



BSI Standards Publication

Space Engineering — Thermal design handbook

Part 16: Thermal Protection System

National foreword

This Published Document is the UK implementation of CEN/CLC/TR 17603-31-16:2021.

The UK participation in its preparation was entrusted to Technical Committee ACE/68, Space systems and operations.

A list of organizations represented on this committee can be obtained on request to its committee manager.

Contractual and legal considerations

This publication has been prepared in good faith, however no representation, warranty, assurance or undertaking (express or implied) is or will be made, and no responsibility or liability is or will be accepted by BSI in relation to the adequacy, accuracy, completeness or reasonableness of this publication. All and any such responsibility and liability is expressly disclaimed to the full extent permitted by the law.

This publication is provided as is, and is to be used at the recipient's own risk.

The recipient is advised to consider seeking professional guidance with respect to its use of this publication.

This publication is not intended to constitute a contract. Users are responsible for its correct application.

This publication is not to be regarded as a British Standard.

© The British Standards Institution 2021
Published by BSI Standards Limited 2021

ISBN 978 0 539 17018 4

ICS 49.140

Compliance with a Published Document cannot confer immunity from legal obligations.

This Published Document was published under the authority of the Standards Policy and Strategy Committee on 31 August 2021.

Amendments/corrigenda issued since publication

Date	Text affected
------	---------------

August 2021

ICS 49.140

English version

Space Engineering - Thermal design handbook - Part 16: Thermal Protection System

Ingénierie spatiale - Manuel de conception thermique -
Partie 16 : Protection Thermique des véhicules
spatiaux

Raumfahrttechnik - Handbuch für thermisches Design -
Teil 16: Thermalschutzsysteme

This Technical Report was approved by CEN on 28 June 2021. It has been drawn up by the Technical Committee CEN/CLC/JTC 5.

CEN and CENELEC members are the national standards bodies and national electrotechnical committees of Austria, Belgium, Bulgaria, Croatia, Cyprus, Czech Republic, Denmark, Estonia, Finland, France, Germany, Greece, Hungary, Iceland, Ireland, Italy, Latvia, Lithuania, Luxembourg, Malta, Netherlands, Norway, Poland, Portugal, Republic of North Macedonia, Romania, Serbia, Slovakia, Slovenia, Spain, Sweden, Switzerland, Turkey and United Kingdom.



CEN-CENELEC Management Centre:
Rue de la Science 23, B-1040 Brussels

Table of contents

European Foreword.....	5
1 Scope.....	6
2 References	7
3 Terms, definitions and symbols.....	8
3.1 Terms and definitions	8
3.2 Abbreviated terms.....	8
4 Introduction.....	9
4.1 General.....	9
4.2 Classification of thermal protection systems	10
5 Ablative systems	14
5.1 General.....	14
5.2 Ablative materials	14
5.3 Basic analysis.....	15
5.3.1 Surface equilibrium	16
5.4 Existing systems.....	19
5.4.1 Galileo probe.....	19
6 Radiative systems	23
6.1 General.....	23
6.2 Radiative materials	23
6.3 Existing systems.....	24
6.3.1 Space shuttle	24
6.4 Other developments	35
6.4.1 X-38	35
Bibliography.....	54
 Figures	
Figure 4-1: Velocity-altitude map for the Space Shuttle. Lifting re-entry from orbit.....	9
Figure 4-2: Summary of re-entry trajectories. From East (1991) [6].	10

Figure 4-3: Sketch of an ablative thermal protection system.....	11
Figure 4-4: Sketch of a radiative thermal protection system.....	11
Figure 4-5: Sketch of a transpiration thermal protection system.	12
Figure 4-6: Typical transpiration cooling system.....	13
Figure 5-1: Surface energy balance.....	17
Figure 5-2: Galileo entry probe.	20
Figure 5-3: Physical model and phenomena considered in material response analysis	20
Figure 5-4: Temperature history at interfaces.	22
Figure 5-5: Comparison of mass loss fluxes.	22
Figure 6-1: Worst case peak predicted surface temperatures. [K] for STS-1. From Dotts et al. (1983) [5].	25
Figure 6-2: Worst case peak predicted structure temperatures. [K] for STS-1. From Dotts et al. (1983) [5].	25
Figure 6-3: Thermal protection subsystems. From Dotts et al. (1983) [5].	26
Figure 6-4: RCC system components. From Curry et al. (1983) [3].	27
Figure 6-5: Nose cap system components. From Curry et al. (1983) [3].	27
Figure 6-6: Wing leading-edge system components. From Curry et al. (1983) [3].	28
Figure 6-7: Tile attachment and gap filler configuration. From Dotts et al. (1983) [5].	29
Figure 6-8: Nose cap RCC surface comparison between prediction and flight data. From Curry et al. (1983) [3].	30
Figure 6-9: Nose cap access door tile surface comparison between prediction and flight data. From Curry et al. (1983) [3].	30
Figure 6-10: Wing leading-edge panel (stagnation area). Comparison between prediction and flight data. From Curry et al. (1983) [3].	31
Figure 6-11: STS-1 flight data analysis comparison for lower mid-fuselage location. From Dotts et al. (1983) [3].	31
Figure 6-12: STS-1 flight data analysis comparison for lower wing location. From Dotts et al. (1983) [3].	32
Figure 6-13: STS-1 flight data analysis comparison for lower inboard elevon location. From Dotts et al. (1983) [3].	32
Figure 6-14: STS-1 flight data analysis comparison for lower mid-fuselage side location. From Dotts et al. (1983) [3].	33
Figure 6-15: Comparison of STS-2 data with analytical predictions. From Normal et al. (1983) [11].	33
Figure 6-16: Comparison of STS-2 data with analytical predictions. From Normal et al. (1983) [11].	34
Figure 6-17: Comparison of STS-2 data with analytical predictions. From Normal et al. (1983) [11].	34
Figure 6-18: In-depth comparison of STS-2 data with analytical predictions for maximum temperatures. From Normal et al. (1983) [11].	35
Figure 6-19: X-39 TPS Configuration.....	36
Figure 6-20: X-38 Reference Heating.....	36

Figure 6-21: CMC Side Panels together with lower CMC Chin Panel	37
Figure 6-22: Stand-off Position and Global Design	38
Figure 6-23: Stand-off Positions and Global Design	39
Figure 6-24: Max. Pressure Load	40
Figure 6-25: Max. Thermal Load at Panel Surface	40
Figure 6-26: Nose Skirt Assembly with Insulation Blankets	41
Figure 6-27: Max. and min. Heat flux time lines applied on the NSK.....	41
Figure 6-28: Simplified description of heat transfer modes within the nose skirt assembly.....	42
Figure 6-29: Temperature distribution over a NSK side panel at $t = 1100s$	44
Figure 6-30: Carrier Panel TPS Design	45
Figure 6-31: X-38 Aeroshell Panel and Blanket Distribution	46
Figure 6-32: X-38 Parafoil System.....	46
Figure 6-33: Parafoil Line Routing and Acreage Blankets.....	46
Figure 6-34: FEI-450 Blanket equipped with Gray FEI-1000High Emittance Coating.....	47
Figure 6-35: Typical look of FEI-650 and Blanket with Gray High Emittance	47
Figure 6-36: Allocation of Blanket Types to the X-38 Lee-Side Surface.....	49
Figure 6-37: Qualification Test Sequence for X-38	50
Figure 6-38: Parameters and Results of the Qualification Tests	50
Figure 6-39: Computer controlled sewing of FEI blankets.....	52
Figure 6-40: FEI-1000 blankets of the Forward Fuselage	52
Figure 6-41: FEI Blankets Integrated on the X-38 V-201	53

European Foreword

This document (CEN/CLC/TR 17603-31-16:2021) has been prepared by Technical Committee CEN/CLC/JTC 5 "Space", the secretariat of which is held by DIN.

It is highlighted that this technical report does not contain any requirement but only collection of data or descriptions and guidelines about how to organize and perform the work in support of EN 16603-31.

This Technical report (TR 17603-31-16:2021) originates from ECSS-E-HB-31-01 Part 16A.

Attention is drawn to the possibility that some of the elements of this document may be the subject of patent rights. CEN [and/or CENELEC] shall not be held responsible for identifying any or all such patent rights.

This document has been prepared under a mandate given to CEN by the European Commission and the European Free Trade Association.

This document has been developed to cover specifically space systems and has therefore precedence over any TR covering the same scope but with a wider domain of applicability (e.g.: aerospace).

1

Scope

The thermal protection system (TPS) of a space vehicle ensures the structural integrity of the surface of the craft and maintains the correct internal temperatures (for crew, electronic equipment, etc.) when the vehicle is under the severe thermal loads of re-entry. These loads are characterised by very large heat fluxes over the relatively short period of re-entry.

The design of thermal protection systems for re-entry vehicles is very complex due to the number and complexity of phenomena involved: the flow around the vehicle is hypersonic, tridimensional and reactive, and its interaction with the vehicle's surface may induce chemical reactions which are not fully understood.

Two TPS concepts for re-entry vehicles, ablative and radiative are examined and there is also an analysis of existing systems using them.

The Thermal design handbook is published in 16 Parts

TR 17603-31-01	Thermal design handbook – Part 1: View factors
TR 17603-31-02	Thermal design handbook – Part 2: Holes, Grooves and Cavities
TR 17603-31-03	Thermal design handbook – Part 3: Spacecraft Surface Temperature
TR 17603-31-04	Thermal design handbook – Part 4: Conductive Heat Transfer
TR 17603-31-05	Thermal design handbook – Part 5: Structural Materials: Metallic and Composite
TR 17603-31-06	Thermal design handbook – Part 6: Thermal Control Surfaces
TR 17603-31-07	Thermal design handbook – Part 7: Insulations
TR 17603-31-08	Thermal design handbook – Part 8: Heat Pipes
TR 17603-31-09	Thermal design handbook – Part 9: Radiators
TR 17603-31-10	Thermal design handbook – Part 10: Phase – Change Capacitors
TR 17603-31-11	Thermal design handbook – Part 11: Electrical Heating
TR 17603-31-12	Thermal design handbook – Part 12: Louvers
TR 17603-31-13	Thermal design handbook – Part 13: Fluid Loops
TR 17603-31-14	Thermal design handbook – Part 14: Cryogenic Cooling
TR 17603-31-15	Thermal design handbook – Part 15: Existing Satellites
TR 17603-31-16	Thermal design handbook – Part 16: Thermal Protection System

2 References

EN Reference	Reference in text	Title
EN 16603-00-01	ECSS-S-ST-00-01	ECSS System - Glossary of terms

All other references made to publications in this Part are listed, alphabetically, in the **Bibliography**.

Terms, definitions and symbols

3.1 Terms and definitions

For the purpose of this Standard, the terms and definitions given in ECSS-S-ST-00-01 apply.

3.2 Abbreviated terms

The following abbreviated terms are defined and used within this Standard.

CAD	computer aided design
CFD	computational fluid dynamics
CMC	ceramics matrix composite
C/SiC	carbon reinforced silicon carbide
FEI	flexible external insulation
FRSI	flexible reusable surface insulation
HTI	high temperature insulation
HRSI	high-temperature reusable surface insulation
IFI	internal flexible insulation
LRSI	low-temperature reusable surface insulation
RCC	reinforced carbon-carbon
RSI	reusable surface insulation
SIP	strain isolation pad
SOML	structural outer mold line
TOML	TPS outer mold line
TPS	thermal protection system

4

Introduction

4.1 General

The thermal protection system (TPS) of a space vehicle consists of those elements needed to protect the structural integrity of the vehicle's surface and maintain the appropriate internal temperatures (for crew, electronic equipment, etc.) when the vehicle is under the severe thermal loads of re-entry. These loads are mainly characterised by very large heat fluxes during relatively short times.

The heat fluxes acting on the TPS are so large because of the great speeds of re-entry vehicles. The velocity-altitude map for the Space Shuttle is represented in Figure 4-1.

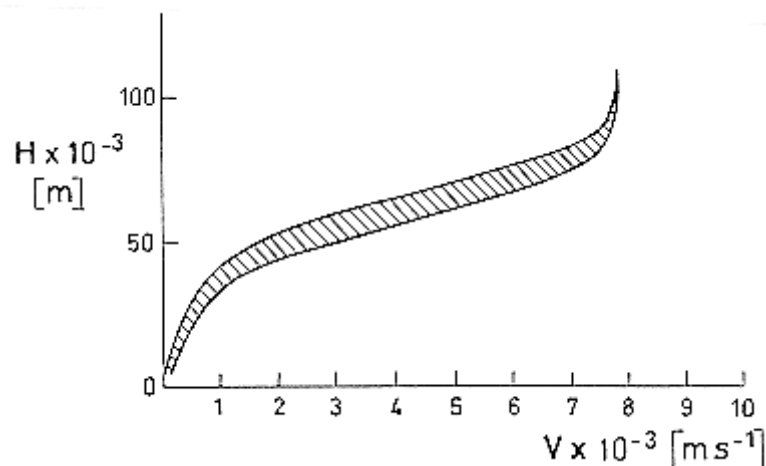


Figure 4-1: Velocity-altitude map for the Space Shuttle. Lifting re-entry from orbit.

The heat fluxes and the time of re-entry are basically determined by the re-entry orbit. These orbits are designed so that the vehicle is captured by the planet and the payload is not damaged by the accelerations; these factors greatly restrict the number of valid trajectories. However, for lifting vehicles which can be manoeuvred those restrictions are alleviated, and re-entry trajectories, other than ballistic, can be achieved. In Figure 4-2 the heat fluxes and re-entry times for different trajectories are summarised.

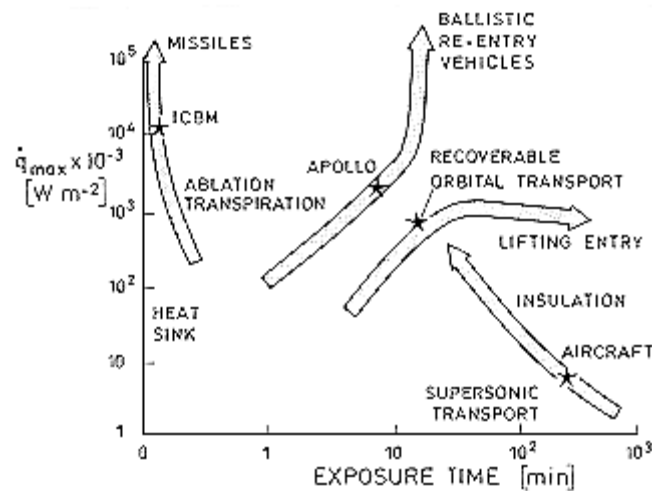


Figure 4-2: Summary of re-entry trajectories. From East (1991) [6].

The design of thermal protection systems for re-entry vehicles is a very complex problem due to the number and complexity of phenomena involved. It suffices to mention here that the flow around the vehicle is hypersonic, tridimensional and reactive, and its interaction with the vehicle's surface may induce chemical reactions which are not fully understood.

4.2 Classification of thermal protection systems

Generally speaking the TPS consists of a material system (shield and/or load carrying member) operating on a given heat dissipation principle. There are several TPS concepts for re-entry vehicles (Hurwicz & Rogan (1973a) [9]):

- Ablative thermal protection
- Radiative thermal protection
- Heat sinks
- Transpiration cooling

ABLATIVE SYSTEMS

Ablative systems operate dissipating the incident thermal energy through the loss of material: these systems lose mass as a consequence of the ablation of the external surface material. They have good thermal characteristics since phase changes absorb a large amount of energy. These systems are not reusable. See Figure 4-3 for a sketch of an ablative system.

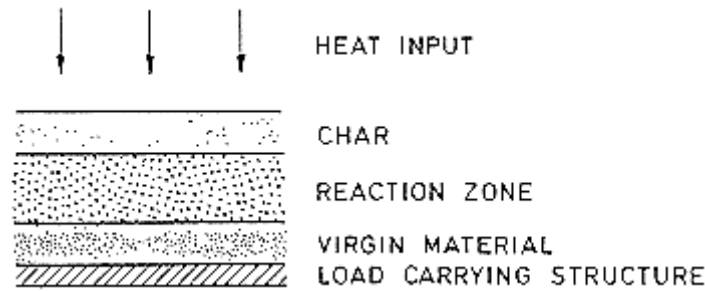


Figure 4-3: Sketch of an ablative thermal protection system.

The ablation process is quite complex and is described in some detail in clause 5.2. One important consequence of the analysis of these systems is that their efficiency is particularly sensitive to material performance. Therefore, it is necessary to treat the subject of materials in detail. In the absence of a universally acceptable ablative material a wide variety of ablative compositions and constructions have been produced, usually tailored to satisfy the requirements of a specific vehicle for a specific mission. A detailed description of ablative materials is given in clause 5.3.

RADIATIVE SYSTEMS

Radiative systems operate re-emitting by radiation the energy received from the surrounding environment. They are composed of two layers: an outer layer which consists of a material that can stand the radiation equilibrium temperature and an inner layer which insulates the outer layer from the structure in order to minimise the heat flow between the two, see Figure 4-4.

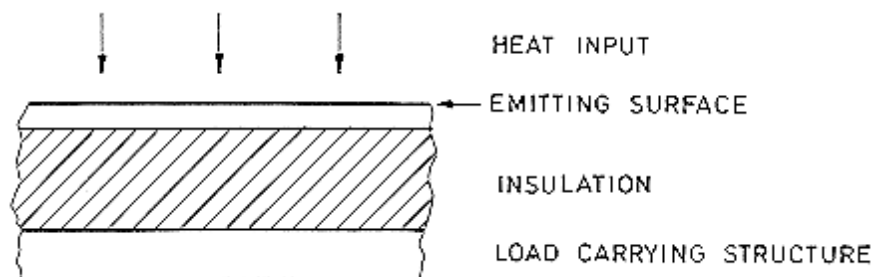


Figure 4-4: Sketch of a radiative thermal protection system.

It will be seen in clause 6.1 that the effectiveness of a radiative system increases very rapidly with increasing surface temperature and surface emissivity. Consequently, the primary development efforts have been concerned with the improvement of high emissivity, high temperature coatings, and with increasing the material service temperatures (including that of the internal insulation). A detailed description of materials used in radiative systems is given in clause 6.2.

These systems can be designed including a cooling subsystem: this is a fluid loop where the working fluid transports heat from the areas where the heat flux is stronger to those where the heat flux is weaker. The actual mechanism for heat transport can be the same as in heat pipes, the fluid is vaporised in areas of higher temperatures, and it is condensed in areas of lower temperatures. However, even though the characteristics of these systems are good, they are not used in practice.

HEAT SINK THERMAL PROTECTION SYSTEM

A heat sink is the simplest type of absorptive thermal protection system. It was used in the design of the early re-entry vehicles (e.g. the first two manned Mercury vehicles).

These systems are composed of an outer layer, comparatively thick, which consists of a material of high conductivity and capacitance. The function of this layer is to absorb the heat input. Since the material heats up, the storage capability is limited by the melting temperature.

Its use is limited to relatively low heating rates and therefore may not be practical for the high heat loads encountered in short re-entry times.

Heat sinks have the advantages of simplicity, dependability, and for reusable vehicles, ease of refurbishment. Their outstanding disadvantage is their low efficiency, this would cause a heat sink sized to satisfy most current re-entry missions to be excessively heavy.

Materials commonly used as heat sinks are

- beryllium
- beryllium oxide (beryllia)
- copper.

Graphite has many desirable heat sink characteristics, but begins to oxidise at temperatures far below those required for best efficiency.

TRANSPIRATION COOLING

Transpiration systems are systems where fluid is injected through a porous medium into the boundary layer. The structure is maintained cool by two basic mechanisms: heat is conducted to the coolant as it flows through the structure, and as the coolant is ejected out the surface it reduces the surface heat transfer rate by cooling and thickening the boundary layer. See Figure 4-5 for a sketch.

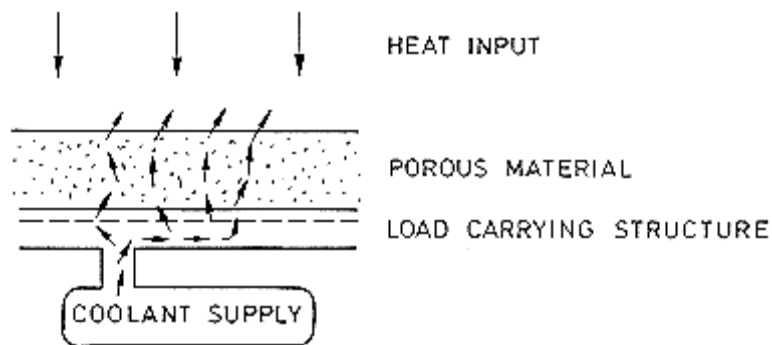


Figure 4-5: Sketch of a transpiration thermal protection system.

In some applications, the shape change caused by the surface recession of an ablating surface is not acceptable for aerodynamic performance reasons. In such cases, if the environment is too severe for radiative or heat sink systems, transpiration cooling may be the only practical solution. This TPS makes possible performance in environments that could not otherwise be withstood. However, its mechanical complexity (see Figure 4-6), with the associated reliability problems, tend to limit its use.

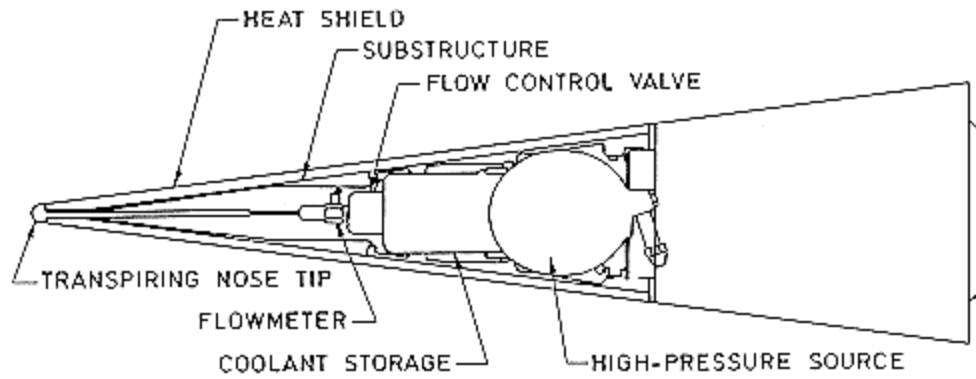


Figure 4-6: Typical transpiration cooling system

For re-entry application, the most acceptable coolants are:

- H_2O
- NH_3
- CF_4
- CO_2

Ablative systems

5.1 General

The ablation mechanism for thermal protection is based on the sublimation, melting or pyrolysis of the heat shield and the removal of the products by the outer stream. The great amount of energy absorbed in phase transition reduces the heat fluxes to the structure of the vehicle. This method has been widely used in most of non-reusable entry vehicles, for its simplicity and its high performance. It has been used in planetary probes, ballistic missiles and space capsules.

The methods of analysing the various heat shield materials vary depending on the melt temperature and oxidation chemistry. These materials may be classified as

- a. Oxidation controlled. The melting temperature is greater than the radiative equilibrium temperature calculated for the convective heat transfer rate.
- b. Simple sublimers. When melting temperature is lower than the radiative equilibrium temperature.
- c. Pyrolytic ablators. The material decomposition into pyrolysis gas and char occurs in depth.

Despite the above classification almost all heat shields are made with carbon based materials. This fact is due to special characteristic combination of very desirable properties as good heat sink, high melt temperature, large heat of sublimation, good radiation properties, and from the structural point of view low dilatation coefficient.

5.2 Ablative materials

Materials commonly used can be classified as follows:

Composites:

Carbon phenolic:

- High strength charring ablator with high ablation temperature.
- Used in high performance re-entry vehicles.
- It is the current heat shield material choice.
- Used in the Galileo probe (clause 5.4.1).

Silica phenolic:

- High strength charring ablator.
- Used in high performance re-entry vehicles.
- Selected for the Huygens probe.

Phenolic nylon:

- High heat shielding capability.
- It has the disadvantage of high erosion rate.
- Limited capability to accommodate high heat loads.
- Used in the Galileo probe aft shield (clause 5.4.1).

Ceramics:**Graphite:**

- Used in high heating rate areas.
- It has disadvantages as brittleness and low resistance to thermal stress.
- ATJ graphite has high strength properties.

Metals:**Tungsten:**

- Refractory metal.
- Used as a porous matrix infiltrated with copper or silver.
- It has good mechanical properties but low thermal performance.

Elastomers:**Silicone polymers:**

- Used reinforced by ceramic microspheres.
- Used in low shear and low pressure regions, and in low to moderate heat flux zones.
- It was used in the Gemini project.

Plastics:**Teflon:**

- Low temperature ablator with moderate efficiency and high ablation rates.
- It has been used in ballistic missiles.

AVCOAT 5026.

- Ablator used in the Apollo capsules.

5.3 Basic analysis

The methods for predicting surface degradation or recession of ablative thermal protection systems have been purely empirical or semitheoretical. The semitheoretical methods are based on simplifying assumptions that will be explained later. These simplifications are due to the extremely complicated physico-chemical phenomena involved in the ablation process. It includes phase change, non-equilibrium effects, multiphase flow, high thermal radiation environment, three dimensional hypersonic flow. The design of high mass heat shields has made it necessary accurate theoretical models for ablation. An accurate design of these heat shields is critical due to the high mass of ablator

necessary for protecting the vehicles. It can be noticed that heavy heat shields may make missions not feasible. An example of accurate theoretical design is the Galileo Jovian probe, it has been designed using numerical detailed flow field predictions, despite the uncertainties on the Jupiter atmosphere composition.

5.3.1 Surface equilibrium

To establish the energy flux to the re-entry vehicle, the energy transfer mechanisms between the boundary layer and the ablating surface of the vehicle must be stated. Despite the fact that the ablation problem is non-steady, the assumption of steady state will be made. This approximation is rather accurate for engineering purposes. It is assumed, also, that the fluid flow near the surface is governed by the equations of the boundary layer with mass injection. In this analysis (Hurwicz & Rogan (1973b) [10]) the conservation of mass and energy are applied to a thin control volume (since this is a moving control volume the Reynolds transport theorem should be taken into account).

First the mass balance is considered. The mass flux leaving the control volume through the gas side is due to gaseous species, that are denoted by $\dot{m}_w$, plus solid and liquid species, denoted by $\dot{m}_r$. The term $\dot{m}_w$ is due to sublimation of the heat shield and products from pyrolysis reactions that occur in the heat shield. The term $\dot{m}_r$ is due to mechanical erosion of liquid and solid products, the solid erosion is due to spallation of solid particles and to the surface stress due to percolation and sublimation, the liquid products are due to heat shield melting. This term, $\dot{m}_r$, should be reduced because this mass does not sublime, thus reducing the performance of the heat shield. The mass flux entering the control volume through the solid side is due to the products of pyrolysis reactions that occur in the heat shield, and to the solid mass that enters due to the movement of the control volume. The first flux is denoted by $\dot{m}_{g_s}$ and the second one by $\dot{m}_{c_s}$. The continuity equation yields

$$\dot{m}_w + \dot{m}_r = \dot{m}_{c_s} + \dot{m}_{g_s} \quad [5-1]$$

with

$$\dot{m}_{c_s} = \sum_{k'} (\dot{m}_{k'})_w, \quad \dot{m}_{g_s} = \sum_{k''} (\dot{m}_{k''})_w \quad [5-2]$$

where k' denotes summation over all the solid species entering the control volume, and k'' denotes summation over the gaseous species.

The energy balance yields

$$\begin{aligned} \dot{q}_s = \dot{q}_w - \dot{m}_w h_w - \dot{m}_r h_r + \dot{m}_{c_s} h_{c_s} + \\ + \dot{m}_{g_s} h_{g_s} - \varepsilon_w \sigma T_w^4 + \alpha_w \dot{q}_{rad} \end{aligned} \quad [5-3]$$

where the meaning of each term is explained below.

$\dot{q}_s$ is the energy flux to the solid, it is positive when the energy flux is to the solid.

$\dot{q}_w$ is the energy flux due to the aerodynamic heat transfer to the ablator from the boundary layer.