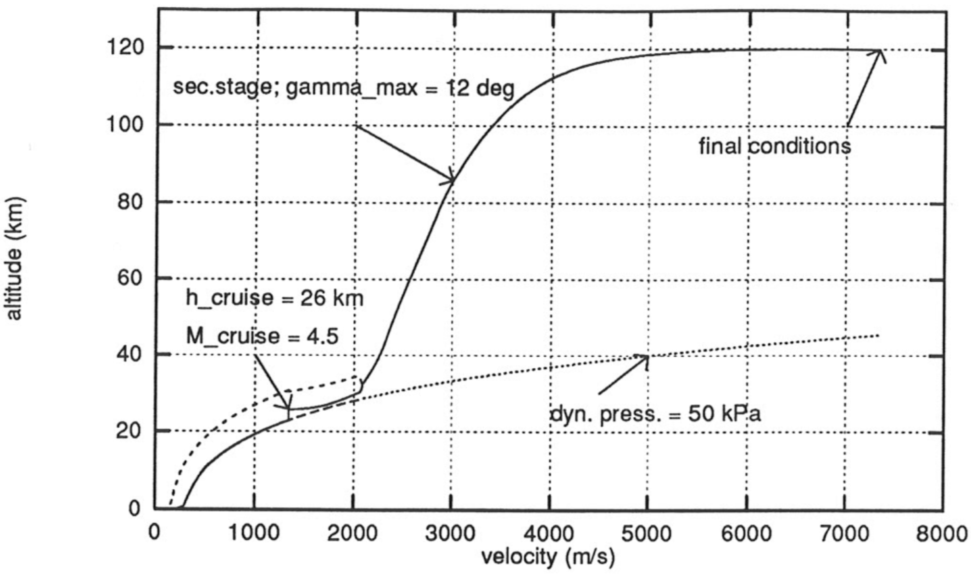


Figure 10: *Typical ascent trajectory of a TSTO space plane*

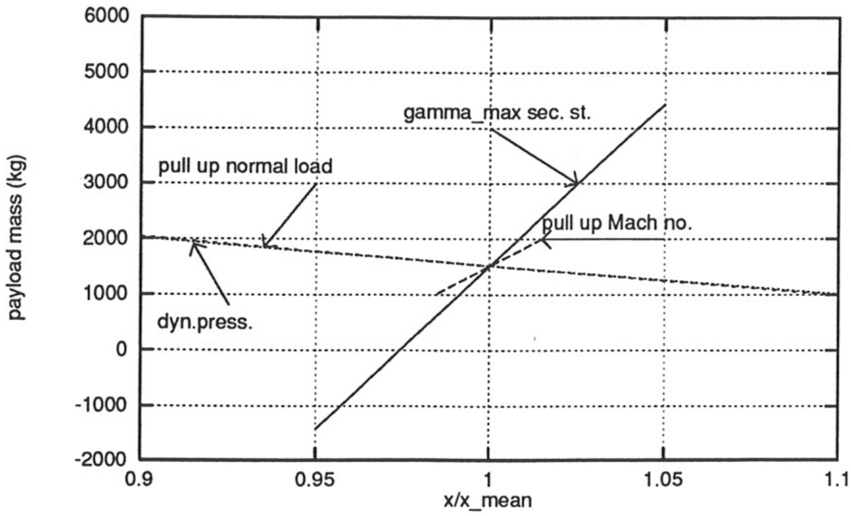


Figure 11: *Effect of first stage pull-up mach number, pull up normal load and dynamic pressure and second stage maximum flight path angle on payload mass.*

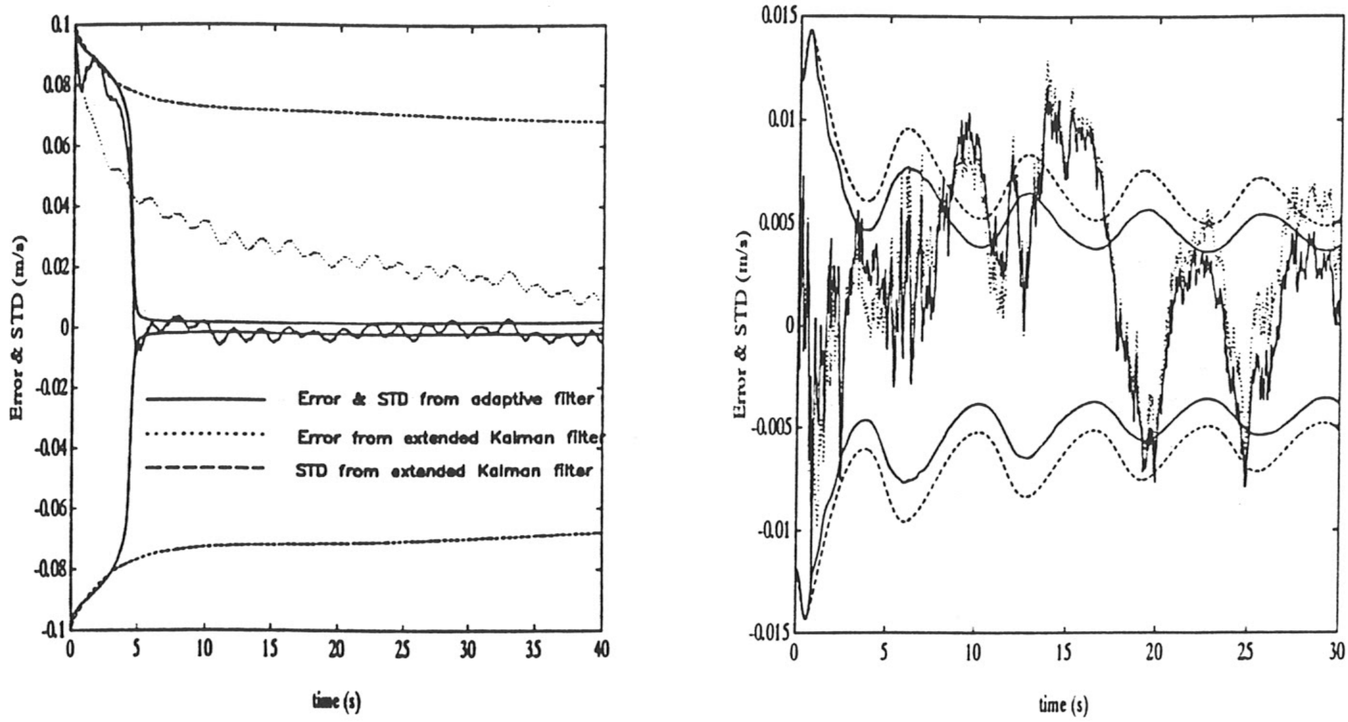


Figure 12: *Reconstructed flight velocities, respectively for aeroplane (A) and spaceplane (B)*

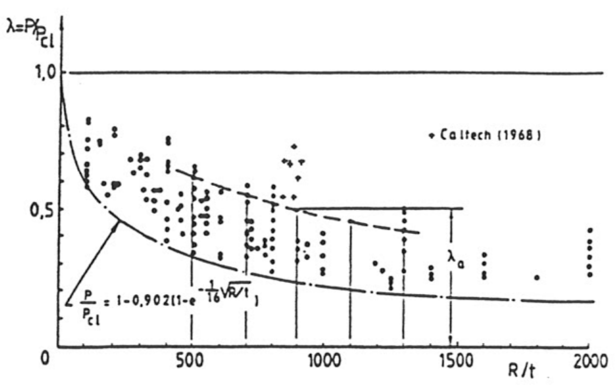


Figure 13: *Shell-structure design curves*

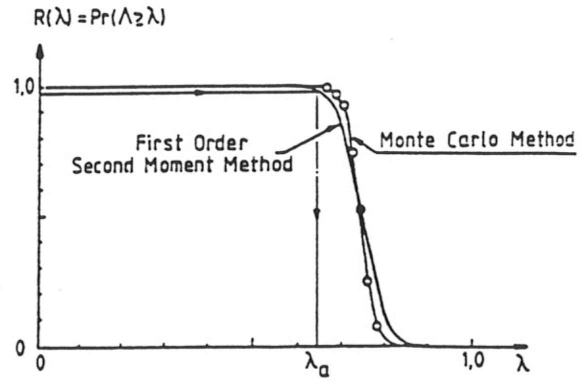


Figure 14: *Reliability function $R(\lambda)$ for a given value of R/t*

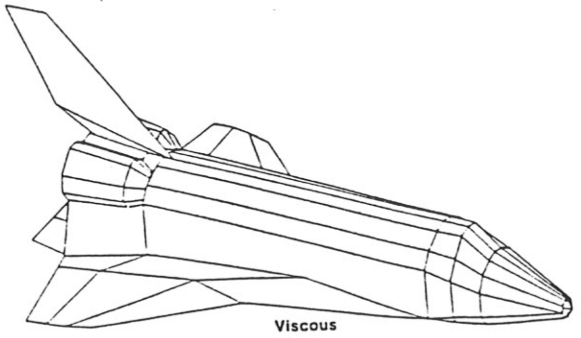


Figure 15: *Panel geometry for heat transfer calculation*

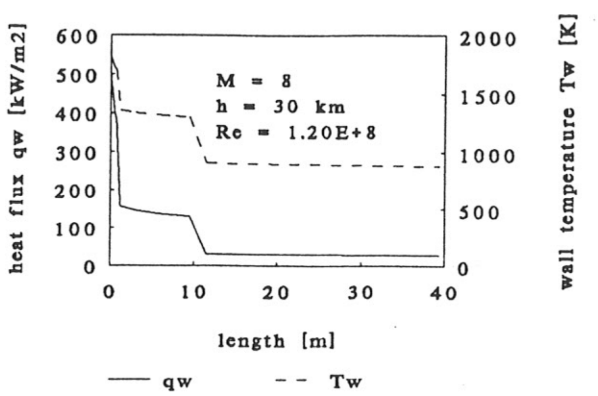


Figure 16: *Typical heat flux and temperature distribution for an axi-symmetrical rocket*

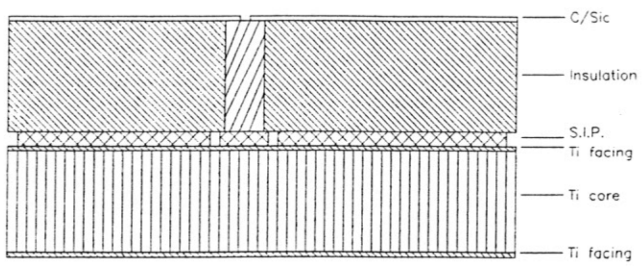


Figure 17: *Typical lay-out of intake ramp*

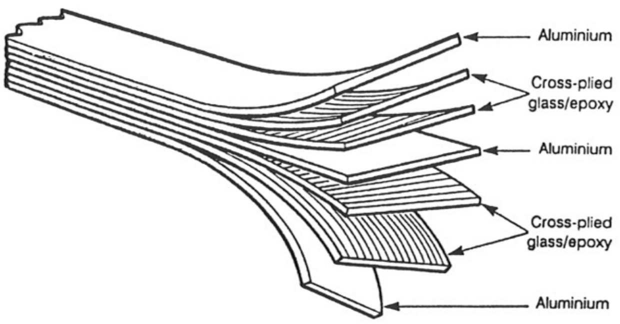


Figure 18: *Typical FML*

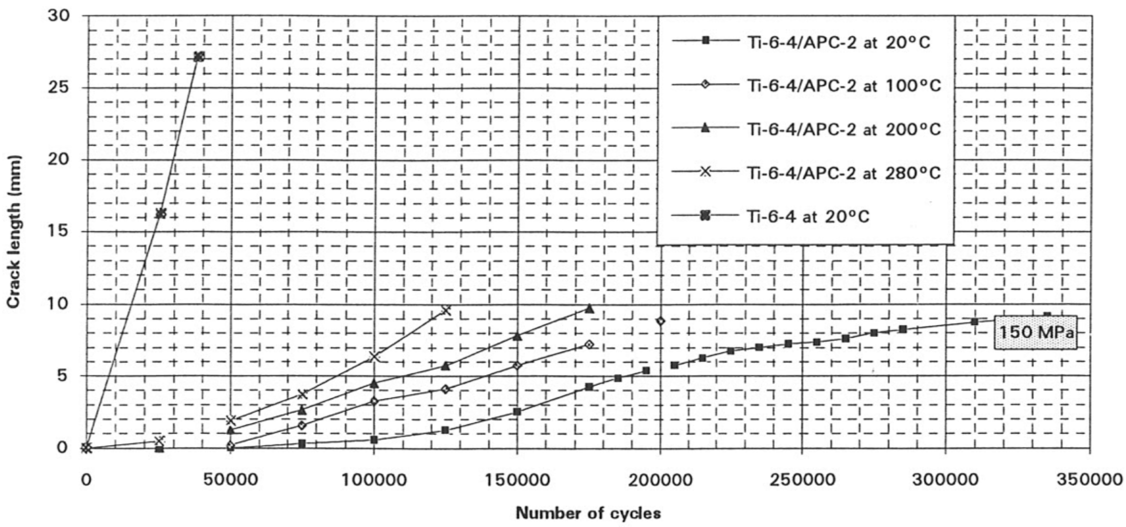


Figure 19: *Fatigue behaviour of Ti-6-4/APC-2 at 200 MPa at various temperatures*