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USER MANUAL FOR THE BASE OF AIRCRAFT DATA (BADA) FAMILY 4

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TITLE:

USER MANUAL FOR THE BASE OF Aircraft DATA (BADA) FAMILY 4

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Abstract :

BADA (Base of Aircraft DAta) is an Aircraft Performance Model (APM) applicable for aircraft trajectory simulation and prediction within the domain of Air Traffic Management (ATM).

The APM adopted by BADA is based on a mass-varying, kinetic approach. This approach models an aircraft as a point and requires modelling of underlying forces that cause aircraft motion. The BADA APM is structured into four models: Actions, Motion, Operations and Limitations. The action model allows the computation of the forces acting on the aircraft which cause its motion (aerodynamic, propulsive and gravitational). The propulsive model provides an associated model to compute fuel consumption. Total Energy Model (TEM) which relates to the geometrical, kinematic and kinetic aspects of the aircraft motion is used to calculate aircraft performances and trajectory. Operations model provides knowledge about the way the aircraft is operated, necessary to compute related aircraft motion. The limitations model restricts the aircraft behaviour in order to keep it between certain limits to ensure the safe operation of the aircraft. The corresponding mathematical models are expressed in the form of polynomial expressions.

This document (User Manual for BADA Family 4) describes these expressions, provides definitions of each of the coefficients of polynomial expressions and explains the file formats.

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SUMMARY

BADA (Base of Aircraft DAta) is an Aircraft Performance Model applicable for aircraft trajectory simulation and prediction within the domain of Air Traffic Management.

The Aircraft Performance Model adopted by BADA is based on a mass-varying, kinetic approach. This approach models an aircraft as a point and requires modelling of underlying forces that cause aircraft motion. The BADA model is structured into four models: Actions, Motion, Operations and Limitations. The action model allows the computation of the forces acting on the aircraft which cause its motion (aerodynamic, propulsive and gravitational) and associated fuel consumption. The Total Energy Model, which relates to the geometrical, kinematic and kinetic aspects of the aircraft motion, is used to calculate aircraft performances and trajectory. The Operations model provides knowledge about the way the aircraft is operated, necessary to compute related aircraft motion. The Limitations model restricts the aircraft behaviour in order to keep it between certain limits to ensure the safe operation of the aircraft. The corresponding mathematical models are expressed in the form of polynomial expressions.

This document (User Manual for BADA Family 4) describes these expressions, provides definitions for each of the coefficients of polynomial expressions and explains the file formats.

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USER MANUAL MODIFICATION HISTORY

Version	Release Date	Comments
1.0	03.07.2012	First release of the document.
1.1	20.11.2012	ACM file format updated following initial feedback.
1.2	07.04.2014	Maximum power for turboprop engines now takes into account propeller efficiency: changes in section 3.3.2.2, two parameters added to TPM model. Fuel flow for turbofan and turboprop engines can no longer be lower than idle fuel flow: changes in sections 3.4.1 and 3.4.2. Correction of drag coefficient in clean configuration above M_{max}: change in section 3.2.4.1.2. Addition of bank angle and clarification of acceptable throttle parameter values in section 5.1. Addition of section 6.3.4: impact of buffet on maximum Mach number and altitude. C_{Vmin} value updated, C_{VminTO} introduced: changes in section 7.2 and Appendix B. Addition of PTF, PTD and GPF files: updated sections 9.6, 9.7 and 9.9. Addition of operational envelope and clarifications on aerodynamic configurations in Appendix B. Correction of several typos across the document.
1.3	11.03.2016	Addition of the MTKF turbofan engine rating: changes in sections 3.3.1, 7.2, 9.4. Addition of the Taxi Fuel Allowance: changes in sections 3.4 (formula 3.4-1), 7.2, 9.4. Modification of the rating model formula for the turbofan engine in the temperature-rated area: changes in sections 3.3.1 (formula 3.3-6), 7.2, 9.4. Correction of the transition altitude formula to manage transition altitudes located above the tropopause: changes in section 2.2.2 (formulas 2.2-27/28). Update of limitations checks in the PTF file description: changes to sections 9.6 and 9.7. Review and update of BADA releases and access description: changes to sections 9.2 and 9.3.

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1. INTRODUCTION

1.1. IDENTIFICATION

This document is the User Manual for the Base of Aircraft Data (BADA) Family 4.

1.2. PURPOSE

BADA (Base of Aircraft DAta) is an Aircraft Performance Model (APM) applicable for aircraft trajectory simulation and prediction within the domain of Air Traffic Management (ATM).

The APM adopted by BADA is based on a mass-varying, kinetic approach. This approach models an aircraft as a point and requires modelling of underlying forces that cause aircraft motion. The BADA APM is structured into four models: Actions, Motion, Operations and Limitations. The action model allows the computation of the forces acting on the aircraft which cause its motion (aerodynamic, propulsive and gravitational). The propulsive model provides an associated model to compute fuel consumption. Total Energy Model (TEM) which relates to the geometrical, kinematic and kinetic aspects of the aircraft motion is used to calculate aircraft performances and trajectory. Operations model provides knowledge about the way the aircraft is operated, necessary to compute related aircraft motion. The limitations model restricts the aircraft behaviour in order to keep it between certain limits to ensure the safe operation of the aircraft. The corresponding mathematical models are expressed in the form of polynomial expressions.

This document (User Manual for BADA Family 4) describes these expressions, provides definitions of each of the coefficients of polynomial expressions and explains the file formats. Details on the theoretical analysis that set the bases for the BADA Family 4 APM can be found in [RD1].

1.3. DOCUMENT ORGANISATION

This document consists of nine sections:

Section 1: Introduction, includes a list of referenced documents along with a summary of abbreviations, acronyms and mathematical notations used throughout the document.

Section 2: Global architecture of the model, introduces the different sub-models that together constitute the BADA aircraft model, as well as the atmosphere model used in BADA.

Section 3: Actions, defines the set of equations that constitute the models for weight, aerodynamic lift and drag, engine thrust, and fuel consumption.

Section 4: Motion and Operations, defines the system of equations that describes the aircraft motion, as well as different ways of operating the aircraft.

Section 5: BADA model use cases, provides guidelines on how to use the BADA actions, motion and operations models, depending on what the user knows about the aircraft (input) and the kind of information he is looking for (output).

Section 6: Limitations, defines the set of limitations that flight conditions need to respect to ensure a realistic aircraft behaviour.

Section 7: Aircraft characteristics, describes the set of coefficients that are used to define each BADA aircraft model.

Section 8: Global Aircraft Parameters, defines the set of global aircraft parameters that are valid for all, or a group of, aircraft.

Section 9: BADA release files, describes the different files in which the BADA aircraft parameters are stored.

In addition, **Appendix A** describes a method to determine the propeller efficiency algorithm (see section 3.3.4), and **Appendix B** defines a set of standard airline procedures (speed and aerodynamic configuration) for the different flight phases.

1.4. REFERENCED DOCUMENTS

RD1 Concept Document for the Base of Aircraft Data (BADA) Family 4, EEC Technical/Scientific Report No. 12/11/22-57, November 2012.

RD2 Aircraft Type Designators, ICAO Document 8643, 2015 edition, http://www.icao.int/publications/doc8643/

RD3 Manual of the ICAO Standard Atmosphere, ICAO Document No. 7488, 2nd Edition, 1964.

RD4 Base of Aircraft Data (BADA) Aircraft Performance Modelling Manual, EEC Technical Report No. 2009-009, April 2009.

RD5 BADA Support Application – User Guide, revision 1.1, August 2009.

RD6 Mathematical Handbook; M.R. Spiegel; 1968; McGraw-Hill book company.

1.5. ABBREVIATIONS AND ACRONYMS

ACM	**A**ir**c**raft **M**odel
AFCM	**A**erodynamic **F**orces and **C**onfigurations **M**odel
AGL	**A**bove **G**round **L**evel
ALM	**A**ircraft **L**imitations **M**odel
APM	**A**ircraft **P**erformance **M**odel
ARPM	**A**i**r**line **P**rocedure **M**odel
ASCII	**A**merican **S**tandard **C**ode for the **I**nterchange of **I**nformation
ATF	**A**ccuracy **T**ables **F**ile
ATM	**A**ir **T**raffic **M**anagement
BADA	**B**ase of **A**ircraft **Da**ta
BLM	**B**uffet **L**imitations **M**odel
CAS	**C**alibrated **A**ir**s**peed
CI	**C**ost **I**ndex
CPFSC	**C**oefficient of **P**ower-**S**pecific **F**uel **C**onsumption
DLM	**D**ynamic **L**imitations **M**odel
DPM	**D**rag **P**olar **M**odel
ECCF	**E**conomy **C**ruise **C**ost **F**unction
ECON	**Econ**omic
EEC	**E**UROCONTROL **E**xperimental **C**entre
ELM	**E**nvironmental **L**imitations **M**odel
ESF	**E**nergy **S**hare **F**actor
FL	**F**light **L**evel
GLM	**G**eometric **L**imitations **M**odel
GS	**G**round **S**peed
ICAO	**I**nternational **C**ivil **A**viation **O**rganisation
ISA	**I**nternational **S**tandard **A**tmosphere
HL	**H**igh-**L**ift
KLM	**K**inematic **L**imitations **M**odel
LDL	**L**an**d**ing **L**ength
LG	**L**anding **G**ear
LIDL	**L**ow **Idl**e
LRC	**L**ong **R**ange **C**ruise

MCMB	**M**aximum **C**li**mb**
MCRZ	**M**aximum **Cr**ui**z**e
MEC	**M**aximum **E**ndurance **C**ruise
MLW	**M**aximum **L**anding **W**eight
MRC	**M**aximum **R**ange **C**ruise
MSL	**M**ean **S**ea **L**evel
MTKF	**M**aximum **T**a**k**e-o**f**f
MTOW	**M**aximum **T**ake-**o**ff **W**eight
MTW	**M**aximum **T**axi **W**eight
MZFW	**M**aximum **Z**ero-**F**uel **W**eight
OEW	**O**perating **E**mpty **W**eight
OPM	**O**peration of configuration **P**arameters **M**odel
PEM	**P**iston **E**ngine **M**odel
PFM	**P**ropulsive **F**orces **M**odel
PTD	**P**erformance **T**able **D**ata
PTF	**P**erformance **T**able **F**ile
RMS	**R**oot-**M**ean-**S**quare
ROCD	**R**ate **o**f **C**limb or **D**escent
SR	**S**pecific **R**ange
TAS	**T**rue **A**irspeed
TEM	**T**otal-**E**nergy **M**odel
TFA	**T**axi **F**uel **A**llowance
TFM	**T**urbo**F**an **M**odel
TOL	**T**ake-**O**ff **L**ength
TPM	**T**urbo**P**rop **M**odel
WTC	**W**ake **T**urbulence **C**ategory

1.6. MATHEMATICAL NOTATION

A list of the symbols used in equations throughout this document is given below along with a description. Where appropriate, the engineering units typically associated with the symbol are also given.

a	speed of sound	[m/s]
C_{CI}	cost index coefficient	[-]
C_D	drag coefficient	[-]
C_f	fuel-related cost	[€/kg]
C_{fx}	fixed cost	[€]
C_F	fuel coefficient	[-]
C_h	time-related cost	[€/min]
C_L	lift coefficient	[-]
C_P	power coefficient	[-]
C_T	thrust coefficient	[-]
C_v	variable cost	[€]
C_W	weight coefficient	[-]
CI	cost index	[kg/min]
D	drag force	[N]
F	fuel flow	[kg/s] or [kg/min] or [kg/h]
g_0	gravitational acceleration	[m/s^2]
$\frac{dh}{dt}$	vertical speed	[m/s] or [ft/min]
h	geodetic altitude	[m] or [ft]
H	geopotential altitude	[m] or [ft]
H_p	geopotential pressure altitude	[m] or [ft]
L	lift force	[N]
L_{HV}	fuel lower heating value	[m^2/s^2]
m	aircraft mass	[kg]
M	Mach number	[-]
n	load factor	[-]
p	atmospheric pressure	[Pa]
r	ground distance	[nautical miles]

R	real gas constant for air	$[m^2/(K \cdot s^2)]$
ROCD	Rate of Climb or Descent	[m/s] or [ft/min]
S	reference wing surface area	$[m^2]$
T	temperature	[K]
Th	thrust force	[N]
V	speed	[m/s] or [kt]
ΔT	temperature difference	[K]
W	weight force	[N]
$\dot{W}_P$	engine power	[W]
β_T	temperature gradient	[K/m]
γ	flight path angle	[rad]
δ	pressure ratio	[-]
δ_{HL}	high-lift devices parameter	[-]
δ_{LG}	landing gear parameter	[-]
δ_{SB}	speed breaks parameter	[-]
δ_T	throttle parameter	[-]
η	propeller efficiency	[-]
θ	temperature ratio	[-]
θ_t	total temperature ratio	[-]
κ	adiabatic index	[-]
ρ	air density	$[kg/m^3]$
σ	density ratio	[-]
Φ	bank angle	[rad]
$\dot{\varphi}$	rate of turn	[rad/s]

2. GLOBAL ARCHITECTURE OF THE MODEL

The BADA model consists of two distinct components:

- an aircraft model, which is the main component,
- an atmosphere model, which is a secondary component that describes the atmospheric properties required to use the aircraft model.

2.1. AIRCRAFT MODEL

The BADA aircraft model is based on a mass-varying, kinetic approach to aircraft performance modeling. It is structured in three parts: the Aircraft Performance Model (APM), the Airline Procedure Model (ARPM) and the Aircraft Characteristics, as depicted in Figure 1.

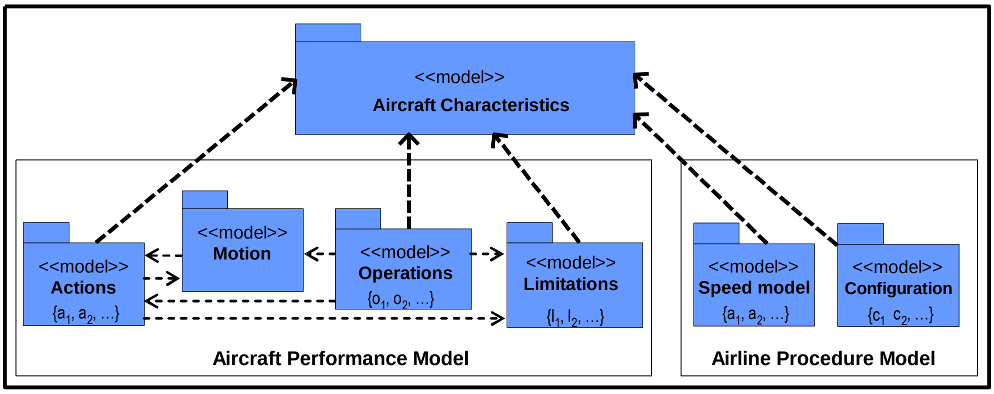

Figure 1 – BADA model structure

Each aircraft model in BADA is characterized with a set of coefficients, called **Aircraft Characteristics**, which are used by the APM and ARPM.

The BADA APM is structured into four sub-models:

- The **Actions** model (section 3) represents the forces acting on the aircraft and responsible for its motion. Those actions are divided into three categories – gravitational, aerodynamic and propulsive – including four forces acting on the aircraft (weight, lift, drag and thrust) plus the fuel consumption, which affects the aircraft mass.

- The **Motion** model (section 4) consists of the system of equations that describes the aircraft motion.

- The **Operations** model (section 4) describes different ways of operating the aircraft that are not part of Actions and Motion models.

- The **Limitations** model (section 6) ensures a realistic aircraft behaviour within certain limits.

Finally, a generic **Airline Procedure Model** is proposed in BADA: it provides nominal speeds and aerodynamic configurations for different flight phases, assuming normal aircraft operations.

2.2. ATMOSPHERE MODEL

This section provides expressions for the atmospheric properties (pressure, temperature, density and speed of sound) as a function of altitude, based on the International Standard Atmosphere (ISA) [RD3]. Those expressions are required for calculation of aircraft performances and movements, as well as conversions from CAS to TAS and Mach number.

The most important equations for atmospheric properties used by BADA and CAS/TAS conversion are summarised in this chapter, while other expressions and more details are provided in [RD1].

2.2.1. Definitions

Geopotential altitude H is that which under the standard constant gravitational field provides the same differential work performed by the standard acceleration of free fall when displacing the unit of mass a distance dH along the line of force, as that performed by the geopotential acceleration when displacing the unit of mass a geodetic distance dh [RD1].

Geopotential pressure altitude H_p is the geopotential altitude H that occurs in the ISA atmospheric conditions [RD1].

Mean Sea Level (MSL) Standard atmosphere conditions are those that occur in the International Standard Atmosphere (ISA) at the point where the geopotential pressure altitude H_p is zero. They are denoted as T_0, p_0, ρ_0 and a_0 with the values listed below:

Standard atmospheric temperature at MSL :	T_0	=	288.15	[K]
Standard atmospheric pressure at MSL :	p_0	=	101325	[Pa]
Standard atmospheric density at MSL :	ρ_0	=	1.225	[kg/m^3]
Speed of sound :	a_0	=	340.294	[m/s]

Mean Sea Level (MSL) atmosphere conditions are those that occur in a non-ISA atmosphere. They are identified by the sub-index MSL and differ from (T_0, p_0, ρ_0, a_0) in non-ISA conditions.

Non-ISA atmospheres are those that follow the same hypotheses as the ISA atmosphere but differ from it in that one or both of the following parameters is not zero:

1. **ΔT**. Temperature differential at MSL. It is the difference in atmospheric temperature at MSL between a given non-standard atmosphere and ISA.
2. **Δp**. Pressure differential at MSL. It is the difference in atmospheric pressure at MSL between a given non-standard atmosphere and ISA.

The values of these two parameters uniquely identify any non-ISA atmosphere. Thus, a non-ISA atmosphere provides expressions for the atmospheric pressure, temperature and density as functions of the geopotential altitude H and its two differentials. [RD1] provides more details on the corresponding analytical expressions.

2.2.2. Expressions

The relationships linking the atmospheric pressure p, temperature T, geopotential pressure altitude H_p and geopotential altitude H for any ISA[1] and non-ISA atmosphere are provided below.

a) Physical constants

Adiabatic index of air :	κ	=	1.4	
Real gas constant for air :	R	=	287.05287	$[m^2/(K\cdot s^2)]$
Gravitational acceleration :	g_0	=	9.80665	$[m/s^2]$
ISA temperature gradient with altitude below the tropopause :	$\beta_{T,<}$	=	- 0.0065	[K/m]

Note that subindex < denotes values below and at the tropopause and subindex > denotes values above the tropopause (as defined by 2.2-11).

b) Standard Mean Sea Level (subindex $H_p = 0$)

The temperature differential ΔT sets the value of the real temperature T in non-standard atmospheres.

$$H_{p,Hp=0} = 0 \tag{2.2-1}$$

$$p_{Hp=0} = p_0 \tag{2.2-2}$$

$$T_{ISA,Hp=0} = T_0 \tag{2.2-3}$$

$$T_{Hp=0} = T_0 + \Delta T \tag{2.2-4}$$

$$H_{Hp=0} = \frac{1}{\beta_{T,<}}\left[T_0 - T_{ISA,MSL} + \Delta T \cdot Ln\left(\frac{T_0}{T_{ISA,MSL}}\right)\right] \tag{2.2-5}$$

where T_{ISA} is the standard atmospheric temperature that occurs in the ISA atmosphere. It is a function of the geopotential pressure altitude H_p.

c) Mean Sea Level (subindex MSL)

[1] By replacing ΔT and Δp parameters with zeros the expressions are made applicable to the case of the standard atmosphere.

The pressure differential Δp sets the value of the atmospheric pressure p.

$$H_{MSL}=0 \tag{2.2-6}$$ [2]

$$p_{MSL} = p_0 + \Delta p \tag{2.2-7}$$

$$H_{p,MSL} = \frac{T_0}{\beta_{T,<}}\left[\left(\frac{p_{MSL}}{p_0}\right)^{-\frac{\beta_{T,<}R}{g_0}} - 1\right] \tag{2.2-8}$$

$$T_{ISA,MSL} = T_0 + \beta_{T,<} H_{p,MSL} \tag{2.2-9}$$

$$T_{MSL} = T_0 + \Delta T + \beta_{T,<} H_{p,MSL} \tag{2.2-10}$$

d) Tropopause

Tropopause is the separation between two different layers: the troposphere, which stands below it, and the stratosphere, which is placed above. Its altitude $H_{p,trop}$ is constant when expressed in terms of geopotential pressure altitude:

$$H_{p,trop} = 11000 \qquad [m] \tag{2.2-11}$$

e) Determination of Temperature

$$T = f(H_p, \Delta T) \tag{2.2-12}$$

$$T_< = T_0 + \Delta T + \beta_{T,<} H_{p,<} \tag{2.2-13}$$

$$T_{ISA,trop} = T_0 + \beta_{T,<} H_{p,trop} \tag{2.2-14}$$

$$T_{trop} = T_0 + \Delta T + \beta_{T,<} H_{p,trop} \tag{2.2-15}$$

$$T_> = T_{trop} \tag{2.2-16}$$

f) Determination of Air Pressure

$$p = f(T, \Delta T) \tag{2.2-17}$$

[2] In order to simplify the expressions, this document assumes that the geopotential altitude at mean sea level is always zero.

$$p_< = p_0 \left(\frac{T_< - \Delta T}{T_0} \right)^{-\frac{g_0}{\beta_{T,<} R}} \tag{2.2-18}$$

$$p_{trop} = p_0 \left(\frac{T_{trop} - \Delta T}{T_0} \right)^{-\frac{g_0}{\beta_{T,<} R}} \tag{2.2-19}$$

$T_> = T_{trop}$, so $p_>$ does not directly depend on temperature $T_>$. For altitudes above the tropopause, the following formula should be used:

$$p_> = p_{trop} \exp\left[-\frac{g_0}{R\,T_{ISA,trop}} \left(H_{p,>} - H_{p,trop} \right) \right] \tag{2.2-20}$$

where altitudes $H_{p,>}$ and $H_{p,trop}$ are expressed in [m].

g) <u>Determination of Air Density</u>

The air density, ρ [kg/m^3], is calculated from the pressure p and the temperature T at altitude using the perfect gas law:

$$\rho = \frac{p}{RT} \tag{2.2-21}$$

h) <u>Determination of Speed of Sound</u>

The speed of sound, a, is the speed at which the pressure waves travel through a fluid and it is given by the expression:

$$a = \sqrt{\kappa R T} \tag{2.2-22}$$

i) <u>CAS/TAS Conversion</u>

The true airspeed, V_{TAS} [m/s], is calculated as a function of the calibrated air speed, V_{CAS} [m/s], as follows:

$$V_{TAS} = \left[\frac{2}{\mu} \frac{p}{\rho} \left\{ \left(1 + \frac{p_0}{p} \left[\left(1 + \frac{\mu}{2} \frac{\rho_0}{p_0} V_{CAS}^{\;2} \right)^{\frac{1}{\mu}} - 1 \right] \right)^{\mu} - 1 \right\} \right]^{\frac{1}{2}} \tag{2.2-23}$$

Similarly, V_{CAS} [m/s] is calculated as a function of V_{TAS} [m/s] as follows:

$$V_{CAS} = \left[\frac{2}{\mu} \frac{p_0}{\rho_0} \left\{ \left(1 + \frac{p}{p_0} \left[\left(1 + \frac{\mu}{2} \frac{\rho}{p} V_{TAS}{}^2 \right)^{\frac{1}{\mu}} - 1 \right] \right)^{\mu} - 1 \right\} \right]^{\frac{1}{2}} \tag{2.2-24}$$

where symbols not previously defined are explained below:

$$\mu = \frac{\kappa - 1}{\kappa} \qquad \left(\mu = \frac{1}{3.5} \text{ if } \kappa = 1.4 \right) \tag{2.2-25}$$

j) Mach/TAS conversion

The true airspeed, V_{TAS} [m/s], is calculated as a function of the Mach number, M, as follows:

$$V_{TAS} = M \cdot \sqrt{\kappa R T} \tag{2.2-26}$$

k) Mach/CAS transition altitude

The transition altitude (also called crossover altitude), $H_{p,trans}$ [m], between a given CAS, V_{CAS} [m/s], and a Mach number, M, is defined to be the geopotential pressure altitude at which V_{CAS} and M represent the same TAS value, and can be calculated as follows:

$$H_{p,trans} = \begin{cases} \dfrac{T_0}{\beta_{T,<}} \cdot \left[\left(\dfrac{p_{trans}}{p_0} \right)^{-\frac{\beta_{T,<} R}{g_0}} - 1 \right] & \text{when } p_{trans} \geq p_{trop} \\ H_{P,trop} - \dfrac{R \cdot T_{ISA,trop}}{g_0} \cdot \mathrm{Ln}\left(\dfrac{p_{trans}}{p_{trop}} \right) & \text{when } p_{trans} < p_{trop} \end{cases} \tag{2.2-27}$$

where p_{trans} is the pressure at the transition altitude,

$$p_{trans} = p_0 \cdot \frac{\left[1 + \left(\dfrac{\kappa - 1}{2} \right) \left(\dfrac{V_{CAS}}{a_0} \right)^2 \right]^{\frac{\kappa}{\kappa - 1}} - 1}{\left[1 + \dfrac{\kappa - 1}{2} M^2 \right]^{\frac{\kappa}{\kappa - 1}} - 1} \tag{2.2-28}$$

l) Atmospheric ratios

Atmospheric ratios are dimensionless variables containing the ratios between the atmospheric properties at a given point and those found in standard mean sea level conditions (see section 2.2.1):

temperature ratio: $$\theta = \frac{T}{T_0} \tag{2.2-29}$$

pressure ratio: $$\delta = \frac{p}{p_0} \tag{2.2-30}$$

density ratio: $$\sigma = \frac{\rho}{\rho_0} \tag{2.2-31}$$

3. ACTIONS

The actions responsible for the aircraft motion are divided into three categories:

- Gravitational actions, including the weight force (section 3.1),
- Aerodynamic actions, including the lift and drag forces (section 3.2),
- Propulsive actions, including the engine thrust (section 3.3) and fuel consumption (section 3.4).

3.1. GRAVITATIONAL

The weight force, W [N], that applies to the aircraft is determined by:

$$W = m \cdot g_0 \tag{3.1-1}$$

Where :

m is the aircraft mass [kg]

g_0 is the gravitational acceleration [m/s^2], see section 2.2.2

3.2. AERODYNAMIC

The aerodynamic forces are the result of the interaction of the aircraft external surface with the atmosphere while flying through it, and as such depend on the external shape defined by the aerodynamic configuration (high-lift devices, landing gear and speed brakes), the atmospheric properties, and the Mach number. The Aerodynamic Forces and Configurations Model (AFCM) describes the aerodynamic forces and the possible aircraft aerodynamic configurations.

3.2.1. Aerodynamic configurations

The aircraft high-lift devices (e.g. flaps and slats) can be set to different positions, the number of which depends on the aircraft. The current position of the high-lift devices is named δ_{HL} in this document, and it equals 0 when all the high-lift devices are retracted. Similarly, the landing gear can be set to two different positions, either retracted (also called "up") or extracted (also called "down"). The current position of the landing gear is named δ_{LG} in this document, and it equals 0 when the landing gear is retracted.

An aerodynamic configuration is the combination of a high-lift devices position and a landing gear position. Such a configuration can be considered either "clean", if all the high-lift devices and the landing gear are retracted, or "non-clean" if any of those devices is extracted.

3.2.2. Lift coefficient

The lift coefficient, C_L, is determined assuming that the flight path angle is zero. However, a correction for a bank angle is made:

$$C_L = \frac{2 \cdot m \cdot g_0}{\delta \cdot p_0 \cdot \kappa \cdot S \cdot M^2 \cdot \cos\phi} \tag{3.2-1}$$

Where :

m is the aircraft mass [kg]

g_0 is the gravitational acceleration [m/s^2], see section 2.2.2

δ is the pressure ratio [-]

p_0 is the standard atmospheric pressure at MSL [Pa], see section 2.2.1

κ is the adiabatic index of air, see section 2.2.2

S is the wing reference area [m^2], from the AFCM

M is the Mach number [-]

Φ is the bank angle [rad]

3.2.3. Lift force

The lift force, L [N], is then determined from the lift coefficient in the standard manner:

$$L = \frac{1}{2} \cdot \delta \cdot p_0 \cdot \kappa \cdot S \cdot M^2 \cdot C_L \tag{3.2-2}$$

Where :

δ is the pressure ratio [-]

p_0 is the standard atmospheric pressure at MSL [Pa], see section 2.2.1

κ is the adiabatic index of air, see section 2.2.2

S is the wing reference area [m^2], from the AFCM

M is the Mach number [-]

3.2.4. Drag coefficient

The drag coefficient, C_D, is specified as a function of the lift coefficient C_L and the Mach number M. The form of this relationship depends on the aircraft aerodynamic configuration (see Section 3.2.1 for additional information about aerodynamic configurations). BADA Family 4 proposes two formulas to calculate the drag coefficient: a detailed one for the clean configuration, which takes into account the compressibility of air at high Mach numbers, and a simplified one for non-clean configurations, since those configurations cannot be used at high Mach numbers.

3.2.4.1. Drag coefficient in clean configuration

The highest Mach number for which the drag model in clean configuration has been identified, named M_{max}, is indicated in the clean drag polar model (DPM). The formula to be used to compute the drag coefficient in clean configuration is different whether the considered Mach number is lower or higher than M_{max}.

3.2.4.1.1. Drag coefficient in clean configuration below M_{max}

In clean configuration when the Mach number is lower or equal to M_{max}, the drag coefficient C_D is modelled as a polynomial of lift coefficient C_L, with coefficients depending on the Mach number M:

$$C_D = scalar \cdot \left[C_0 + \left(C_2 \cdot C_L^2\right) + \left(C_6 \cdot C_L^6\right)\right] \tag{3.2-3}$$

With:

$$C_0 = d_1 + \frac{d_2}{\left(1-M^2\right)^{\frac{1}{2}}} + \frac{d_3}{\left(1-M^2\right)} + \frac{d_4}{\left(1-M^2\right)^{\frac{3}{2}}} + \frac{d_5}{\left(1-M^2\right)^2} \tag{3.2-4}$$

$$C_2 = d_6 + \frac{d_7}{\left(1-M^2\right)^{\frac{3}{2}}} + \frac{d_8}{\left(1-M^2\right)^3} + \frac{d_9}{\left(1-M^2\right)^{\frac{9}{2}}} + \frac{d_{10}}{\left(1-M^2\right)^6} \tag{3.2-5}$$

$$C_6 = d_{11} + \frac{d_{12}}{\left(1-M^2\right)^7} + \frac{d_{13}}{\left(1-M^2\right)^{\frac{15}{2}}} + \frac{d_{14}}{\left(1-M^2\right)^8} + \frac{d_{15}}{\left(1-M^2\right)^{\frac{17}{2}}} \tag{3.2-6}$$

Where:

scalar is a scaling factor [-]

C_L is the lift coefficient [-]

M is the Mach number [-]

d_1 to d_{15} are clean drag coefficients [-], from the DPM

Note: This model takes up to 15 d_i coefficients to calculate C_D. The above equation shows the complete case containing all d_i coefficients. The number of coefficients, however, is not fixed and depends on the quantity and quality of the reference data with which the coefficients are identified, together with the modeller preferences. In many occasions this results in simpler expressions where some d_i coefficients are deactivated (i.e. equal to zero).

3.2.4.1.2. Drag coefficient in clean configuration above M_{max}

In clean configuration when the Mach number is higher than M_{max}, the drag coefficient C_D is extrapolated using the following formula:

$$C_D = C_{D,(M_{max}-0.01)} + \left[\left(\frac{M-(M_{max}-0.01)}{0.01}\right)^{\frac{3}{2}} \cdot \left(C_{D,M_{max}} - C_{D,(M_{max}-0.01)}\right)\right] \tag{3.2-7}$$

Where:

M is the Mach number [-]

M_{max} is the highest Mach number used for identification [-], from the DPM

$C_{D,Mmax}$ is the drag coefficient at M_{max} computed using (3.2-3) [-]

$C_{D,(Mmax-0.01)}$ is the drag coefficient at M_{max}-0.01 computed using (3.2-3) [-]

3.2.4.2. Drag coefficient in non-clean configurations

In non-clean configurations, the drag coefficient C_D is modelled as a polynomial of lift coefficient C_L, with coefficients depending on the aircraft aerodynamic configuration:

$$C_D = d_{1,\delta_{HL},\delta_{LG}} + d_{2,\delta_{HL},\delta_{LG}} \cdot C_L + d_{3,\delta_{HL},\delta_{LG}} \cdot C_L^2 \tag{3.2-8}$$

Where:

C_L is the lift coefficient [-]

δ_{HL} is the position of the high-lift devices [-], see section 3.2.1

δ_{LG} is the position of the landing gear [-], see section 3.2.1

$d_{1,\delta HL,\delta LG}$ to $d_{3,\delta HL,\delta LG}$ are non-clean drag coefficients [-], from the AFCM

3.2.4.3. Drag coefficient during transition between configurations

If the high-lift devices are in transition between two positions, the drag coefficient is obtained by linear interpolation between its values at the initial and final positions of the transition:

$$C_D = C_{D,1} + \frac{\delta_{HL} - \delta_{HL,1}}{\delta_{HL,2} - \delta_{HL,1}} \cdot \left(C_{D,2} - C_{D,1}\right) \tag{3.2-9}$$

Where :

δ_{HL} is the current position of the high-lift devices [-], see section 3.2.1

$\delta_{HL,1}$ is the initial position of the high-lift devices [-], see section 3.2.1

$\delta_{HL,2}$ is the final position of the high-lift devices [-], see section 3.2.1

$C_{D,1}$ is the drag coefficient in the initial configuration [-]

$C_{D,2}$ is the drag coefficient in the final configuration [-]

If the landing gear is in transition between two positions, the drag coefficient is obtained by linear interpolation between its values at the initial and final positions of the transition:

$$C_D = C_{D,1} + \delta_{LG} \cdot \left(C_{D,2} - C_{D,1}\right) \tag{3.2-10}$$

Where :

δ_{LG} is the current position of the landing gear [-], see section 3.2.1

$C_{D,1}$ is the drag coefficient in the initial configuration [-]

$C_{D,2}$ is the drag coefficient in the final configuration [-]

3.2.4.4. Drag coefficient increase when speed brakes are used

BADA Family 4 includes a simple speed brakes model that increases the drag coefficient as follows:

$$C_{D_{SB}} = C_{D_{noSB}} + 0.03 \cdot \delta_{SB} \tag{3.2-11}$$

Where:

$C_{D,SB}$ is the drag coefficient when speed brakes are used [-]

$C_{D,noSB}$ is the drag coefficient when speed brakes are not used [-]

δ_{SB} is the position of the speed brakes, either 0 (speed brakes not in use) or 1 (speed brakes in use) [-]

3.2.5. Drag force

The drag force, D [N], is then determined from the drag coefficient in the standard manner:

$$D = \frac{1}{2} \cdot \delta \cdot p_0 \cdot \kappa \cdot S \cdot M^2 \cdot C_D \quad (3.2\text{-}12)$$

Where :

δ is the pressure ratio [-]

p_0 is the standard atmospheric pressure at MSL [Pa], see section 2.2.1

κ is the adiabatic index of air, see section 2.2.2

S is the wing reference area [m^2]

M is the Mach number [-]

3.3. ENGINE THRUST

The BADA Family 4 model provides three separate thrust models as part of the Propulsive Forces Model (PFM), depending on the type of engine:

- turbofan: TurboFan Model (TFM),
- turboprop: TurboProp Model (TPM),
- piston: Piston Engine Model (PEM).

Each model includes the contribution from all engines and provides the thrust as a function of airspeed, throttle setting and atmospheric conditions. The general formulation of the thrust force, Th [N], is:

$$Th = \delta \cdot W_{mref} \cdot C_T \qquad (3.3\text{-}1)$$

Where :

δ is the pressure ratio [-]

m_{ref} is the reference mass [kg], from the PFM

W_{mref} is the weight force at m_{ref} [N], see section 3.1

C_T is the thrust coefficient [-], see sections 3.3.1.1, 3.3.2.1 and 3.3.3.1

The subsections below provide the additional formulas of the thrust model for each of the engine types.

3.3.1. Turbofan thrust model

A turbofan engine may be operated either by direct control of the throttle, or through the use of predefined settings, called ratings. The following ratings are modelled for turbofan engines: low idle thrust (LIDL), maximum climb thrust (MCMB), maximum cruise thrust (MCRZ) and maximum take-off thrust (MTKF). The MCMB, MCRZ and MTKF ratings have their own respective set of coefficients but share the same formulas, whereas the LIDL rating is modelled by different formulas, as detailed in the following subsections.

3.3.1.1. Turbofan thrust coefficient

This section provides the formulas to compute the thrust coefficient used in formula (3.3-1), depending on the engine rating.

3.3.1.1.1. Idle rating

The idle rating model for the turbofan engine model directly provides the thrust coefficient C_T as a function of the Mach number and the atmospheric conditions:

$$\begin{aligned} C_T = \; & ti_1\delta^{-1} + ti_2 + ti_3\delta + ti_4\delta^2 \\ & + \left(ti_5\delta^{-1} + ti_6 + ti_7\delta + ti_8\delta^2\right)\cdot M \\ & + \left(ti_9\delta^{-1} + ti_{10} + ti_{11}\delta + ti_{12}\delta^2\right)\cdot M^2 \end{aligned} \qquad (3.3\text{-}2)$$

Where :

δ is the pressure ratio [-]

M is the Mach number [-]

ti_1 to ti_{12} are idle rating thrust coefficients [-], from the TFM

3.3.1.1.2. Non-idle ratings (MCMB, MCRZ, MTKF) and no rating (direct throttle parameter input)

The generalized thrust form for the turbofan engine model provides C_T as a function of the Mach number M and the throttle parameter δ_T. C_T is calculated as a fifth order polynomial of δ_T with coefficients that are fifth order polynomials of M:

$$
\begin{aligned}
C_T = \; & a_1 + a_2 M + a_3 M^2 + a_4 M^3 + a_5 M^4 + a_6 M^5 \\
& + \left(a_7 + a_8 M + a_9 M^2 + a_{10} M^3 + a_{11} M^4 + a_{12} M^5\right) \cdot \delta_T \\
& + \left(a_{13} + a_{14} M + a_{15} M^2 + a_{16} M^3 + a_{17} M^4 + a_{18} M^5\right) \cdot \delta_T^2 \\
& + \left(a_{19} + a_{20} M + a_{21} M^2 + a_{22} M^3 + a_{23} M^4 + a_{24} M^5\right) \cdot \delta_T^3 \\
& + \left(a_{25} + a_{26} M + a_{27} M^2 + a_{28} M^3 + a_{29} M^4 + a_{30} M^5\right) \cdot \delta_T^4 \\
& + \left(a_{31} + a_{32} M + a_{33} M^2 + a_{34} M^3 + a_{35} M^4 + a_{36} M^5\right) \cdot \delta_T^5
\end{aligned}
\qquad (3.3\text{-}3)
$$

Where :

δ_T is the throttle parameter [-], see section 3.3.1.2

M is the Mach number [-]

a_1 to a_{36} are non-idle rating thrust coefficients [-], from the TFM

Note: This model takes up to 36 a_i coefficients to calculate C_T. The above equation shows the complete case containing all a_i coefficients. However, the number of coefficients is not fixed and depends on the quantity and quality of the reference data with which the coefficients are identified, together with the modeller preferences. In many occasions this results in simpler expressions where some a_i coefficients are deactivated (i.e. equal to zero).

3.3.1.2. Turbofan rating models

This section provides the formulas to compute the throttle parameter δ_T used in formula (3.3-3) when the engine is operated through a non-idle rating. A rating model is provided to determine the throttle position, and this rating model is made available for several different ratings, namely:

- maximum cruise (MCRZ),
- maximum climb (MCMB),
- maximum take-off (MTKF).

Turbofan engines behave differently whether they are operated at a temperature deviation below or above a threshold temperature deviation called the kink point, which defines two operation areas:

- The flat-rated area,
- The temperature-rated area.

When the atmospheric conditions result in a temperature deviation inferior to the kink point, the turbofan operates in the flat-rated area, in which the engine behavior is limited by the internal pressure. When the temperature deviation exceeds the kink point, the amount of fuel being injected into the combustion chamber must be reduced to control the turbine entry temperature: the turbofan then operates in the temperature-rated area. The two areas are thus inherently different and as such are modeled by two independent functions, each with its respective coefficients:

$$\delta_T = \begin{cases} \delta_{T,flat} & \text{when } \Delta T \leq \Delta T_{kink} \\ \delta_{T,temp} & \text{when } \Delta T > \Delta T_{kink} \end{cases} \tag{3.3-4}$$

Where :

δ_T is the throttle parameter [-]

$\delta_{T,flat}$ is the throttle parameter in the flat-rated area [-], see section 3.3.1.2.1

$\delta_{T,temp}$ is the throttle parameter in the temperature-rated area [-], see section 3.3.1.2.2

ΔT_{kink} is the kink point [K], from the TFM

3.3.1.2.1. Flat-rated area

The rating model for the turbofan engine in the flat-rated area provides the throttle parameter $\delta_{T,flat}$ as a function of the Mach number and the atmospheric conditions:

$$\begin{aligned} \delta_{T,flat} = \; & b_1 + b_2 M + b_3 M^2 + b_4 M^3 + b_5 M^4 + b_6 M^5 \\ & + \left(b_7 + b_8 M + b_9 M^2 + b_{10} M^3 + b_{11} M^4 + b_{12} M^5\right) \cdot \delta \\ & + \left(b_{13} + b_{14} M + b_{15} M^2 + b_{16} M^3 + b_{17} M^4 + b_{18} M^5\right) \cdot \delta^2 \\ & + \left(b_{19} + b_{20} M + b_{21} M^2 + b_{22} M^3 + b_{23} M^4 + b_{24} M^5\right) \cdot \delta^3 \\ & + \left(b_{25} + b_{26} M + b_{27} M^2 + b_{28} M^3 + b_{29} M^4 + b_{30} M^5\right) \cdot \delta^4 \\ & + \left(b_{31} + b_{32} M + b_{33} M^2 + b_{34} M^3 + b_{35} M^4 + b_{36} M^5\right) \cdot \delta^5 \end{aligned} \tag{3.3-5}$$

Where :

δ is the pressure ratio [-]

M is the Mach number [-]

b_1 to b_{36} are flat-rated area throttle coefficients [-], from the TFM

3.3.1.2.2. Temperature-rated area

The rating model for the turbofan engine in the temperature-rated area provides the throttle parameter $\delta_{T,temp}$ as a function of the Mach number and the atmospheric conditions:

$$\begin{aligned} \delta_{T,temp} = \; & c_1 + c_2 M + c_3 M^2 + c_4 M^3 + c_5 M^4 \\ & + \left(c_6 + c_7 M + c_8 M^2 + c_9 M^3 + c_{10} M^4\right) \cdot \theta_t \\ & + \left(c_{11} + c_{12} M + c_{13} M^2 + c_{14} M^3 + c_{15} M^4\right) \cdot \theta_t^2 \\ & + \left(c_{16} + c_{17} M + c_{18} M^2 + c_{19} M^3 + c_{20} M^4\right) \cdot \theta_t^3 \\ & + \left(c_{21} + c_{22} M + c_{23} M^2 + c_{24} M^3 + c_{25} M^4\right) \cdot \theta_t^4 \\ & + \left(c_{26} + c_{27} M + c_{28} M^2 + c_{29} M^3 + c_{30} M^4\right) \cdot \delta \\ & + \left(c_{31} + c_{32} M + c_{33} M^2 + c_{34} M^3 + c_{35} M^4\right) \cdot \delta^2 \\ & + \left(c_{36} + c_{37} M + c_{38} M^2 + c_{39} M^3 + c_{40} M^4\right) \cdot \delta^3 \\ & + \left(c_{41} + c_{42} M + c_{43} M^2 + c_{44} M^3 + c_{45} M^4\right) \cdot \delta^4 \end{aligned} \tag{3.3-6}$$

Where :

M is the Mach number [-]

c_1 to c_{45} are temperature-rated area throttle coefficients [-], from the TFM

δ is the pressure ratio [-]

θ_t is the total temperature ratio [-]:

$$\theta_t = \theta \cdot \left(1 + \frac{M^2 \cdot (\kappa - 1)}{2}\right) \tag{3.3-7}$$

Where :

θ is the temperature ratio [-]

M is the Mach number [-]

κ is the adiabatic index of air [-], see section 2.2.2

3.3.2. Turboprop thrust model

A turboprop engine may be operated either by direct control of the throttle, or through the use of predefined settings, called ratings. The following ratings are modelled for turboprop engines: low idle thrust (LIDL), maximum climb thrust (MCMB) and maximum cruise thrust (MCRZ). The MCMB and MCRZ ratings have their own respective set of coefficients but share the same formulas, whereas the LIDL rating is modelled by different formulas, as detailed in the following subsections.

3.3.2.1. Turboprop thrust coefficient

This section provides the formulas to compute the thrust coefficient used in formula (3.3-1), depending on the engine rating.

3.3.2.1.1. Idle rating

The idle rating model for the turboprop engine model directly provides the thrust coefficient C_T as a function of the Mach number and the atmospheric conditions:

$$\begin{aligned} C_T = \; & ti_1\delta^{-1} + ti_2 + ti_3\delta + ti_4\delta^2 \\ & + \left(ti_5\delta^{-1} + ti_6 + ti_7\delta + ti_8\delta^2\right)\cdot M \\ & + \left(ti_9\delta^{-1} + ti_{10} + ti_{11}\delta + ti_{12}\delta^2\right)\cdot M^2 \\ & + ti_{13}\theta^{\frac{1}{2}} + ti_{14}\theta + ti_{15}\theta^{-\frac{1}{2}} + ti_{16}\theta^2 \\ & + \left(ti_{17}\delta^{-1} + ti_{18}\delta + ti_{19}\delta^2 + ti_{20}M + ti_{21}M^2\right)\cdot \theta^{-\frac{1}{2}} \\ & + ti_{22}M^{-1} + ti_{23}M^{-1}\delta + ti_{24}M^3 \\ & + \left(ti_{25}M + ti_{26}M^2 + ti_{27} + ti_{28}M\delta^{-1}\right)\cdot \theta^{-1} \\ & + ti_{29}M\delta^{-1}\theta^{-2} + ti_{30}M^2\delta^{-1}\theta^{-2} + ti_{31}M^2\delta^{-1}\theta^{-\frac{1}{2}} + ti_{32}\delta\theta^{-1} \end{aligned} \tag{3.3-8}$$

Where :

δ is the pressure ratio [-]

θ is the temperature ratio [-]

M is the Mach number [-]

ti_1 to ti_{32} are idle rating thrust coefficients [-], from the TPM

3.3.2.1.2. Non-idle ratings (MCMB, MCRZ) and no rating (direct throttle parameter input)

For non-idle ratings, the turboprop engine model is similar to the turbofan engine model. The major difference is that the turboprop model uses a power coefficient instead of a thrust coefficient. To obtain the thrust coefficient, the power coefficient has to be divided by the Mach number:

$$C_T = \frac{C_P}{M} \tag{3.3-9}$$

Where :

C_T is the thrust coefficient [-]

C_P is the power coefficient [-], see section 3.3.2.2

M is the Mach number [-]

3.3.2.2. Turboprop power coefficient

The generalized power form for the turboprop engine model provides the general power coefficient $C_{P,gen}$ as a function of the Mach number M and the throttle parameter δ_T. $C_{P,gen}$ is calculated as a fifth order polynomial of δ_T with coefficients that are fifth order polynomials of M:

$$\begin{aligned} C_{P,gen} = \; & a_1 + a_2M + a_3M^2 + a_4M^3 + a_5M^4 + a_6M^5 \\ & + \left(a_7 + a_8M + a_9M^2 + a_{10}M^3 + a_{11}M^4 + a_{12}M^5\right)\cdot \delta_T \\ & + \left(a_{13} + a_{14}M + a_{15}M^2 + a_{16}M^3 + a_{17}M^4 + a_{18}M^5\right)\cdot \delta_T^2 \\ & + \left(a_{19} + a_{20}M + a_{21}M^2 + a_{22}M^3 + a_{23}M^4 + a_{24}M^5\right)\cdot \delta_T^3 \\ & + \left(a_{25} + a_{26}M + a_{27}M^2 + a_{28}M^3 + a_{29}M^4 + a_{30}M^5\right)\cdot \delta_T^4 \\ & + \left(a_{31} + a_{32}M + a_{33}M^2 + a_{34}M^3 + a_{35}M^4 + a_{36}M^5\right)\cdot \delta_T^5 \end{aligned} \tag{3.3-10}$$

Where :

δ_T is the throttle parameter [-], see section 3.3.2.3

M is the Mach number [-]

a_1 to a_{36} are non-idle rating power coefficients [-], from the TPM

Note: This model takes up to 36 a_i coefficients to calculate $C_{P,gen}$. The above equation shows the complete case containing all a_i coefficients. However, the number of coefficients is not fixed and depends on the quantity and quality of the reference data with which the coefficients are identified, together with the modeller preferences. In many occasions this results in simpler expressions where some a_i coefficients are deactivated (i.e. equal to zero).

When a non-idle rating is applicable, the power produced by the power plants, $\dot{W}_P$, shall be inferior to the maximum all-engine power stipulated for the individual rating, $\dot{W}_{P_{max}}$. The power produced by the power plants is determined as follows:

$$\dot{W}_P = \eta^{-1} \cdot Th \cdot V_{TAS} = \eta^{-1} \cdot \delta \cdot W_{mref} \cdot a \cdot C_P \tag{3.3-11}$$

Where :

η is the propeller efficiency [-], see section 3.3.4

δ is the pressure ratio [-]

m_{ref} is the reference mass [kg], from the PFM

W_{mref} is the weight force at m_{ref} [N], see section 3.1

a is the speed of sound [m/s], see expression (2.2-22)

C_P is the power coefficient [-]

The power coefficient C_P is thus determined by:

$$C_P = \begin{cases} C_{P,gen} & \text{when no rating is used} \\ \min\left(C_{P,gen}, \dot{W}_{P_{max}} \cdot \eta \cdot \delta^{-1} \cdot W_{mref}^{-1} \cdot a^{-1}\right) & \text{for non-idle ratings} \end{cases} \quad (3.3\text{-}12)$$

Where :

$C_{P,gen}$ is the general power coefficient [-], see expression (3.3-10)

η is the propeller efficiency [-], see section 3.3.4

$\dot{W}_{P_{max}}$ is the maximum all-engine power [W], from the TPM

δ is the pressure ratio [-]

m_{ref} is the reference mass [kg], from the PFM

W_{mref} is the weight force at m_{ref} [N], see section 3.1

a is the speed of sound [m/s], see expression (2.2-22)

3.3.2.3. Turboprop rating model

This section provides the formulas to compute the throttle parameter δ_T used in formula (3.3-10) when the engine is operated through a non-idle rating. A rating model is provided to determine the throttle position, and this rating model is made available for several different ratings, namely:

- maximum cruise (MCRZ),
- maximum climb (MCMB).

Each rating model for the turboprop engine provides the throttle parameter δ_T as a function of the Mach number and the atmospheric conditions:

$$\begin{aligned} \delta_T = \; & p_1 + p_2 M + p_3 M^2 + p_4 M^3 + p_5 M^4 + p_6 M^5 \\ & + \left(p_7 + p_8 M + p_9 M^2 + p_{10} M^3 + p_{11} M^4 + p_{12} M^5\right) \cdot \theta \\ & + \left(p_{13} + p_{14} M + p_{15} M^2 + p_{16} M^3 + p_{17} M^4 + p_{18} M^5\right) \cdot \theta^2 \\ & + \left(p_{19} + p_{20} M + p_{21} M^2 + p_{22} M^3 + p_{23} M^4 + p_{24} M^5\right) \cdot \theta^3 \\ & + \left(p_{25} + p_{26} M + p_{27} M^2 + p_{28} M^3 + p_{29} M^4 + p_{30} M^5\right) \cdot \theta^4 \\ & + \left(p_{31} + p_{32} M + p_{33} M^2 + p_{34} M^3 + p_{35} M^4 + p_{36} M^5\right) \cdot \theta^5 \end{aligned} \quad (3.3\text{-}13)$$

Where :

M is the Mach number [-]

p_1 to p_{36} are non-idle rating throttle coefficients [-], from the TPM

θ is the temperature ratio [-]

3.3.3. Piston thrust model

This section provides the formulas to compute the thrust coefficient used in formula (3.3-1) for piston engines. A piston engine is usually operated by direct control of the throttle, so the notion of rating will be briefly introduced only for the sake of consistency with the turbofan and turboprop thrust models.

3.3.3.1. *Piston thrust coefficient*

The generalized thrust form for the piston engine model provides the thrust coefficient C_T as a function of the power coefficient C_P:

$$C_T = \eta \cdot a_0 \cdot C_P \cdot \delta^{-1} \cdot V_{TAS}^{-1} \tag{3.3-14}$$

Where :

η is the propeller efficiency [-], see section 3.3.4

a_0 is the speed of sound in standard atmosphere at MSL [m/s], see section 2.2.1

C_P is the power coefficient [-], see section 3.3.3.2

δ is the pressure ratio [-]

V_{TAS} is the true airspeed [m/s]

3.3.3.2. *Piston power coefficient*

The relationship between the engine power and the power coefficient is:

$$\dot{W}_P = W_{mref} \cdot a_0 \cdot C_P \tag{3.3-15}$$

Where :

$\dot{W}_P$ is the all-engine power [W]

m_{ref} is the reference mass [kg], from the PFM

W_{mref} is the weight force at m_{ref} [N], see section 3.1

a_0 is the speed of sound in standard atmosphere at MSL [m/s], see section 2.2.1

C_P is the power coefficient [-]

Expression (3.3-15) is employed to obtain the maximum power coefficient in standard atmosphere at MSL, based on the maximum sea level power that the piston engine can generate in standard atmosphere:

$$C_{P_{max,std,MSL}} = W_{mref}^{-1} \cdot a_0^{-1} \cdot \dot{W}_{P_{1,max,std,MSL}} \cdot n_{eng} \tag{3.3-16}$$

Where :

$C_{Pmax,std,MSL}$ is the maximum power coefficient in standard atmosphere at MSL [-]

m_{ref} is the reference mass [kg], from the PFM

W_{mref} is the weight force at m_{ref} [N], see section 3.1

a_0 is the speed of sound in standard atmosphere at MSL [m/s], see section 2.2.1

$\dot{W}_{P_{1,max,std,MSL}}$ is the maximum one-engine power in standard atmosphere at MSL [W], from the PEM where it is expressed in [hp]

n_{eng} is the number of engines of the aircraft [-], from the PFM

The power coefficient in standard atmosphere at MSL is then obtained by:

$$C_{P_{std,MSL}} = C_{P_{max,std,MSL}} \cdot \delta_T \tag{3.3-17}$$

Where :

$C_{Pstd,MSL}$ is the power coefficient in standard atmosphere at MSL [-]

$C_{Pmax,std,MSL}$ is the maximum power coefficient in standard atmosphere at MSL [-]

δ_T is the throttle parameter [-], see section 3.3.3.3

For turbocharged piston engines, the turbocharger can maintain the sea level admission pressure as long as the air density is greater than a minimum density ρ_{turbo}. This air density limit is usually expressed through the equivalent maximum density altitude $H_{\rho,turbo}$, which is the altitude where the air density equals ρ_{turbo} in standard atmosphere. Consequently, at density altitudes lower than $H_{\rho,turbo}$ (i.e. altitudes where $\rho \geq \rho_{turbo}$ and $\sigma \geq \sigma_{\rho,turbo}$), the power coefficient in standard atmosphere at MSL can be maintained, but at density altitudes higher than $H_{\rho,turbo}$ (i.e. altitudes where $\rho < \rho_{turbo}$ and $\sigma < \sigma_{turbo}$), the power coefficient diminishes with altitude:

$$C_P = \begin{cases} C_{P_{std,MSL}} & \text{when } \sigma \geq \sigma_{\rho,turbo} \\ \min\left(C_{P_{std,MSL}}, C_{P_{max,std,MSL}} \cdot \delta \cdot \delta_{\rho_{turbo}}^{-1} \cdot \theta^{-\frac{1}{2}} \cdot \theta_{\rho_{turbo}}^{\frac{1}{2}} \right) & \text{when } \sigma < \sigma_{\rho,turbo} \end{cases} \tag{3.3-18}$$

Where :

$C_{Pstd,MSL}$ is the power coefficient in standard atmosphere at MSL [-]

$C_{Pmax,std,MSL}$ is the maximum power coefficient in standard atmosphere at MSL [-]

δ is the pressure ratio for the current atmospheric conditions and altitude [-]

θ is the temperature ratio for the current atmospheric conditions and altitude [-]

σ is the density ratio for the current atmospheric conditions and altitude [-]

$\delta_{\rho,turbo}$ is the pressure ratio at the altitude where $\rho = \rho_{turbo}$ in the current atmospheric conditions [-]

$\theta_{\rho,turbo}$ is the temperature ratio at the altitude where $\rho = \rho_{turbo}$ in the current atmospheric conditions [-]

$\sigma_{\rho,turbo}$ is the density ratio $\frac{\rho_{turbo}}{\rho_0}$ [-]

For atmospheric piston engines, in which no turbocharger is present, $H_{\rho,turbo}$ is set to zero.

3.3.3.3. Piston rating model

A piston engine is usually operated by direct control of the throttle, so only a basic rating model is defined. The rating model for the piston engine provides the throttle parameter δ_T as a function of the rating:

$$\delta_T = \begin{cases} 1 & \text{when rating is MCMB or MCRZ} \\ 0 & \text{when rating is idle} \end{cases} \tag{3.3-19}$$

3.3.4. Propeller efficiency

The propeller efficiency η of a piston or turboprop engine can be approximated through the momentum theory:

$$\eta = 2 \cdot \eta_{max} \cdot \left[1 + \left(1 + 2 \cdot \eta \cdot \frac{\dot{W}_P}{n_{eng}} \cdot \left(\sigma \cdot \rho_0 \cdot D_P^2 \cdot \frac{\pi}{4} \cdot V_{TAS}^3 \right)^{-1} \right)^{\frac{1}{2}} \right]^{-1} \tag{3.3-20}$$

Where :

η_{max} is the maximum empirical efficiency of the propeller [-], from the PEM or TPM

$\dot{W}_P$ is the all-engine power [W], see section 3.3.3.2

n_{eng} is the number of engines of the aircraft [-], from the PFM

σ is the density ratio for the current atmospheric conditions and altitude [-]

ρ_0 is the air density in standard atmosphere at MSL [kg/m^3], see section 2.2.1

D_P is the propeller diameter [m], from the PEM or TPM

V_{TAS} is the true airspeed [m/s]

Expression (3.3-20) can be reordered into a third degree polynomial expression in η, and η can then be determined by finding the roots of the resulting expression:

$$2 \cdot \frac{\dot{W}_P}{n_{eng}} \cdot \left(\sigma \cdot \rho_0 \cdot D_P^2 \cdot \pi \cdot V_{TAS}^3 \cdot \eta_{max} \right)^{-1} \cdot \eta^3 + \eta - \eta_{max} = 0 \tag{3.3-21}$$

Appendix A presents one possible way of finding the roots of this expression and determining the propeller efficiency.

3.4. FUEL CONSUMPTION

The BADA Family 4 model provides three separate fuel consumption models as part of the Propulsive Forces Model (PFM), depending on the type of engine:

- turbofan: TurboFan Model (TFM),
- turboprop: TurboProp Model (TPM),
- piston: Piston Engine Model (PEM).

Each model includes the contribution from all engines and provides the fuel consumption as a function of airspeed, throttle parameter and atmospheric conditions. The general formulation of the fuel consumption, F [kg/s], is:

$$F = \begin{cases} TFA \cdot 60^{-1} & \text{during taxi (if TFA is defined)} \\ \delta \cdot \theta^{\frac{1}{2}} \cdot W_{mref} \cdot a_0 \cdot L_{HV}^{-1} \cdot C_F & \text{otherwise} \end{cases} \quad (3.4\text{-}1)$$

Where :

TFA is the taxi fuel allowance [kg/min], from the PFM

δ is the pressure ratio [-]

θ is the temperature ratio [-]

m_{ref} is the reference mass [kg], from the PFM

W_{mref} is the weight force at m_{ref} [N], see section 3.1

a_0 is the speed of sound at MSL in standard atmosphere [m/s], see section 2.2.1

L_{HV} is the fuel lower heating value [m^2/s^2], from the PFM

C_F is the fuel coefficient [-], see sections 3.4.1, 3.4.2 and 3.4.3

The subsections below provide the additional formulas of the fuel consumption model for each of the engine types.

3.4.1. Turbofan fuel coefficient

This section provides the formulas to compute the fuel coefficient used in formula (3.4-1), depending on the engine rating (see section 3.3.1).

The fuel coefficient C_F is determined by:

$$C_F = \begin{cases} C_{F,idle} & \text{when idle rating is used} \\ \max\left(C_{F,gen}, C_{F,idle}\right) & \text{when a non-idle rating or no rating is used} \end{cases} \quad (3.4\text{-}2)$$

Where :

$C_{F,idle}$ is the idle fuel coefficient [-], see section 3.4.1.1

$C_{F,gen}$ is the general fuel coefficient [-], see section 3.4.1.2

3.4.1.1. Idle rating

The idle rating model for the turbofan engine model directly provides the idle fuel coefficient $C_{F,idle}$ as a function of the Mach number and the atmospheric conditions:

$$C_{F,idle} = \begin{pmatrix} fi_1 + fi_2\delta + fi_3\delta^2 \\ + \left(fi_4 + fi_5\delta + fi_6\delta^2\right)\cdot M \\ + \left(fi_7 + fi_8\delta + fi_9\delta^2\right)\cdot M^2 \end{pmatrix} \cdot \delta^{-1}\theta^{-\frac{1}{2}} \tag{3.4-3}$$

Where :

δ is the pressure ratio [-]

θ is the temperature ratio [-]

M is the Mach number [-]

fi_1 to fi_9 are idle rating fuel coefficients [-], from the TFM

3.4.1.2. Non-idle ratings (MCMB, MCRZ, MTKF) and no rating (direct throttle parameter input)

The generalized fuel form for the turbofan engine model provides the general fuel coefficient $C_{F,gen}$ as a function of the Mach number M and the thrust coefficient C_T. $C_{F,gen}$ is calculated as a fourth order polynomial of M with coefficients that are fourth order polynomials of C_T:

$$\begin{aligned} C_{F,gen} = \; & f_1 + f_2C_T + f_3C_T^2 + f_4C_T^3 + f_5C_T^4 \\ & + \left(f_6 + f_7C_T + f_8C_T^2 + f_9C_T^3 + f_{10}C_T^4\right)\cdot M \\ & + \left(f_{11} + f_{12}C_T + f_{13}C_T^2 + f_{14}C_T^3 + f_{15}C_T^4\right)\cdot M^2 \\ & + \left(f_{16} + f_{17}C_T + f_{18}C_T^2 + f_{19}C_T^3 + f_{20}C_T^4\right)\cdot M^3 \\ & + \left(f_{21} + f_{22}C_T + f_{23}C_T^2 + f_{24}C_T^3 + f_{25}C_T^4\right)\cdot M^4 \end{aligned} \tag{3.4-4}$$

Where :

C_T is the thrust coefficient [-], see section 3.3.1.1

M is the Mach number [-]

f_1 to f_{25} are non-idle rating fuel coefficients [-], from the TFM

Note: This model takes up to 25 f_i coefficients to calculate $C_{F,gen.}$ The above equation shows the complete case containing all f_i coefficients. However, the number of coefficients is not fixed and depends on the quantity and quality of the reference data with which the coefficients are identified, together with the modeller preferences. In many occasions this results in simpler expressions where some f_i coefficients are deactivated (i.e. equal to zero).

3.4.2. Turboprop fuel coefficient

This section provides the formulas to compute the fuel coefficient used in formula (3.4-1), depending on the engine rating (see section 3.3.2).

The fuel coefficient C_F is determined by:

$$C_F = \begin{cases} C_{F,idle} & \text{when idle rating is used} \\ \max\left(C_{F,gen}, C_{F,idle}\right) & \text{when a non-idle rating or no rating is used} \end{cases} \quad (3.4\text{-}5)$$

Where :

$C_{F,idle}$ is the idle fuel coefficient [-], see section 3.4.2.1

$C_{F,gen}$ is the general fuel coefficient [-], see section 3.4.2.2

3.4.2.1. Idle rating

The idle rating model for the turboprop engine model directly provides the idle fuel coefficient $C_{F,idle}$ as a function of the Mach number and the atmospheric conditions:

$$C_{F,idle} = \begin{pmatrix} fi_1 + fi_2\delta + fi_3\delta^2 \\ +\left(fi_4 + fi_5\delta + fi_6\delta^2\right)\cdot M \\ +\left(fi_7 + fi_8\delta + fi_9\delta^2\right)\cdot M^2 \\ + fi_{10}\theta + fi_{11}\theta^2 + fi_{12}M\theta + fi_{13}M\delta\theta^{\frac{1}{2}} + fi_{14}M\delta\theta \end{pmatrix} \cdot \delta^{-1}\theta^{-\frac{1}{2}} \quad (3.4\text{-}6)$$

Where :

δ is the pressure ratio [-]

θ is the temperature ratio [-]

M is the Mach number [-]

fi_1 to fi_{14} are idle rating fuel coefficients [-], from the TPM

3.4.2.2. Non-idle ratings (MCMB, MCRZ) and no rating (direct throttle parameter input)

The generalized fuel form for the turboprop engine model provides the general fuel coefficient $C_{F,gen}$ as a function of the Mach number M and the power coefficient C_P. $C_{F,gen}$ is calculated as a fourth order polynomial of M with coefficients that are fourth order polynomials of C_P:

$$\begin{aligned} C_{F,gen} = \; & f_1 + f_2C_P + f_3C_P^2 + f_4C_P^3 + f_5C_P^4 \\ & +\left(f_6 + f_7C_P + f_8C_P^2 + f_9C_P^3 + f_{10}C_P^4\right)\cdot M \\ & +\left(f_{11} + f_{12}C_P + f_{13}C_P^2 + f_{14}C_P^3 + f_{15}C_P^4\right)\cdot M^2 \\ & +\left(f_{16} + f_{17}C_P + f_{18}C_P^2 + f_{19}C_P^3 + f_{20}C_P^4\right)\cdot M^3 \\ & +\left(f_{21} + f_{22}C_P + f_{23}C_P^2 + f_{24}C_P^3 + f_{25}C_P^4\right)\cdot M^4 \end{aligned} \quad (3.4\text{-}7)$$

Where :

C_P is the power coefficient [-], see section 3.3.2.2

M is the Mach number [-]

f_1 to f_{25} are non-idle rating fuel coefficients [-], from the TPM

Note: This model takes up to 25 f_i coefficients to calculate $C_{F,gen}$. The above equation shows the complete case containing all f_i coefficients. However, the number of coefficients is not fixed and depends on the quantity and quality of the reference data with which the coefficients are identified, together with the modeller preferences. In many occasions this results in simpler expressions where some f_i coefficients are deactivated (i.e. equal to zero).

3.4.3. Piston fuel coefficient

The generalized fuel form for the piston engine model provides the fuel coefficient C_F as a function of the power coefficient C_P and the atmospheric conditions:

$$C_F = CPSFC \cdot C_P \cdot \delta^{-1} \cdot \theta^{-\frac{1}{2}} \qquad (3.4\text{-}8)$$

Where :

CPSFC is the power-specific fuel consumption coefficient [-], from the PEM

C_P is the power coefficient [-], see section 3.3.3.2

δ is the pressure ratio [-]

θ is the temperature ratio [-]

4. MOTION AND OPERATIONS

This section defines the system of equations that describes the aircraft motion, as well as different ways of operating the aircraft.

4.1. TOTAL-ENERGY MODEL

BADA models the aircraft motion through the Total-Energy Model (TEM), based on the actions defined in section 3. The following subsections describe the TEM equations and how they can be used to compute aircraft performances such as the rate of climb/descent.

4.1.1. General formulation

The Total-Energy Model equates the rate of work done by forces acting on the aircraft to the rate of increase in potential and kinetic energy, that is:

$$(Th - D) \cdot V_{TAS} = mg_0 \frac{dh}{dt} + mV_{TAS} \frac{dV_{TAS}}{dt} \tag{4.1-1}$$

The symbols are defined below with metric units specified:

Th	-	thrust acting parallel to the aircraft velocity vector	[N]
D	-	aerodynamic drag	[N]
m	-	aircraft mass	[kg]
h	-	geodetic altitude	[m]
g_0	-	gravitational acceleration	[m/s^2], see section 2.2.2
V_{TAS}	-	true airspeed	[m/s]
$\frac{d}{dt}$	-	time derivative	[s^{-1}]

Note that true airspeed is often calculated in knots and altitude calculated in feet thus requiring the appropriate conversion factors.

4.1.2. Aircraft control laws

Without considering the use of devices such as spoilers, leading-edge slats or trailing-edge flaps, there are two independent control inputs available for affecting the aircraft trajectory in the vertical plane. These are the throttle and the elevator.

These inputs allow any two of the three variables of thrust, speed, or rate of climb or descent (ROCD) to be controlled. The other variable is then determined by equation (4.1-1). The three resulting control possibilities are elaborated on below:

(a) Speed and Throttle Controlled - Calculation of ROCD

Assuming that velocity and thrust are independently controlled, then equation (4.1-1) is used to calculate the resulting ROCD. This is a fairly common case for climbs and descents in which the throttle is set to some fixed position (maximum climb thrust or idle for descent) and the speed is maintained at some constant value of calibrated airspeed (CAS) or Mach number.

(b) ROCD and Throttle Controlled - Calculation of Speed

Assuming that the ROCD and thrust are independently controlled, then equation (4.1-1) is used to calculate the resulting speed.

(c) Speed and ROCD Controlled - Calculation of Thrust

Assuming that both ROCD and speed are controlled, then equation (4.1-1) can be used to calculate the necessary thrust. This thrust must be within the available limits for the desired ROCD and speed to be maintained.

In addition, advanced control laws in which the speed is determined according to various optimization criteria are presented in section 5.2.

4.1.3. Calculation of rate of climb/descent

Case (a), above, is the most common such that equation (4.1-1) is most often used to calculate the rate of climb or descent. To facilitate this calculation, equation (4.1-1) can be rearranged as follows:

$$(Th - D) \cdot V_{TAS} = mg_0 \frac{dh}{dt} + m V_{TAS} \left(\frac{dV_{TAS}}{dh}\right)\left(\frac{dh}{dt}\right) \qquad (4.1\text{-}2)$$

Isolating the vertical speed on the left hand side gives:

$$\frac{dh}{dt} = \frac{(Th - D) \cdot V_{TAS}}{mg_0}\left[1 + \left(\frac{V_{TAS}}{g_0}\right)\left(\frac{dV_{TAS}}{dh}\right)\right]^{-1} \qquad (4.1\text{-}3)$$

Vertical speed is defined as the variation with time of the aircraft geodetic altitude h. The assumption of a standard constant gravity field derives in identical geodetic altitude h and geopotential altitude H [RD1].

The ROCD is defined as the variation with time of the aircraft geopotential pressure altitude H_p. It is the preferred way of presenting the performances of an aircraft as it eliminates possible variations caused by the atmospheric conditions:

$$ROCD = \frac{dH_p}{dt} = \frac{T - \Delta T}{T}\,\frac{(Th - D) \cdot V_{TAS}}{mg_0}\left[1 + \left(\frac{V_{TAS}}{g_0}\right)\left(\frac{dV_{TAS}}{dh}\right)\right]^{-1} \qquad (4.1\text{-}4)$$

Where :

T - atmosphere temperature [K];

ΔT - temperature differential [K].

4.1.4. Energy share factor

The last term of equation (4.1-4) can be replaced by an energy share factor [RD1] as a function of the Mach number, f{M}:

$$f\{M\} = \left[1 + \left(\frac{V_{TAS}}{g_0}\right) \cdot \left(\frac{dV_{TAS}}{dh}\right)\right]^{-1} \quad (4.1\text{-}5)$$

This leads to:

$$\frac{dh}{dt} = \left[\frac{(Th - D) \cdot V_{TAS}}{m g_0}\right] f\{M\} \quad (4.1\text{-}6)$$

$$ROCD = \frac{dH_P}{dt} = \frac{T - \Delta T}{T} \left[\frac{(Th - D) \cdot V_{TAS}}{m g_0}\right] f\{M\} \quad (4.1\text{-}7)$$

This energy share factor f{M} specifies how much of the available power is allocated to climb as opposed to acceleration while following a selected speed profile during climb.

For several common flight conditions, equation (4.1-5) can be rewritten as is done below [RD1]:

(a) Constant Mach number in stratosphere (i.e. above tropopause)

$$f\{M\} = 1.0 \quad (4.1\text{-}8)$$

Note that above the tropopause the air temperature and the speed of sound are constant. Maintaining a constant Mach number therefore requires no acceleration and all available power can be allocated to a change in altitude.

(b) Constant Mach number below tropopause:

$$f\{M\} = \left[1 + \frac{\kappa R \beta_{T,<}}{2 g_0} M^2 \frac{T - \Delta T}{T}\right]^{-1} \quad (4.1\text{-}9)$$

In this case, for a typical Mach number of 0.8 the energy share factor allocated to climb is 1.09.

This number is greater than 1 because below the tropopause, the temperature and thus, speed of sound decreases with altitude. Maintaining a constant Mach number during climb thus means that the true airspeed decreases with altitude. Consequently, the rate of climb benefits from not only all the available power but also a transfer of kinetic energy to potential energy.

(c) Constant Calibrated Airspeed (CAS) below tropopause

$$f\{M\} = \left\{1 + \frac{\kappa R \beta_{T,<}}{2 g_0} M^2 \frac{T - \Delta T}{T} + \left(1 + \frac{\kappa - 1}{2} M^2\right)^{\frac{-1}{\kappa - 1}} \left\{\left(1 + \frac{\kappa - 1}{2} M^2\right)^{\frac{\kappa}{\kappa - 1}} - 1\right\}\right\}^{-1} \quad (4.1\text{-}10)$$

In this case the energy share factor is less than one. A Mach number of 0.6 for example yields an energy share factor of 0.85.

This number is less than 1 because as density decreases with altitude, maintaining a constant CAS during climb requires maintaining a continual increase in true airspeed. Thus, some of the available power needs to be allocated to acceleration leaving the remainder for climb.

(d) Constant Calibrated Airspeed (CAS) above tropopause.

$$f\{M\} = \left\{1 + \left(1 + \frac{\kappa - 1}{2}M^2\right)^{\frac{-1}{\kappa-1}} \left\{\left(1 + \frac{\kappa - 1}{2}M^2\right)^{\frac{\kappa}{\kappa-1}} - 1\right\}\right\}^{-1} \tag{4.1-11}$$

This formula is identical to (4.1-10), except that β_T is now null since we are above the tropopause.

The energy share factors given above apply equally well to descent as to climb. The difference being that the available power is negative for descent.

In cases where neither constant Mach number nor constant CAS is maintained, the following energy share factors are used:

- acceleration in climb: $f\{M\} = 0.3$
- deceleration in descent: $f\{M\} = 0.3$
- deceleration in climb: $f\{M\} = 1.7$
- acceleration in descent: $f\{M\} = 1.7$

Note that, for the cases of acceleration in climb or deceleration in descent, the majority of the available power is devoted to a change in speed.

For the cases of deceleration in climb or acceleration in descent, the energy share factor is greater than 1 since the change of altitude benefits from a transfer of kinetic energy.

4.2. OPERATION OF CONFIGURATION PARAMETERS

Several restrictions are imposed on the high-lift devices and the landing gear behavior (in terms of valid fixed positions and transition times among them) by the physical mechanisms that drive them. The Operation of configuration Parameters Model (OPM) defines the valid fixed positions and transition times for any configuration parameter, and it is applicable to both the high-lift devices and the landing gear configuration parameters.

Each modeled mechanism gets assigned a parameter worth 0 in its retracted position; this parameter also gets assigned a value for all other positions where the mechanism can be set in a stable way (see section 3.2.1). The model contains the time that it takes to move any mechanism from any position to the following more deployed one (towards higher parameter values) and to the previous less deployed one (towards lower parameter values).

5. BADA MODEL USE CASES

This section provides guidelines on how to use the BADA actions, motion and operations models, depending on what the user knows about the aircraft (input) and the kind of information the user is looking for (output). The first part will deal with a set of common use cases, corresponding to the way aircraft performance models are already used in ATM applications. The second part will consider advanced use cases that are enabled by the high levels of realism and accuracy of the BADA Family 4 model.

5.1. COMMON USE CASES

5.1.1. Determination of input data

The following input information, if required in the use case of interest, has to be provided by the user to the BADA model:

- Aircraft type and corresponding BADA **aircraft model**
- Current **altitude** of the aircraft
- Current **flight phase** (climb, cruise or descent)

Additional information can either be provided (if known) by the user, or estimated (if not known) by the following process:

1. Is the atmosphere **temperature** known?
    a. Yes: determine the temperature deviation from ISA conditions using the atmosphere model (section 2.2) and the known information about temperature
    b. No: assume ISA conditions
2. Is the aircraft **mass** known?
    a. Yes: -
    b. No: pick any value between OEW and MTOW provided in the BADA aircraft model (section 6.4.1)
3. Is the aircraft **speed** known?
    a. Yes: -
    b. No: determine nominal speed using the Airline Procedure Model (Appendix B)
4. Is the aircraft aerodynamic **configuration** known?
    a. Yes: -
    b. No: determine nominal configuration using the Airline Procedure Model (Appendix B)
5. Is the aircraft **bank angle** known?
    a. Yes: -
    b. No: assume bank angle is zero

5.1.2. Determination of output data

Note: for the sake of simplicity, the aircraft control law in which speed and throttle are known will be assumed in most of this section (see section 4.1.2 for details about control laws).

5.1.2.1. Main process

- If you are interested in the aircraft rate of climb/descent, for example to compute the aircraft trajectory:
  1. The following information is required: aircraft model, altitude, flight phase, temperature, speed, mass, configuration, bank angle.
  2. Estimate the drag using the procedure from section 5.1.2.2,
  3. Estimate the thrust using the procedure from section 5.1.2.3,
  4. Determine the energy share factor using section 4.1.4,
  5. Compute the ROCD using formula (4.1-7).
- If you are interested in the aircraft thrust value, for example to feed a noise model:
  1. The following information is required: aircraft model, altitude, flight phase, temperature, speed.
  2. Estimate the thrust using the procedure from section 5.1.2.3.
- If you are interested in the aircraft fuel consumption, for example to feed an emissions model, or to accurately model the aircraft mass evolution in the computation of the aircraft trajectory:
  1. The following information is required: aircraft model, altitude, flight phase, temperature, speed.
  2. Estimate the fuel consumption using the procedure from section 5.1.2.4.

5.1.2.2. Drag determination process

1. Determine the lift coefficient using section 3.2.2
2. Determine the drag coefficient according to the current configuration using section 3.2.4
3. Determine the drag force using section 3.2.5

5.1.2.3. Thrust determination process

If the aircraft is:

- climbing:
  1. Either choose one of the available engine ratings (MCMB, MCRZ or MTKF) and compute the throttle parameter accordingly (using section 3.3.1.2 for a jet, section 3.3.2.3 for a turboprop and section 3.3.3.3 for a piston), or choose directly a throttle parameter value (acceptable values are those that produce a positive rate of climb and a thrust coefficient comprised between $C_{T,LIDL}$ and $C_{T,MCMB}$),
  2. Compute the thrust coefficient (using section 3.3.1.1.2 for a jet, section 3.3.2.1.2 for a

turboprop and section 3.3.3.1 for a piston),

3. Compute the thrust force using formula (3.3-1).

- descending:
  - to model an idle descent:
    1. Choose the idle engine rating,
    2. Compute the thrust coefficient (using section 3.3.1.1.1 for a jet, section 3.3.2.1.1 for a turboprop and section 3.3.3.1 for a piston),
    3. Compute the thrust force using formula (3.3-1).
  - to model a powered descent:
    1. Choose a throttle parameter value (acceptable values are those that produce a negative rate of climb and a thrust coefficient comprised between $C_{T,LIDL}$ and $C_{T,MCMB}$),
    2. Compute the thrust coefficient (using section 3.3.1.1.2 for a jet, section 3.3.2.1.2 for a turboprop and section 3.3.3.1 for a piston),
    3. Compute the thrust force using section formula (3.3-1).
- levelling-off:
  - to model a level-off segment at constant speed:
    1. Estimate the drag using the procedure from section 5.1.2.2,
    2. Assume that thrust equals drag.
  - to model an acceleration segment, use the same process as when the aircraft is climbing,
  - to model a deceleration segment, use the same process as when the aircraft is descending.

5.1.2.4. Fuel determination process

If the aircraft is:

- climbing:
  1. Either choose one of the available engine ratings (MCMB, MCRZ or MTKF) and compute the throttle parameter accordingly (using section 3.3.1.2 for a jet, section 3.3.2.3 for a turboprop and section 3.3.3.3 for a piston), or choose directly a throttle parameter value (acceptable values are those that produce a positive rate of climb and a thrust coefficient comprised between $C_{T,LIDL}$ and $C_{T,MCMB}$),
  2. Compute the thrust coefficient using section 3.3.1.1.2 for a jet, or the power coefficient using section 3.3.2.2 for a turboprop and section 3.3.3.2 for a piston,
  3. Compute the fuel coefficient (using section 3.4.1.2 for a jet, section 3.4.2.2 for a turboprop and section 3.4.3 for a piston),
  4. Compute the fuel consumption using formula (3.4-1).
- descending:
  - to model an idle descent:
    1. Choose the idle engine rating,

2. Compute the fuel coefficient (using section 3.4.1.1 for a jet, section 3.4.2.1 for a turboprop and section 3.4.3 for a piston),
3. Compute the fuel consumption using formula (3.4-1).

- to model a powered descent:
  1. Choose a throttle parameter value (acceptable values are those that produce a negative rate of climb and a thrust coefficient comprised between $C_{T,LIDL}$ and $C_{T,MCMB}$),
  2. Compute the thrust coefficient using section 3.3.1.1.2 for a jet, or the power coefficient using section 3.3.2.2 for a turboprop and section 3.3.3.2 for a piston,
  3. Compute the fuel coefficient (using section 3.4.1.2 for a jet, section 3.4.2.2 for a turboprop and section 3.4.3 for a piston),
  4. Compute the fuel consumption using formula (3.4-1).

- levelling-off:
  - to model a level-off segment at constant speed:
    1. Estimate the drag using the procedure from section 5.1.2.2,
    2. Assume that thrust equals drag,
    3. Compute the thrust coefficient using formula (3.3-1),
    4. For turboprop (resp. piston) engines, compute the power coefficient using section 3.3.2.1.2 (resp. 3.3.3.1),
    5. Compute the fuel coefficient (using section 3.4.1.2 for a jet, section 3.4.2.2 for a turboprop and section 3.4.3 for a piston),
    6. Compute the fuel consumption using formula (3.4-1).
  - to model an acceleration segment, use the same process as when the aircraft is climbing,
  - to model a deceleration segment, use the same process as when the aircraft is descending.

5.2. ADVANCED USE CASES – OPTIMIZATIONS

Management of aircraft operations in terms of optimization of the flight cost has increasing relevance in the context of today's economical and environmental aspects of ATM systems. This section provides an analysis of optimized flight operations to obtain the equations in which they derive, and describes how to use the BADA Family 4 APM to support modeling and simulation of complex operations.

The following complex operations are considered:

- Cruise management including: maximum range cruise, long range cruise, optimum altitude, holding (maximum endurance cruise),
- Cost management based on Cost Index (CI) in climb, cruise and descent.

Since the computational requirements of such optimizations may be high, some pre-computed results are provided for these complex operations with each BADA aircraft model equipped with turbofan engines (see section 9.8).

5.2.1. Cruise management

The following sub-sections will present several criteria that are commonly used to optimize the cruise phase of a flight depending on the aircraft weight (W), cruise geopotential pressure altitude (H_p) and atmospheric conditions expressed as the temperature deviation (ΔT) from the ISA conditions.

5.2.1.1. Maximum Range Cruise (MRC)

The objective of the MRC is to maximize the flight range for given values of fuel load and atmospheric conditions, usually at constant H_p. The solution is the maximum range Mach number M_{mrc}, which achieves the minimum fuel consumption with respect to distance, or equivalently the maximum distance the aircraft can fly with the given fuel at the given altitude. It is considered as the extreme case of CI cruise management (see section 5.2.2.1) with a CI value equal to zero.

In the BADA MRC optimization procedure, the maximization of flight range at constant H_p is reduced to the maximization of the specific range, SR [NM/kg], with respect to Mach number. The specific range is defined as the distance that can be flown per unit of fuel:

$$SR = -\frac{dr}{dm} = \frac{V_{GS}}{F} \qquad (5.2\text{-}1)$$

Where:

r is the flown ground distance [NM]

m is the aircraft mass [kg]

V_{GS} is the ground speed [kt]

F is the fuel consumption [kg/h]

The BADA MRC procedure is the following:

1. For the given W, H_p and ΔT, find:
$$M' = \left\{ M_i \mid i \in N, M_i \in R, 0 < M_i \leq M_{mo}, \frac{dSR}{dM}(M_i) = 0, \frac{d^2SR}{dM^2}(M_i) < 0 \right\}$$
2. Find M_k, $M_k \in M'$, with the maximum SR value, $SR(M_k) = \max\{SR(M_i) \mid M_i \in M'\}$. The result is the maximum range Mach number M_{mrc}, $M_{mrc} = M_k$.

5.2.1.2. Long Range Cruise (LRC)

In comparison to the MRC, a slight increase in fuel consumption allows a significant increase in Mach number and a reduction in flight time for the same conditions. The specific range of the LRC corresponds to 99% of the specific range of the MRC, and LRC Mach number M_{lrc} for the given W, H_p and ΔT is defined as:

$$0.99 * SR(M_{mrc}) = SR(M_{lrc}), M_{lrc} > M_{mrc} \qquad (5.2\text{-}2)$$

The BADA LRC optimization procedure is based on the BADA MRC optimization procedure, extended to determine M_{lrc} according to its definition.

5.2.1.3. Holding, Maximum Endurance Cruise (MEC)

When holding is requested, the knowledge of maximum holding time (endurance) is of most importance for the decision making process. It is defined as the maximization of the time an aircraft can remain airborne with a given amount of fuel, i.e. the fuel consumption is minimized with respect to time. As for MRC and LRC, the most important case is the constant H_p case and the minimization of fuel consumption with respect to the Mach number for the given W and atmospheric conditions. The result of the minimization is the holding speed, called holding Mach number M_{mec}, defined as the speed at minimum allowable fuel flow (flame-out). Since this speed falls into the speed-instability region (near the minimum drag speed and the maximum lift to drag ratio speed), it is usually increased slightly to provide easier aircraft control.

The BADA Holding optimization procedure is the following:

1. For the given W, H_p and ΔT, find $M' = \left\{ M_i \mid i \in N, M_i \in R, 0 < M_i \leq M_{mo}, \frac{dF}{dM}(M_i) = 0 \right\}$
2. Find M_k, $M_k \in M'$, with the minimum value of F, $F(M_k) = \min\{F(M_i) \mid M_i \in M'\}$. The result is the maximum endurance Mach number M_{mec}, $M_{mec} = M_k$.
3. At the end of the procedure M_{mec} is slightly increased: $M_{mec} = M_{mec} + \Delta M$.

5.2.1.4. Optimum Altitude

The optimum altitude $H_{p,opt}$ is defined as the geopotential pressure altitude at which the specific range is maximum for given values of Mach number, aircraft weight and atmospheric conditions. This may be considered as an MRC case with constant M instead of constant H_p. The optimum altitude corresponds to the maximum lift to drag ratio (L/D) or the maximum lift coefficient to drag coefficient ratio (C_L/C_D): the optimum altitude optimization procedure may thus be reduced to the maximization of the lift to drag ratio.

The BADA optimum altitude optimization procedure is the following:

1. For the given W, M and ΔT, find $H_p' = \left\{ H_{p,i} \mid i \in N, H_{p,i} \in R, 0 < H_{p,i} \leq H_{mo}, \frac{dSR}{dH_p}(H_{p,i}) = 0 \right\}$ where H_{mo} is the maximum operating altitude.
2. Find $H_{p,k}$, $H_{p,k} \in H_p'$, with the maximum value of SR, $SR(H_{p,k}) = \max\{SR(H_{p,i}) \mid H_{p,j} \in H_p'\}$. The result is the optimum altitude $H_{p,opt}$: $H_{p,opt} = H_{p,k}$.

5.2.2. Cost management

The objective of the cost management is the minimization of the total flight cost C_t (also called direct operating cost):

$$\begin{aligned} C_t &= C_{fx} + C_v \\ &= C_{fx} + C_f\,\Delta m + C_h\,\Delta t \\ &= C_{fx} + C_f\,(\Delta m + \Delta t\, C_h / C_f) \end{aligned} \tag{5.2-3}$$

Where:

C_{fx} is the fixed cost [€]

C_v is the variable cost [€]

C_h is the time-related cost [€/min]

C_f is the fuel-related cost [€/kg]

Δt is the period of time [min]

Δm is the fuel consumed [kg]

As C_{fx} and C_f do not change frequently, at least during a flight, the minimization can be done by the optimization of C_h/C_f ratio.

The cost index, CI [kg/min], is defined as the ratio between time and fuel related costs:

$$CI = \frac{C_h}{C_f} \tag{5.2-4}$$

The fundamental rationale of the cost index concept is to achieve minimum flight trip cost by means of a trade-off between operating costs per hour and incremental fuel burn. Airlines usually define cost indices per flight leg taking into account the airline cost structure and operating priorities. They are seasonally readjusted to account for recurring fluctuations. The determination of CI varies across airlines and will not be considered in this document.

The range of CI values usually varies from 0 to 99 or 999 [kg/min] depending on the aircraft manufacturer. Extreme values CI = 0 ($C_h << C_f$) and CI = CI_{max} ($C_h >> C_f$) represent minimum fuel mode and minimum flight time mode respectively. CI = 20 may be interpreted as the cost of 20 kilograms of fuel being equal to the cost of 1 flight minute.

For a predefined value of CI, minimum flight cost is achieved by adopting an operational speed that properly proportions both fuel and time related costs. The CI is entered into the aircraft Flight Management Computer (FMC) which calculates the most economic (ECON) speed for each phase of a specific flight. The identification of the ECON speed is considered as an advanced function that an APM can provide. The procedure and equations to obtain it are presented hereafter.

5.2.2.1. Cost Index cruise management

CI cruise management is based on the determination of an optimum cruise speed, called economic Mach number M_{ECON}. M_{ECON} minimizes the total cost of the phase, C_t, for given values of: CI, aircraft weight (W), cruise geopotential pressure altitude (H_p) and atmospheric conditions expressed as the temperature deviation (ΔT) from the ISA conditions – in the sake of simplicity, the effect of wind and pressure deviation are neglected in this document.

The minimization of the total cost is reduced to the minimization of the economy cruise cost function (ECCF):

$$ECCF = \frac{C_v}{C_f \Delta r} = \frac{CI + F}{V_{GS}} \tag{5.2-5}$$

Where:

C_v is the variable cost [€]

C_f is the fuel-related cost [€/kg]

F is the fuel consumption [kg/min]

Δr is the flown distance [NM]

V_{GS} is the ground speed [kt]

The solution of the minimization of ECCF is the economic Mach number, M_{ECON}.

Finally, the procedure to determine M_{ECON} is the following:

1. For the given CI, W, H_p and ΔT, find $M' = \left\{ M_i \mid i \in N, M_i \in R, 0 < M_i \leq M_{mo}, \frac{dECCF}{dM}(M_i) = 0 \right\}$ where M_{mo} is the maximum operating Mach number.
2. Find M_k, $M_k \in M'$, with the minimum ECCF value, $ECCF(M_k) = \min\{ECCF(M_i) \mid M_i \in M'\}$. The resulting Mach number is M_{ECON}, $M_{ECON} = M_k$.

5.2.2.2. Cost Index climb management

CI climb management is based on the determination of an optimum climb speed schedule, composed of economic climb speed CAS'_{ECON} (calibrated airspeed) and Mach number M'_{ECON} that form the economic climb speed schedule. For typical jet aircraft, this climb speed schedule takes the form 250/CAS'_{ECON}/M'_{ECON}, i.e. 250 kt CAS below FL100, CAS'_{ECON} above FL100 until transition with M'_{ECON}. This climb speed schedule minimizes the total cost of the climb phase for given values of: CI, aircraft weight, cruise geopotential pressure altitude ($H_{p,CR}$) and atmospheric conditions. The optimization is performed over a specified distance, called range, which includes the climb phase to the given cruise altitude and the economy cruise phase from the top of climb (TOC) to the end of the specified range. M'_{ECON} is assumed to coincide with M_{ECON} from the cruise phase at the given cruise altitude, appropriate aircraft weight and atmospheric conditions.

The extreme values of CI, CI = 0 and CI = CI_{max} represent maximum rate of climb (minimization of fuel consumption) and maximum climb speed (minimization of flight time: $CAS'_{ECON} = V_{mo}$, where V_{mo} is the maximum operating speed) respectively.

The procedure to determine the economic climb speed schedule in BADA is the following:

1. Definition of three climb segments (named after their respective speeds CAS_1, CAS_2, M) and one economy cruise segment (M) across a specified range, for example 500 [nm].
2. Definition of the total cost function as:
   cost = cost(CAS_1) + cost(CAS_2) + cost_climb(M) + cost_cruise(M)
3. Determination of CAS_2 and M (CAS_1 = 250) and calculation of the total cost.
4. Minimization of the cost function and determination of CAS'_{ECON} and M'_{ECON}.

The cost function, for a given airspeed in the segment, is defined as: cost(CAS/M) = CI Δt + Δm [kg], where Δt and Δm are the time spent and the fuel consumed in the segment for a given CAS/M.

5.2.2.3. Cost Index descent management

CI descent management is based on the determination of an optimum descent speed schedule, composed of economic descent speed CAS''_{ECON} (calibrated airspeed) and Mach number M''_{ECON} that form the economic descent speed schedule. For typical jet aircraft, this descent speed schedule takes the form 250/CAS''_{ECON}/M''_{ECON}, i.e. 250 kt CAS below FL100, CAS''_{ECON} above FL100 until transition with M''_{ECON}. This descent speed schedule minimizes the total cost of the descent phase for given values of: CI, aircraft weight, cruise geopotential pressure altitude ($H_{p,CR}$) and atmospheric conditions. The optimization is performed over a specified range which includes the economy cruise phase at the given cruise altitude up to the top of descent (TOD) and the descent phase from the TOD to the end of the specified range. M''_{ECON} is assumed to coincide with M_{ECON} from the cruise phase at the given cruise altitude, appropriate aircraft weight and atmospheric conditions.

The extreme values of CI, CI = 0 and CI = CI_{max} represent minimum descent speed (minimization of fuel consumption) and maximum descent speed ($CAS''_{ECON} = V_{mo}$, reduction of flight time).

Note that the descent optimization problem is less complex than the climb one: since the aircraft weight at the beginning of the cruise phase is now given, $M_{ECON} = M''_{ECON}$ can here be determined directly from the cruise segment using the economy cruise procedure.

The procedure to determine the economic descent speed schedule in BADA is the following:

1. Definition of one economy cruise segment (M) and three descent segments (CAS_1, CAS_2, M) across a specified range, for example 400 [nm].
2. Definition of the cost function (the same as for the climb management).
3. Determination of CAS_2 and M (C_1 = 250), calculation of the cost.
4. Minimization of the cost function and determination of CAS''_{ECON} and M''_{ECON}.

6. LIMITATIONS

The aerodynamic and propulsive forces estimated with the drag and thrust models described in sections 3.2 and 3.3 are only valid if the aircraft respects certain limitations, without which those models do not reflect valid physical principles and as such cannot be used to model the aircraft. The Aircraft Limitations Model (ALM) is divided into five types of limitations: geometric (section 6.1), kinematic (section 6.2), buffet (section 6.3), dynamic (section 6.4), and environmental (section 6.5).

6.1. GEOMETRIC LIMITATIONS

The Geometric Limitations Model (GLM) contains the maximum geopotential pressure altitude for which the aircraft is certified. In general, it depends on whether the high-lift devices are deployed or not:

$$H_{P\max} = \begin{cases} h_{MO} & \text{when } \delta_{HL}=0 \\ mfa & \text{when } \delta_{HL}>0 \end{cases} \qquad (6.1\text{-}1)$$

Where :

H_{Pmax} is the maximum geopotential pressure altitude [ft]

h_{MO} is the maximum operating altitude [ft], from the GLM

mfa is the maximum high-lift devices use altitude [ft], from the GLM

δ_{HL} is the position of the high-lift devices [-], see section 3.2.1

Note: if mfa is not defined, then mfa shall be set to h_{MO} in formula (6.1-1).

6.2. KINEMATIC LIMITATIONS

The Kinematic Limitations Model (KLM) contains the **maximum** calibrated airspeed $V_{CAS,max}$ and Mach number M_{max} for which the aircraft is certified. In general, they depend on the aerodynamic configuration:

$$V_{CAS,max} = \begin{cases} V_{MO} & \text{when } \delta_{HL}=0 \text{ and } \delta_{LG}=0 \\ V_{FE,\delta_{HL}} & \text{when } \delta_{HL}>0 \text{ and } \delta_{LG}=0 \\ V_{LE} & \text{when } \delta_{HL}=0 \text{ and } \delta_{LG}=1 \\ \min(V_{FE,\delta_{HL}}, V_{LE}) & \text{when } \delta_{HL}>0 \text{ and } \delta_{LG}=1 \end{cases} \qquad (6.2\text{-}1)$$

Where :

$V_{CAS,max}$ is the maximum calibrated airspeed [kt]

V_{MO} is the maximum operating speed [kt], from the KLM

$V_{FE,\delta HL}$ is the high-lift placard speed of the current high-lift devices position [kt], from the AFCM section corresponding to δ_{HL}

V_{LE} is the maximum speed with landing gear extended [kt], from the KLM

δ_{HL} is the position of the high-lift devices [-], see section 3.2.1

δ_{LG} is the position of the landing gear [-], see section 3.2.1

Note: if $V_{FE,\delta HL}$ (resp. V_{LE}) is not defined, then $V_{FE,\delta HL}$ (resp. V_{LE}) shall be set to V_{MO} in formula (6.2-1).

$$M_{max} = \begin{cases} M_{MO} & \text{when } \delta_{LG}=0 \\ M_{LE} & \text{when } \delta_{LG}=1 \end{cases} \tag{6.2-2}$$

Where :

M_{max} is the maximum Mach number [-]

M_{MO} is the maximum operating Mach number [-], from the KLM

M_{LE} is the maximum Mach number with landing gear extended [-], from the KLM

δ_{LG} is the position of the landing gear [-], see section 3.2.1

Note 1: if M_{MO} is not defined, then Mach number limitations do not apply to the aircraft and only $V_{CAS,max}$ shall be considered.

Note 2: if M_{LE} is not defined and M_{MO} is defined, then M_{LE} shall be set to M_{MO} in formula (6.2-2).

The **minimum** Mach number M at which the aircraft can operate is modeled through the buffet model (section 6.3).

6.3. BUFFET LIMITATIONS

The Buffet Limitations Model (BLM) determines the maximum lift coefficient $C_{L,max}$ at which the aircraft can operate, depending on the aircraft aerodynamic configuration (see section 3.2.1).

6.3.1. Maximum lift coefficient in clean configuration

For the clean configuration, the $C_{L,max}$ model is based on a fourth order polynomial of the Mach number:

$$C_{L_{poly}}(M) = bf_1 + bf_2 \cdot M + bf_3 \cdot M^2 + bf_4 \cdot M^3 + bf_5 \cdot M^4 \tag{6.3-1}$$

Where :

M is the Mach number [-]

bf_1 to bf_5 are coefficients from the BLM [-]

The domain of applicability of this polynomial is defined by a lower boundary, M_{min}, and an upper boundary, M_{max}, that are specific to each aircraft model. For Mach numbers lower than M_{min} (resp. higher than M_{max}), the value of $C_{L,max}$ is based on a linear interpolation (resp. extrapolation), resulting in the following generic $C_{L,max}$ model:

$$C_{L_{max}} = \begin{cases} C_{L,Mach0} + \dfrac{M}{M_{min}} \cdot \left(C_{L_{poly}}(M_{min}) - C_{L,Mach0}\right) & \text{when } M<M_{min} \\ C_{L_{poly}}(M) & \text{when } M_{min} \le M \le M_{max} \\ C_{L_{poly}}(M_{max}) + (M - M_{max}) \cdot C_{L_{der}}(M_{max}) & \text{when } M>M_{max} \end{cases} \tag{6.3-2}$$

Where :

$C_{L,max}$ is the maximum lift coefficient [-]

M is the Mach number [-]

M_{min} is the lower boundary of the domain of applicability of $C_{L,poly}$ [-], from the BLM

M_{max} is the upper boundary of the domain of applicability of $C_{L,poly}$ [-], from the BLM

$C_{L,Mach0}$ is the maximum lift coefficient at M=0 [-], from the BLM

$C_{L,der}$ is the derivative of $C_{L,poly}$ used to compute the slope of the extrapolation:

$$C_{L_{der}}(M) = \frac{dC_{L_{poly}}}{dM}(M) = bf_2 + 2 \cdot bf_3 \cdot M + 3 \cdot bf_4 \cdot M^2 + 4 \cdot bf_5 \cdot M^3 \qquad (6.3\text{-}3)$$

where bf_2 to bf_5 are coefficients from the BLM [-]

Note: the clean $C_{L,max}$ model may not be available for some aircraft, in which case the procedure of section 6.3.2 is to be followed for the clean configuration. This is in particular the case of all turboprop and piston aircraft.

6.3.2. Maximum lift coefficient in non-clean configurations

For non-clean configurations, $C_{L,max}$ is defined for each high-lift devices and landing gear position as a constant, without any influence from the Mach number:

$$C_{L_{max}} = C_{L_{max},,\delta_{HL},\delta_{LG}} \qquad (6.3\text{-}4)$$

Where :

$C_{L,max}$ is the maximum lift coefficient [-]

$C_{L,max,\delta HL,\delta LG}$ is the maximum lift coefficient of the current aerodynamic configuration [-], from the AFCM section corresponding to δ_{HL} and δ_{LG}

δ_{HL} is the position of the high-lift devices [-], see section 3.2.1

δ_{LG} is the position of the landing gear [-], see section 3.2.1

6.3.3. Minimum Mach number

The minimum Mach number can be determined for given aircraft weight, load factor, and geopotential pressure altitude as the minimum Mach number at which the lift coefficient, obtained with formula (3.2-1), is lower than the maximum lift coefficient, obtained with formula (6.3-2) or (6.3-4).

6.3.4. Impact of buffet on maximum Mach number and altitude

When the clean $C_{L,max}$ model is available for an aircraft, additional limitations may apply to the maximum Mach number and altitude: the buffet-limited maximum Mach number (resp. altitude) can be determined for given aircraft weight, load factor, and geopotential pressure altitude (resp. Mach number) as the maximum Mach number (resp. altitude) at which the lift coefficient, obtained with formula (3.2-1), is lower than the maximum lift coefficient, obtained with formula (6.3-2).

Additional details can be found in section 6.2 of [RD1].

6.4. DYNAMIC LIMITATIONS

The Dynamic Limitations Model (DLM) contains the minimum and maximum weights[3] allowed by the aircraft, plus the maximum and minimum load factors.

6.4.1. Mass limitations

Up to seven mass values are specified for each aircraft:

OEW	- Operating Empty Weight [kg]
MTW	- Maximum Taxi Weight [kg]
MTOW	- Maximum Take-Off Weight [kg]
MLW	- Maximum Landing Weight [kg]
MZFW	- Maximum Zero-Fuel Weight [kg]
MPL	- Maximum Payload [kg]
MFL	- Maximum Fuel Load [kg]

Note that the specified mass limits are taken from aircraft performance reference data which is available in the BADA library. Depending on specific aircraft certified limitations, a particular aircraft version of a given aircraft type (model) may have different limits. More details on the way the mass limits are selected in BADA are provided in [RD4].

6.4.2. Load factor limitations

The maximum and minimum load factors depend on whether the high-lift devices are deployed or not:

$$n_{max} = \begin{cases} n_1 & \text{when } \delta_{HL}=0 \\ n_{f1} & \text{when } \delta_{HL}>0 \end{cases} \quad (6.4\text{-}1)$$

$$n_{min} = \begin{cases} n_3 & \text{when } \delta_{HL}=0 \\ n_{f3} & \text{when } \delta_{HL}>0 \end{cases} \quad (6.4\text{-}2)$$

Where :

n_{max} is the maximum load factor [-]

n_{min} is the minimum load factor [-]

n_1, n_3, n_{f1} and n_{f3} are load factor coefficients [-], from the DLM

δ_{HL} is the position of the high-lift devices [-], see section 3.2.1

6.5. ENVIRONMENTAL LIMITATIONS

The Environmental Limitations Model (ELM) provides the maximum and minimum temperature deviations under which the aircraft can be operated, as a function of geopotential pressure altitude. Two different environmental envelopes are provided, depending on the type of operation: the first

[3] Although they are labeled as weights to follow the common terminology, those values are in fact masses, expressed in mass units.

one is to be used in flight, and the second one for landing and take-off, where more restrictive limitations are in place.

$$\Delta T_{max} = \begin{cases} T_{max,f}(H_p) & \text{in flight} \\ T_{max,g}(H_p) & \text{during take-off and landing} \end{cases} \tag{6.5-1}$$

$$\Delta T_{min} = \begin{cases} T_{min,f}(H_p) & \text{in flight} \\ T_{min,g}(H_p) & \text{during take-off and landing} \end{cases} \tag{6.5-2}$$

Where :

ΔT_{max} is the maximum temperature deviation [K]

ΔT_{min} is the minimum temperature deviation [K]

H_P is the geopotential pressure altitude [ft]

$T_{max,f}$, $T_{max,g}$, $T_{min,f}$, $T_{min,g}$ are tables from the ELM

The $T_{max,f}$, $T_{max,g}$, $T_{min,f}$, $T_{min,g}$ tables provide maximum and minimum temperature deviations at key altitudes, between which a linear interpolation has to be used. Figure 2 presents an example of determination of maximum and minimum temperature deviations: the two black lines represent the values from the ELM tables $T_{min,f}$ (left) and $T_{max,f}$ (right), and the red line shows how to determine the interpolated values at an intermediate pressure altitude.

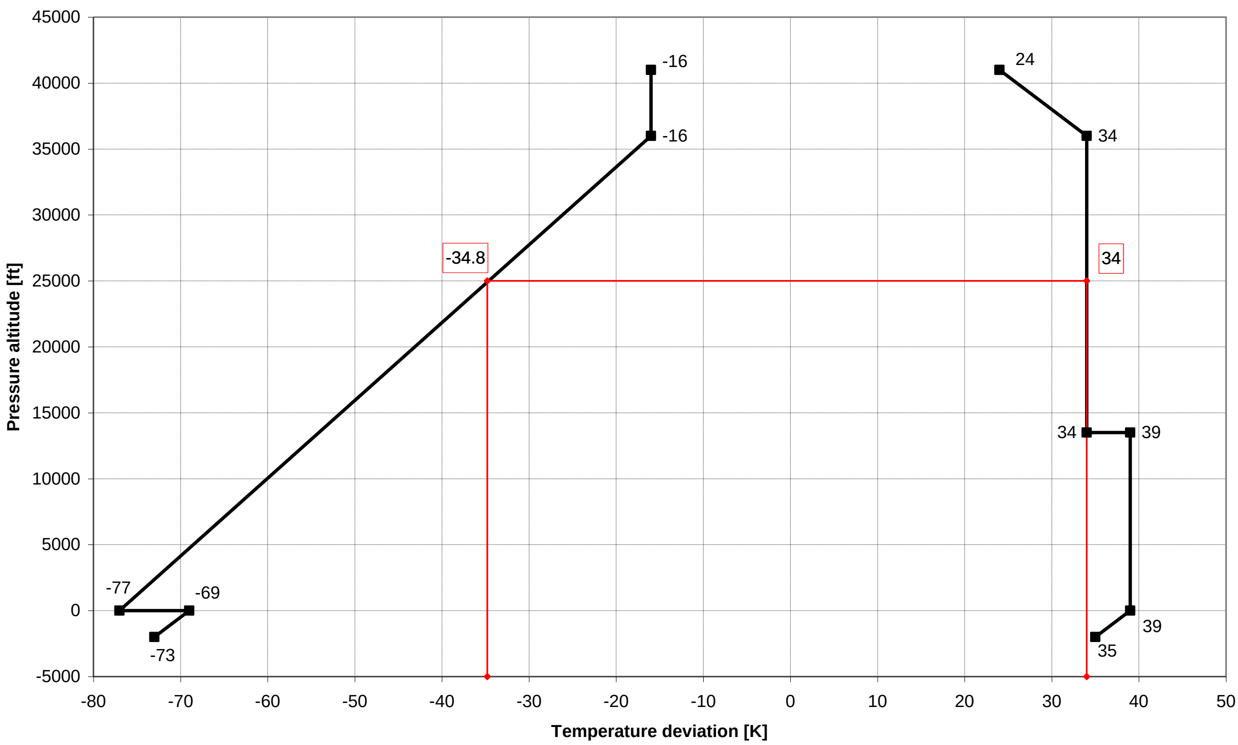

Figure 2 – Determination of environmental envelope at a given pressure altitude

7. AIRCRAFT CHARACTERISTICS

Each aircraft model in BADA is characterized with a set of coefficients, called Aircraft Characteristics, which are used by the APM and ARPM. The following subsections will present the remaining coefficients that have not been described in the previous sections, as well as a summary of all the coefficients that can be provided in a BADA Family 4 aircraft model.

7.1. ADDITIONAL PARAMETERS

In addition to the parameters used in the formulas of the previous sections, some supplementary parameters are provided for each aircraft type and will be presented in this subsection.

7.1.1. Engine type

Each supported aircraft type is is equipped with engines whose type can be one of three values:

- Jet (also called Turbofan in this document)
- Turboprop
- Piston

7.1.2. ICAO information

Each supported aircraft type is assigned by the International Civil Aviation Organisation (ICAO) a 4-character designation code, called designator [RD2].

ICAO also associates a wake turbulence category (WTC) to each aircraft type designator [RD2], which can be one of four values:

- J : jumbo
- H : heavy
- M: medium
- L : light

Both the ICAO designator and WTC are provided for each BADA model.

7.1.3. Ground Movement

Four values are specified that can be of use when simulating ground movements. These parameters are:

- TOL: FAR Take-Off Length [m] with MTOW on a dry, hard, level runway under ISA conditions and no wind.
- LDL: FAR Landing Length [m] with MLW on a dry, hard, level runway under ISA conditions and no wind.
- span: aircraft wingspan [m]
- length: aircraft length [m]

7.2. SUMMARY OF PERFORMANCE PARAMETERS

A summary of the parameters specified for each BADA Family 4 model is supplied in Table 1. The location of each parameter in the XML file (see section 9.4) is indicated, together with the symbol used to represent this parameter in the BADA expressions where it appears (if any). Detailed information on how these parameters have been obtained during the process of BADA aircraft model identification using the aircraft performance reference documents is provided in [RD4].

Important notice: Parameters listed in Table 1 should not be modified by the user as such modifications may impact the validity of the model.

Table 1 – BADA Performance Parameters Summary

Model or Category	Symbol	Unit	XML location		Description
			Element	Attribute	
Aircraft type	-	-	ACM\model	-	name of the aircraft model
	-	-	ACM\type	-	engine type (Jet, Turboprop or Piston)
	-	-	ACM\engine	-	engine name
	-	-	ACM\description	-	additional information on the model
ICAO information	-	-	ACM\ICAO\designator	-	aircraft type designator
	-	-	ACM\ICAO\WTC	-	wake turbulence category (J, H, M or L)
AFCM	S	m^2	ACM\AFCM\S	-	wing reference area
	-	-	ACM\AFCM\Configuration	HLid	identifier of the high-lift devices configuration
	δ_{HL}	-	ACM\AFCM\Configuration\HLPosition	-	position of the high-lift devices
	-	-	ACM\AFCM\Configuration\name	-	name of the high-lift devices configuration
	$d_1 ... d_{15}$	-	ACM\AFCM\Configuration\LGUP\DPM_clean\CD_clean	-	clean drag model coefficients
	scalar	-	ACM\AFCM\Configuration\LGUP\DPM_clean\scalar	-	clean drag model scaling factor
	M_{max}	-	ACM\AFCM\Configuration\LGUP\DPM_clean\M_max	-	highest Mach number used for identification
	$d_{1,\delta HL,\delta LG}$... $d_{3,\delta HL,\delta LG}$	-	ACM\AFCM\Configuration\LG[UP|DN]\DPM_nonclean\CD_nonclean	-	non-clean drag model coefficients
OPM	-	-	ACM\OPM\HLO\Transition\from	-	identifier of the initial high-lift devices configuration
	-	-	ACM\OPM\HLO\Transition\to	-	identifier of the final high-lift devices configuration
	-	s	ACM\OPM\HLO\Transition\time	-	transition time from initial to final high-lift devices configuration
	-	-	ACM\OPM\LGO\Transition\from	-	identifier of the initial landing gear configuration
	-	-	ACM\OPM\LGO\Transition\to	-	identifier of the final landing gear configuration
	-	s	ACM\OPM\LGO\Transition\time	-	transition time from initial to final landing gear configuration
PFM	L_{HV}	m^2/s^2	ACM\PFM\LHV	-	fuel lower heating value
	m_{ref}	kg	ACM\PFM\MREF	-	reference mass
	-	kg/m^3	ACM\PFM\rho	-	fuel density
	n_{eng}	-	ACM\PFM\n_eng	-	number of engines
	TFA	kg/min	ACM\PFM\TFA	-	taxi fuel allowance (all engines)
TFM	$ti_1..ti_{12}$	-	ACM\PFM\TFM\LIDL\CT	-	turbofan idle rating thrust coefficients
	$fi_1..fi_9$	-	ACM\PFM\TFM\LIDL\CF	-	turbofan idle rating fuel coefficients
	$a_1..a_{36}$	-	ACM\PFM\TFM\CT	-	turbofan non-idle rating thrust coefficients
	$f_1..f_{25}$	-	ACM\PFM\TFM\CF	-	turbofan non-idle rating fuel coefficients

Model or Category	Symbol	Unit	XML location		Description
			Element	Attribute	
	ΔT_{kink}	K	ACM\PFM\TFM\M[CMB|CRZ|TKF]\kink	-	turbofan kink point
	$b_1..b_{36}$	-	ACM\PFM\TFM\M[CMB|CRZ|TKF]\flat_rating	-	turbofan flat-rated area throttle coefficients
	$c_1..c_{45}$	-	ACM\PFM\TFM\M[CMB|CRZ|TKF]\temp_rating	-	turbofan temperature-rated area throttle coefficients
	-	-	ACM\PFM\TFM\throttle\low	-	turbofan low throttle parameter
	-	-	ACM\PFM\TFM\throttle\high	-	turbofan high throttle parameter
TPM	$ti_1..ti_{32}$	-	ACM\PFM\TPM\LIDL\CT	-	turboprop idle rating thrust coefficients
	$fi_1..fi_{14}$	-	ACM\PFM\TPM\LIDL\CF	-	turboprop idle rating fuel coefficients
	$a_1..a_{36}$	-	ACM\PFM\TPM\CP	-	turboprop non-idle rating power coefficients
	$f_1..f_{25}$	-	ACM\PFM\TPM\CF	-	turboprop non-idle rating fuel coefficients
	$p_1..p_{36}$	-	ACM\PFM\TPM\M[CMB|CRZ]\rating	-	turboprop non-idle rating throttle coefficients
	$\dot{W}_{P_{max}}$	W	ACM\PFM\TPM\M[CMB|CRZ]\max_power	-	maximum all-engine power
	η_{max}	-	ACM\PFM\TPM\max_eff	-	maximum empirical efficiency of the propeller
	D_P	m	ACM\PFM\TPM\prop_dia	-	propeller diameter
	-	-	ACM\PFM\TPM\throttle\low	-	turboprop low throttle parameter
	-	-	ACM\PFM\TPM\throttle\high	-	turboprop high throttle parameter
PEM	$\dot{W}_{P_{1,max,std,MSL}}$	hp	ACM\PFM\PEM\P	-	maximum one-engine power in standard atmosphere at MSL
	$H_{\rho,turbo}$	ft	ACM\PFM\PEM\Hd_turbo	-	maximum density altitude at which the turbocharger can maintain the sea level admission pressure
	η_{max}	-	ACM\PFM\PEM\max_eff	-	maximum empirical efficiency of the propeller
	D_P	m	ACM\PFM\PEM\prop_dia	-	propeller diameter
	CPSFC	-	ACM\PFM\PEM\CPSFC	-	power-specific fuel consumption
GLM	h_{MO}	ft	ACM\ALM\GLM\hmo	-	max. operating altitude
	mfa	ft	ACM\ALM\GLM\mfa	-	maximum altitude with high-lift devices extended
KLM	M_{MO}	-	ACM\ALM\KLM\mmo	-	max. operating Mach number
	M_{LE}	-	ACM\ALM\KLM\mle	-	max. operating Mach number with landing gear extended
	-	-	ACM\ALM\KLM\mlo	-	max. operating Mach number with landing gear extending or retracting
	V_{MO}	kt	ACM\ALM\KLM\vmo	-	max. operating speed
	$V_{FE,\,\delta HL}$	kt	ACM\ACFM\Configurations\vfe	-	max. operating speed with high-lift devices extended
	V_{LE}	kt	ACM\ALM\KLM\vle	-	max. operating speed with landing gear extended
	-	kt	ACM\ALM\KLM\vloe	-	max. operating speed with landing gear extending
	-	kt	ACM\ALM\KLM\vlor	-	max. operating speed with landing gear retracting
BLM	$bf_1...bf_5$	-	ACM\AFCM\Configurations\LGUP\BLM_clean\CL_clean	-	clean configuration buffet coefficients
	$C_{L,Mach0}$	-	ACM\AFCM\Configurations\LGUP\BLM_clean\CL_Mach0	-	maximum lift coefficient at M=0
	M_{min}	-	ACM\AFCM\Configurations\LGUP\BLM_clean\Mmin	-	lower boundary of the domain of applicability of the clean configuration buffet model
	M_{max}	-	ACM\AFCM\Configurations\LGUP\BLM_clean\Mmax	-	upper boundary of the domain of applicability of the clean configuration buffet model

Model or Category	Symbol	Unit	XML location		Description
			Element	Attribute	
	$C_{L,max,\delta HL,\delta LG}$	-	ACM\AFCM\Configurations\LG[UP|DN]\BLM\CL_max	-	maximum lift coefficient
DLM	MTW	kg	ACM\ALM\DLM\MTW	-	max. taxi weight
	MTOW	kg	ACM\ALM\DLM\MTOW	-	max. take-off weight
	MLW	kg	ACM\ALM\DLM\MLW	-	max. landing weight
	MZFW	kg	ACM\ALM\DLM\MZFW	-	max. zero-fuel weight
	OEW	kg	ACM\ALM\DLM\OEW	-	operating empty weight
	MPL	kg	ACM\ALM\DLM\MPL	-	max. payload
	MFL	kg	ACM\ALM\DLM\MFL	-	max. fuel load
	n_1	-	ACM\ALM\DLM\n1	-	max. load factor with high-lift devices retracted
	n_3	-	ACM\ALM\DLM\n3	-	min. load factor with high-lift devices retracted
	n_{f1}	-	ACM\ALM\DLM\nf1	-	max. load factor with high-lift devices extended
	n_{f3}	-	ACM\ALM\DLM\nf3	-	min. load factor with high-lift devices extended
ELM	$T_{max,f}$	K	ACM\ALM\ELM\Tmax_f	-	max. temperature deviation in flight
	$T_{max,g}$	K	ACM\ALM\ELM\Tmax_g	-	max. temperature deviation during take-off and landing
	$T_{min,f}$	K	ACM\ALM\ELM\Tmin_f	-	min. temperature deviation in flight
	$T_{min,g}$	K	ACM\ALM\ELM\Tmin_g	-	min. temperature deviation during take-off and landing
Ground movement	-	m	ACM\Ground\Dimensions\span	-	aircraft span
	-	m	ACM\Ground\Dimensions\length	-	aircraft length
	-	m	ACM\Ground\Runway\TOL	-	take-off length
	-	m	ACM\Ground\Runway\LDL	-	landing length
ARPM		-	ACM\ARPM\AeroConfSchedule\AeroPhase\name	-	aerodynamic phase (TO, IC, CR, AP or LD)
		-	ACM\ARPM\AeroConfSchedule\AeroPhase\HLid	-	aerodynamic configuration identifier
		-	ACM\ARPM\AeroConfSchedule\AeroPhase\LG	-	landing gear position (UP or DN)
		-	ACM\ARPM\SpeedScheduleList\SpeedSchedule\CompanyName	-	airline company name
		-	ACM\ARPM\SpeedScheduleList\SpeedSchedule\SpeedPhase\name	-	flight phase (Climb, Cruise or Descent)
	V_1	kt	ACM\ARPM\SpeedScheduleList\SpeedSchedule\SpeedPhase\CAS1	-	CAS to be flown below FL100
	V_2	kt	ACM\ARPM\SpeedScheduleList\SpeedSchedule\SpeedPhase\CAS2	-	CAS to be flown between FL100 and crossover altitude
	M	-	ACM\ARPM\SpeedScheduleList\SpeedSchedule\SpeedPhase\Mach	-	Mach number to be flown above crossover altitude

8. GLOBAL AIRCRAFT PARAMETERS

8.1. INTRODUCTION

A number of parameters mentioned in Appendix B (Airline Procedure Model) have values that are independent of the aircraft type or model for which they are used. The values of these and other parameters which have general use, have been put in the Global Parameters File (see section 9.9). This increases the flexibility and allows an easier evaluation of the values that are used.

The next section gives an overview of the parameters that are defined in the Global Parameters File. If relevant, it also indicates the formula in which the parameter should be used.

8.2. CONFIGURATION ALTITUDE THRESHOLD

For 4 flight segments (see Appendix B), altitude thresholds have been specified in BADA: take-off (TO), initial climb (IC), approach (AP) and landing (LD). Note that the selection of the take-off and initial climb configurations is defined only with the altitude. The selection of the approach and landing configurations is done through the use of airspeed and altitude (see Appendix B), while the altitudes at which the configuration change takes place should not be higher than the ones given below. The altitude values are expressed in terms of geopotential pressure altitude.

Name:	Description:	Value [ft]:
$H_{max,TO}$	Maximum altitude threshold for take-off	400
$H_{max,IC}$	Maximum altitude threshold for initial climb	2,000
$H_{max,AP}$	Maximum altitude threshold for approach	8,000
$H_{max,LD}$	Maximum altitude threshold for landing	3,000

8.3. MINIMUM SPEED COEFFICIENTS

Two minimum speed coefficients are specified, which are to be used in the expressions of Appendix B:

Name:	Description:	Value [-]:
$C_{Vmin,TO}$	Minimum speed coefficient for take-off	1.13
C_{Vmin}	Minimum speed coefficient (all other phases)	1.23

8.4. SPEED SCHEDULES

The speed schedules applicable below FL100 for climb and descent are based on a factored stall speed plus increment valid for a specified geopotential pressure altitude range.

Name:	Description:	Value [KCAS]:
$Vd_{CL,1}$	Climb speed increment below 1500 ft (jet)	5
$Vd_{CL,2}$	Climb speed increment below 3000 ft (jet)	10
$Vd_{CL,3}$	Climb speed increment below 4000 ft (jet)	30

$Vd_{CL,4}$	Climb speed increment below 5000 ft (jet)	60
$Vd_{CL,5}$	Climb speed increment below 6000 ft (jet)	80
$Vd_{CL,6}$	Climb speed increment below 500 ft (turbo/piston)	20
$Vd_{CL,7}$	Climb speed increment below 1000 ft (turbo/piston)	30
$Vd_{CL,8}$	Climb speed increment below 1500 ft (turbo/piston)	35
$Vd_{DES,1}$	Descent speed increment below 1000 ft (jet/turboprop)	5
$Vd_{DES,2}$	Descent speed increment below 1500 ft (jet/turboprop)	10
$Vd_{DES,3}$	Descent speed increment below 2000 ft (jet/turboprop)	20
$Vd_{DES,4}$	Descent speed increment below 3000 ft (jet/turboprop)	50
$Vd_{DES,5}$	Descent speed increment below 500 ft (piston)	5
$Vd_{DES,6}$	Descent speed increment below 1000 ft (piston)	10
$Vd_{DES,7}$	Descent speed increment below 1500 ft (piston)	20

These values are to be used in the expressions of Appendix B.

9. BADA RELEASE FILES

9.1. FILE TYPES

All data provided by BADA Family 4 are organised into six types of files:

- a set of Aircraft Model files,
- a set of Accuracy Tables Files,
- a set of Performance Table Files,
- a set of Performance Table Data,
- a set of Optimized Performance Tables,
- a Global Parameter File.

There is one Aircraft Model file (ACM) provided for each aircraft type which is directly supported. This file specifies parameter values for the flight envelope, drag, engine thrust and fuel consumption that are described in Section 2, as well as nominal manoeuvre speeds and aerodynamic configurations that are described in Appendix B. Details on the format of the ACM file are given in Section 9.4.

There is one Accuracy Tables File (ATF) for each directly supported aircraft type. This file contains summary tables of the accuracy of the BADA model compared to the manufacturer reference data, in terms of RMS error in vertical speed and fuel consumption, at various speeds, altitudes and temperature deviations. Details on the format of the ATF file are given in Section 9.5.

There are two Performance Table Files (PTF) for each directly supported aircraft type. Each file contains a summary table of speeds, climb/descent rates and fuel consumption at various flight levels and one temperature deviation. Details on the format of the PTF file are given in Section 9.6.

There are two Performance Table Data (PTD) files for each directly supported aircraft type. Each file contains a detailed table of computed performance values at various flight levels and one temperature deviation. Details on the format of the PTD file are given in Section 9.7.

There are five (resp. four) Optimized Performance Tables (OPT) files for each directly supported aircraft type with turbofan (resp. turboprop) engines. These files contain tables of pre-computed optimum flight parameters (either speed schedule or cruise flight level) at various flight conditions. Details on the format of the OPT files are given in Section 9.8.

Finally there is one Global Parameter File (GPF). This file contains parameters that are described in Section 8 and are valid for all aircraft or a group of aircraft (for instance all jet aircraft). Details on the format of the GPF file are given in Section 9.9.

9.2. BADA RELEASE

9.2.1. Release

The release is a label that indicates the version of the project, in this case the release of BADA files. BADA releases are usually identified by a two digit number, e.g. 4.0. No information on the current BADA release is given inside the BADA files.

9.2.2. Release Summary file

The release summary file, named *release.csv*, provides a list of all the aircraft models delivered as part of the BADA release. It lists, for each BADA model, the file name and several key aircraft parameters, as well as the BADA release identification, which is the BADA release in which the model was last modified.

9.3. ACCESS TO BADA RELEASES

The release files and documents associated with BADA Family 4 are accessible through the BADA Support Application (BSA). The BSA is a Web application that provides BADA users with the ability to exchange requests, as well as data files and documents, with the BADA team members. It is also used as data repository for the BADA release files and documents.

The right to use the BSA is granted to the licensed user of BADA. The application can be accessed by using a dedicated login and password as provided by the BADA team.

Once granted the access right to BSA, the user can access the application at this address:

https://remedyweb.eurocontrol.fr

by using the BADA Support Application link and logging in with the provided login/password.

Once logged in, the user can access BADA releases through the Librairies→Releases item located in the main menu:

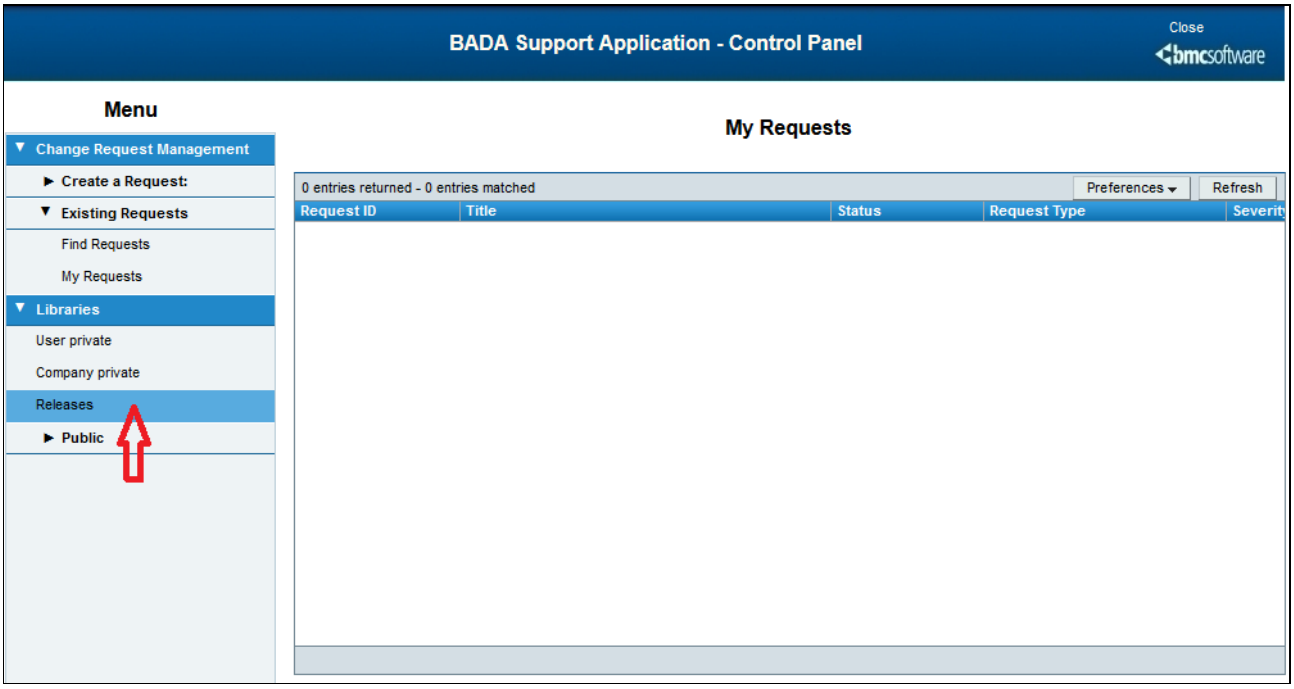

Figure 3 – BADA Support Application – Main menu

Clicking this item opens up the release library page, from where the user can download the BADA release files and documents:

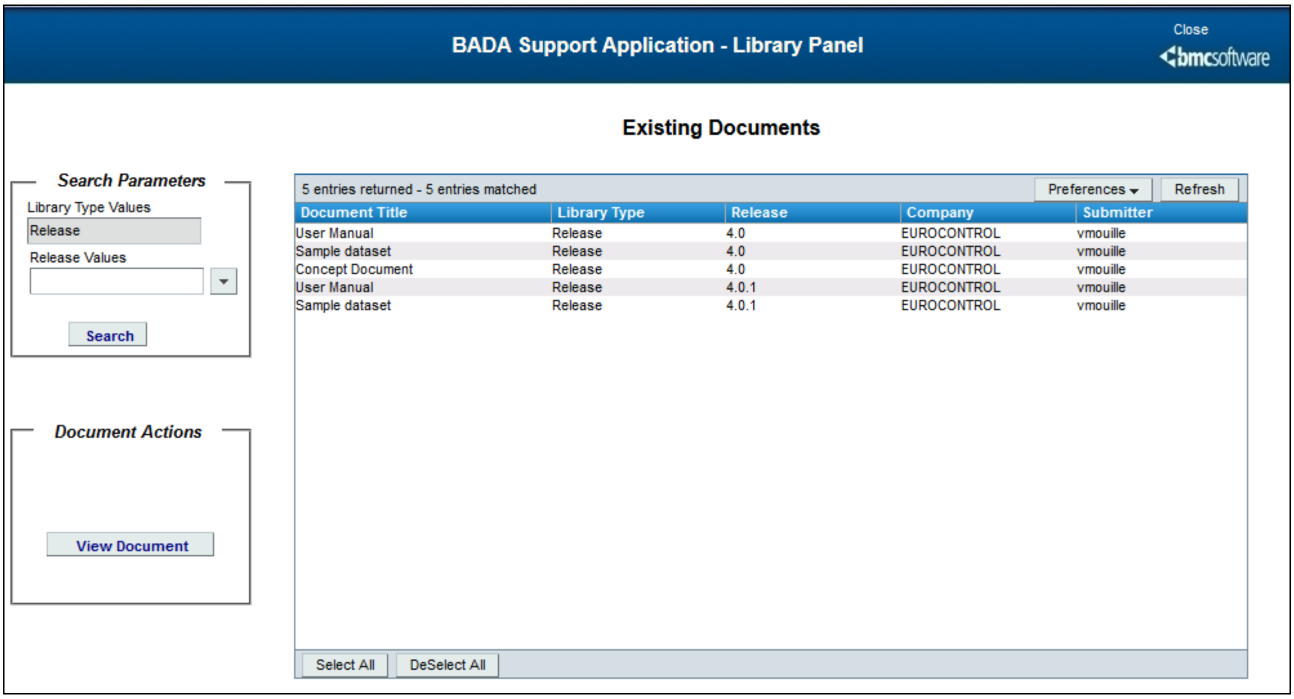

Figure 4 – BADA Support Application – Release Library

This process, as well as the general usage of the BSA application, is described in detail in [RD5].

Important:

- Only a limited set of sample BADA Family 4 aircraft data files are available from the BSA, for the specific purpose of software verification.
- The aircraft-specific datasets are provided separately. The Focal Point is informed about the delivery process once all required permissions are obtained from EUROCONTROL partner data providers.

9.4. ACM FILE FORMAT

The Aircraft Model file (ACM) is an XML file which, for a particular aircraft type, specifies the performance parameters summarized in Section 7.2. The format of this file is described in an XML schema presented in section 9.4.1, and an example of ACM file is shown in section 9.4.2.

9.4.1. Format of the ACM file

The format of the ACM file is described in an XML schema, which can be used to guide an XML parser in the reading of the ACM XML file. This XML schema is provided in a file named *ACM_BADA4.xsd*, and its content is presented below. For the sake of clarity, it has been split into its different submodels.

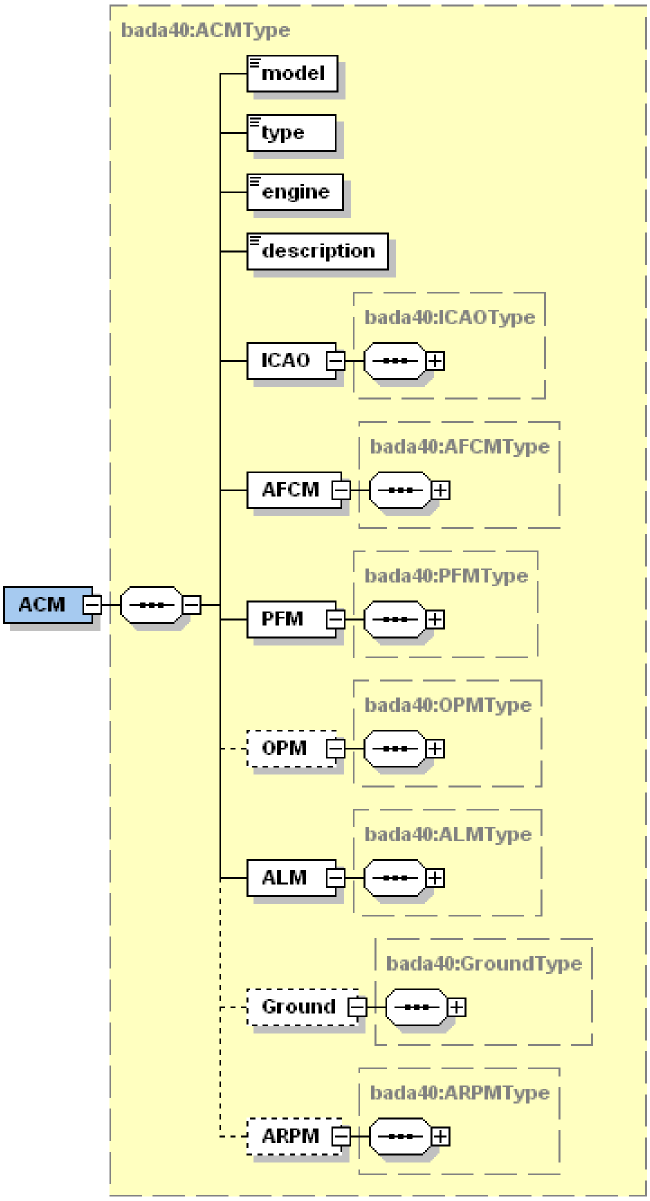

Figure 5 – XML schema of ACM file – ACM element type

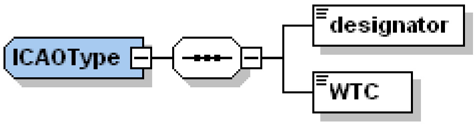

Figure 6 – XML schema of ACM file – ICAO element type

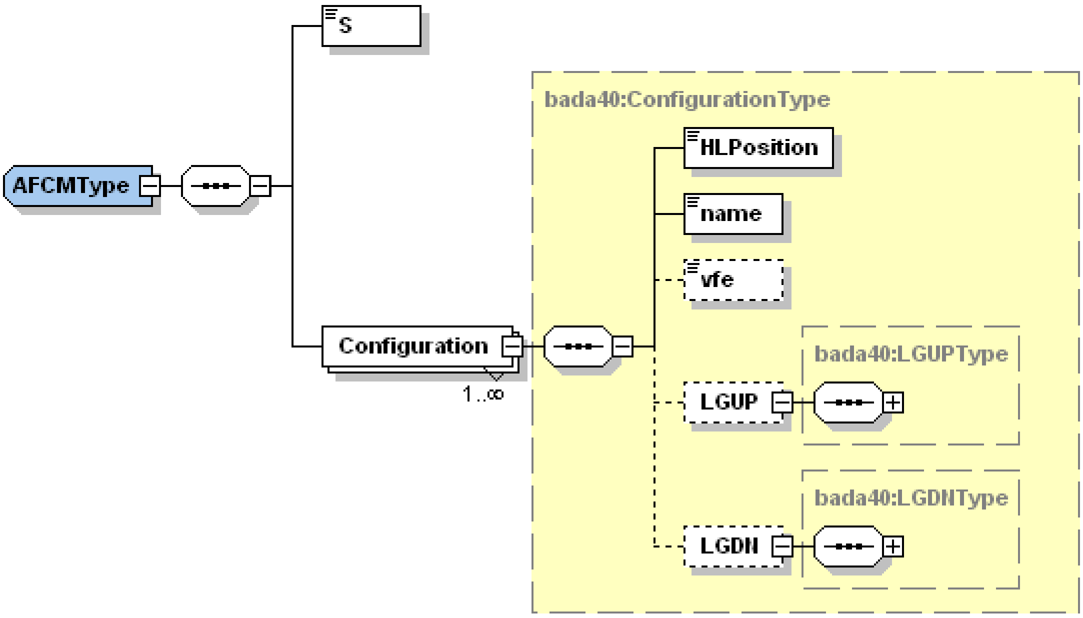

Figure 7 – XML schema of ACM file – AFCM element type

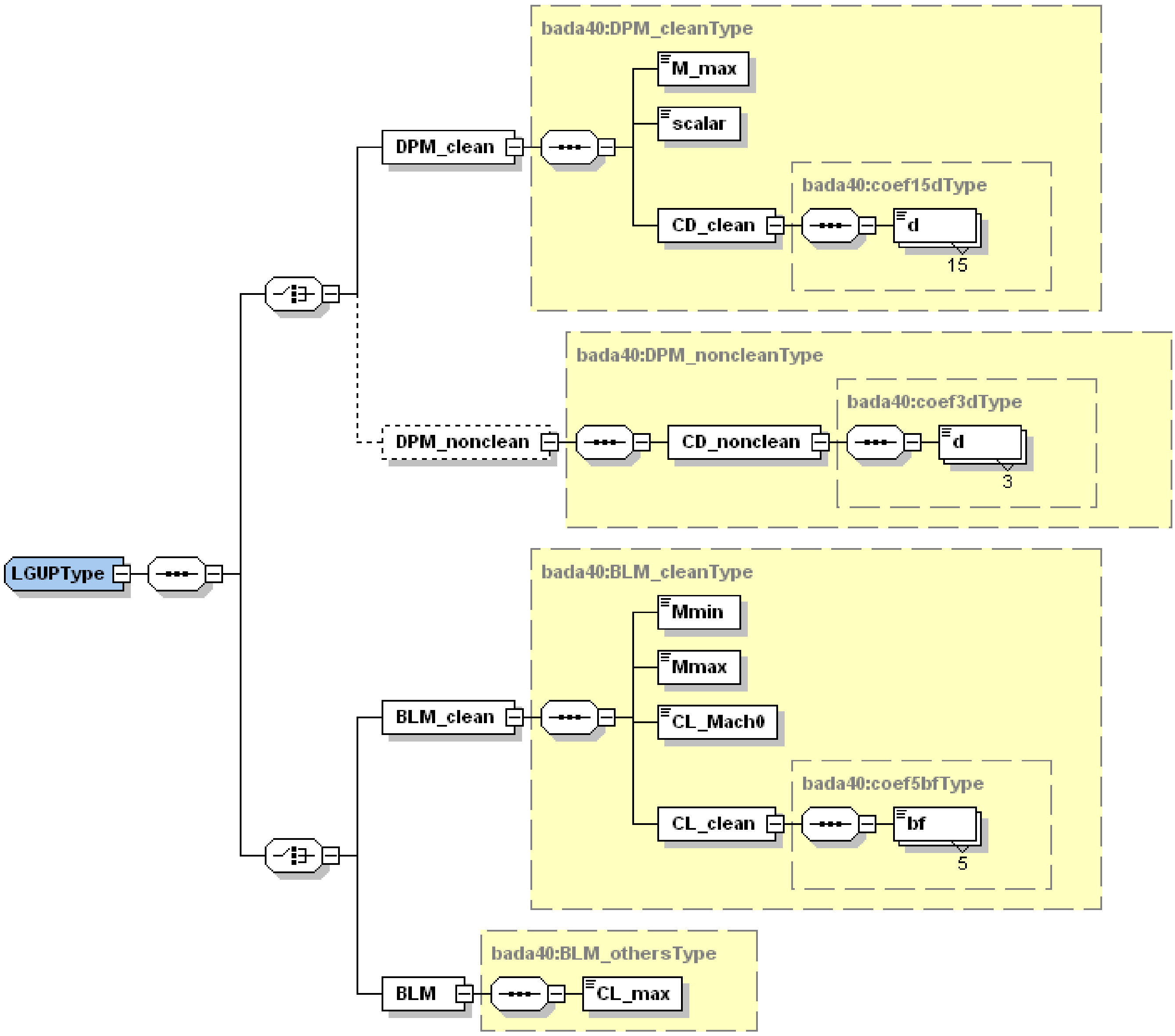

Figure 8 – XML schema of ACM file – LGUP element type

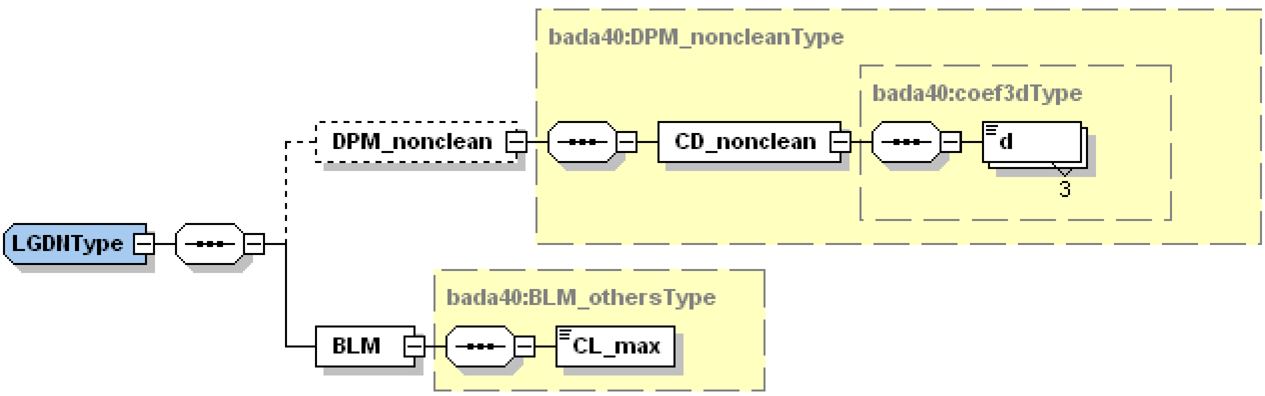

Figure 9 – XML schema of ACM file – LGDN element type

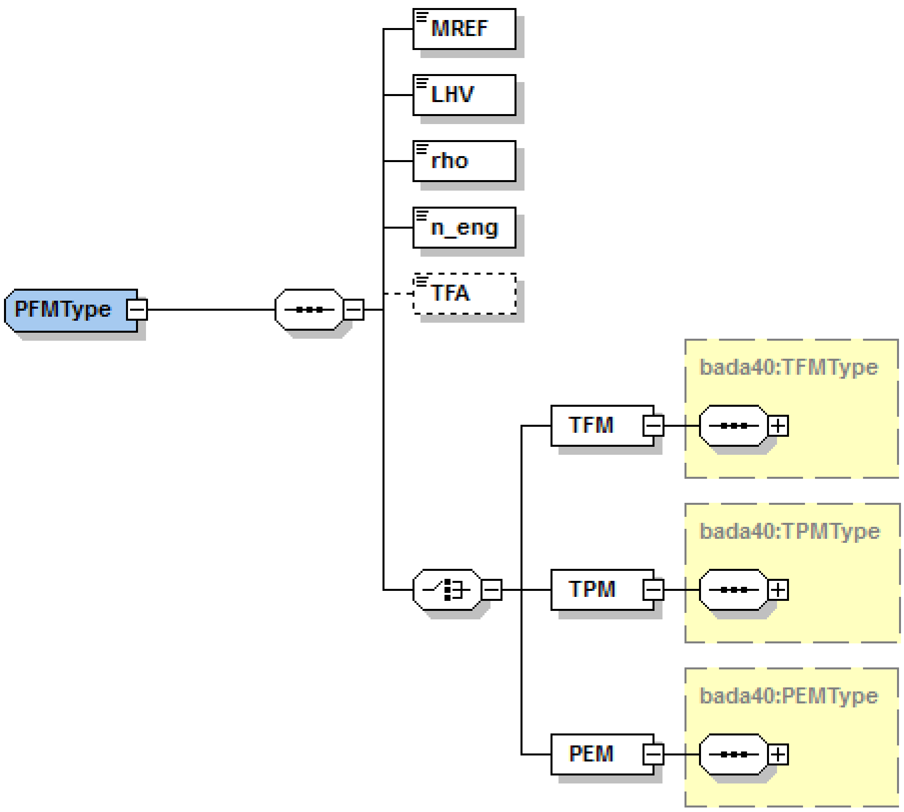

Figure 10 – XML schema of ACM file – PFM element type

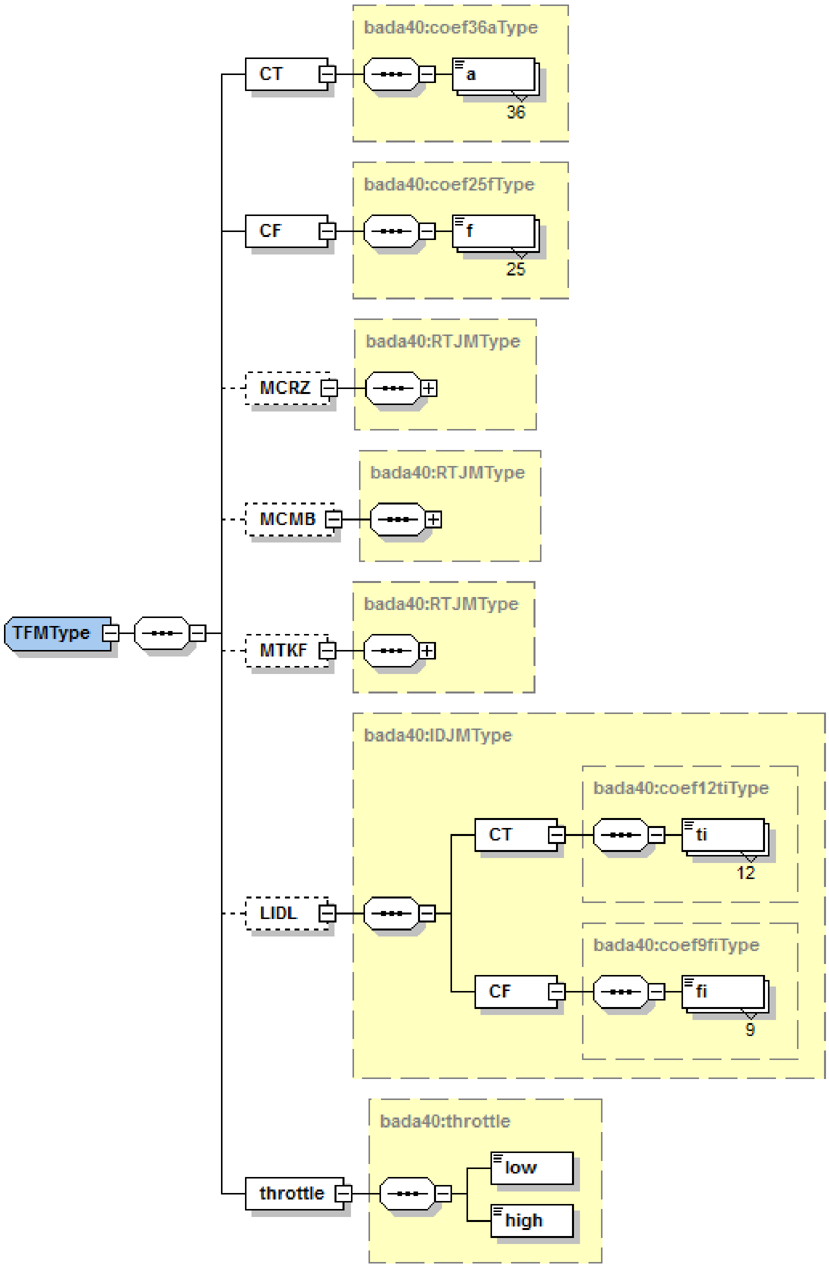

Figure 11 – XML schema of ACM file – TFM element type

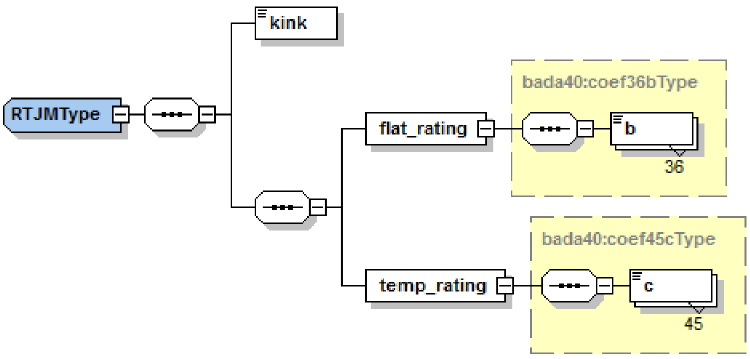

Figure 12 – XML schema of ACM file – RTJM element type

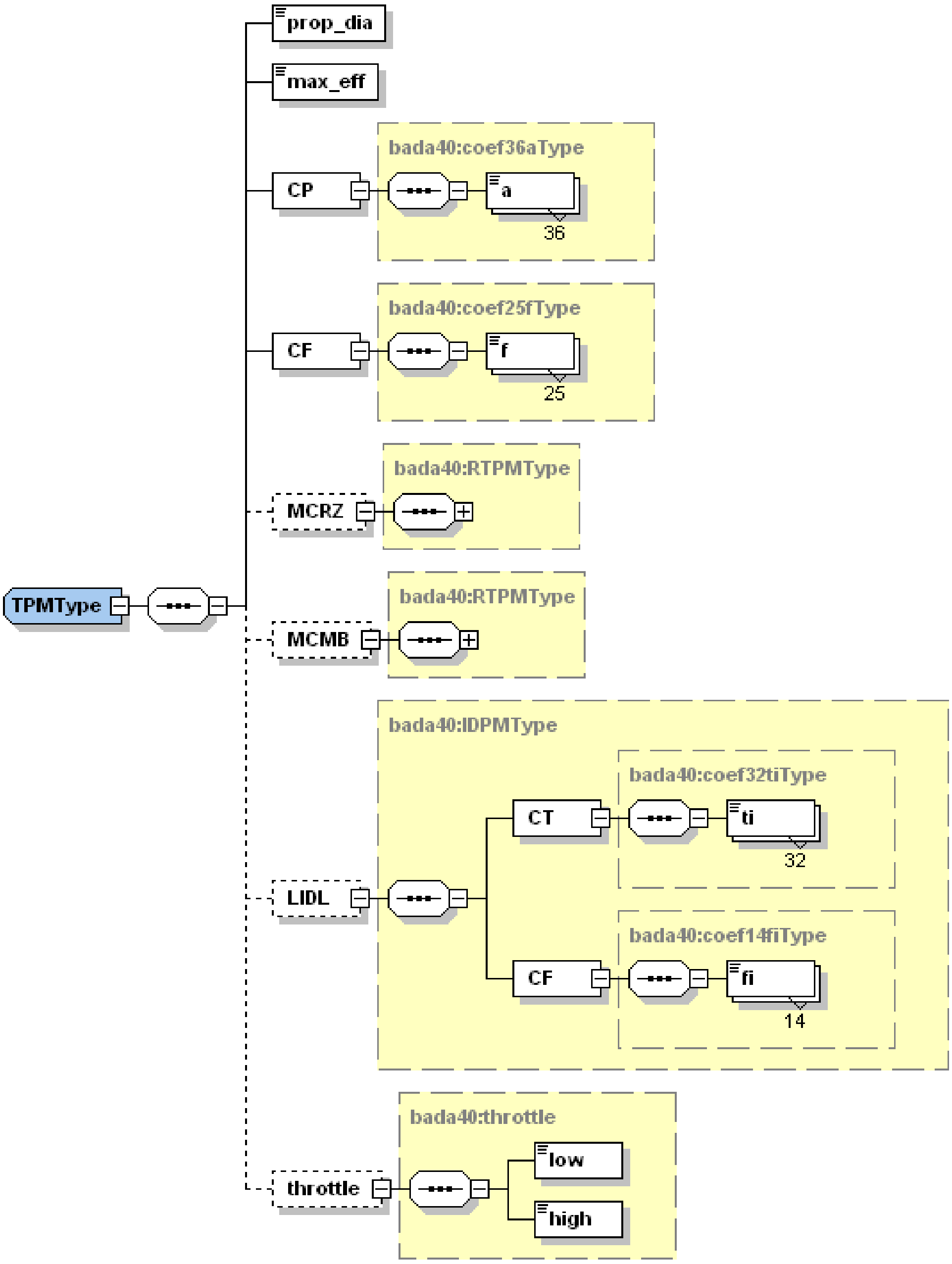

Figure 13 – XML schema of ACM file – TPM element type

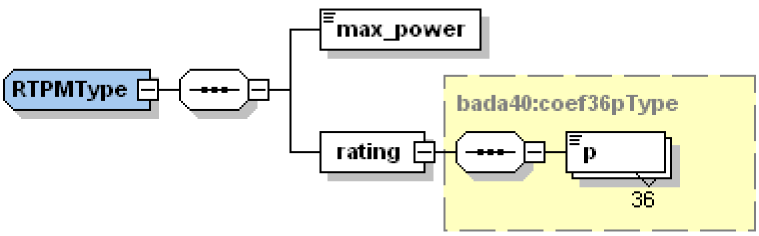

Figure 14 – XML schema of ACM file – RTPM element type

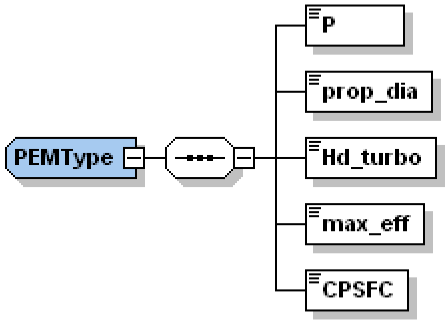

Figure 15 – XML schema of ACM file – PEM element type

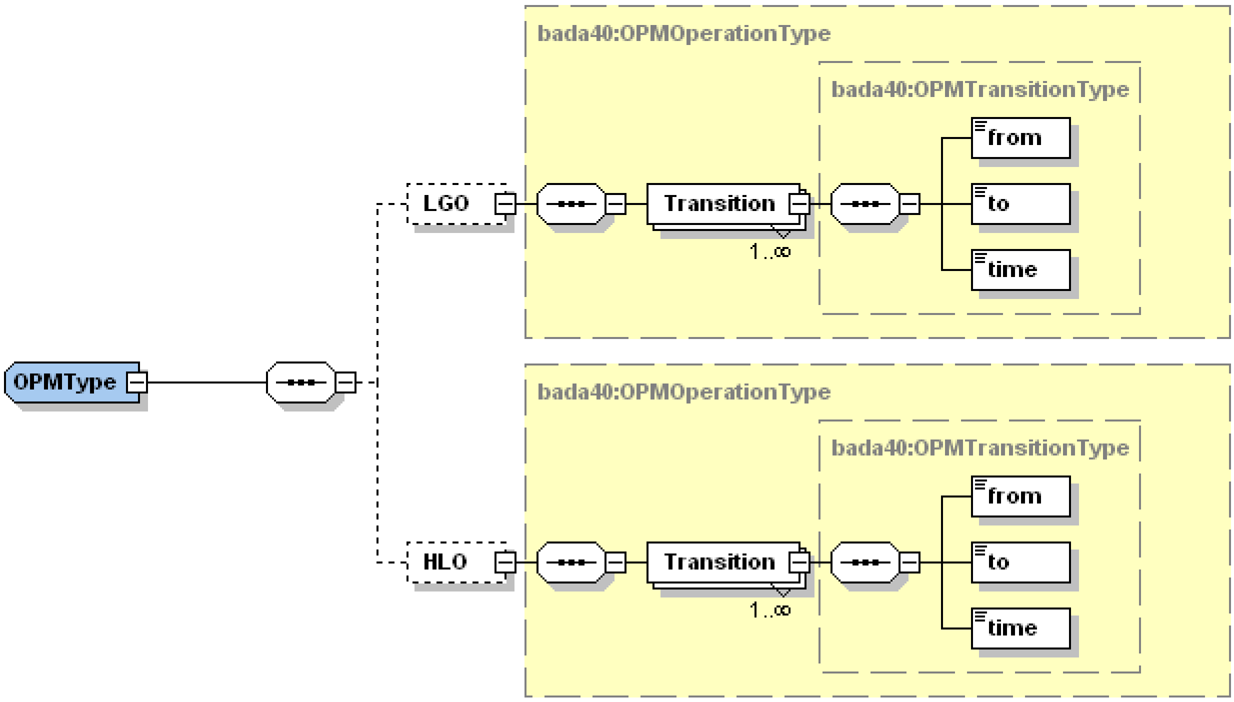

Figure 16 – XML schema of ACM file – OPM element type

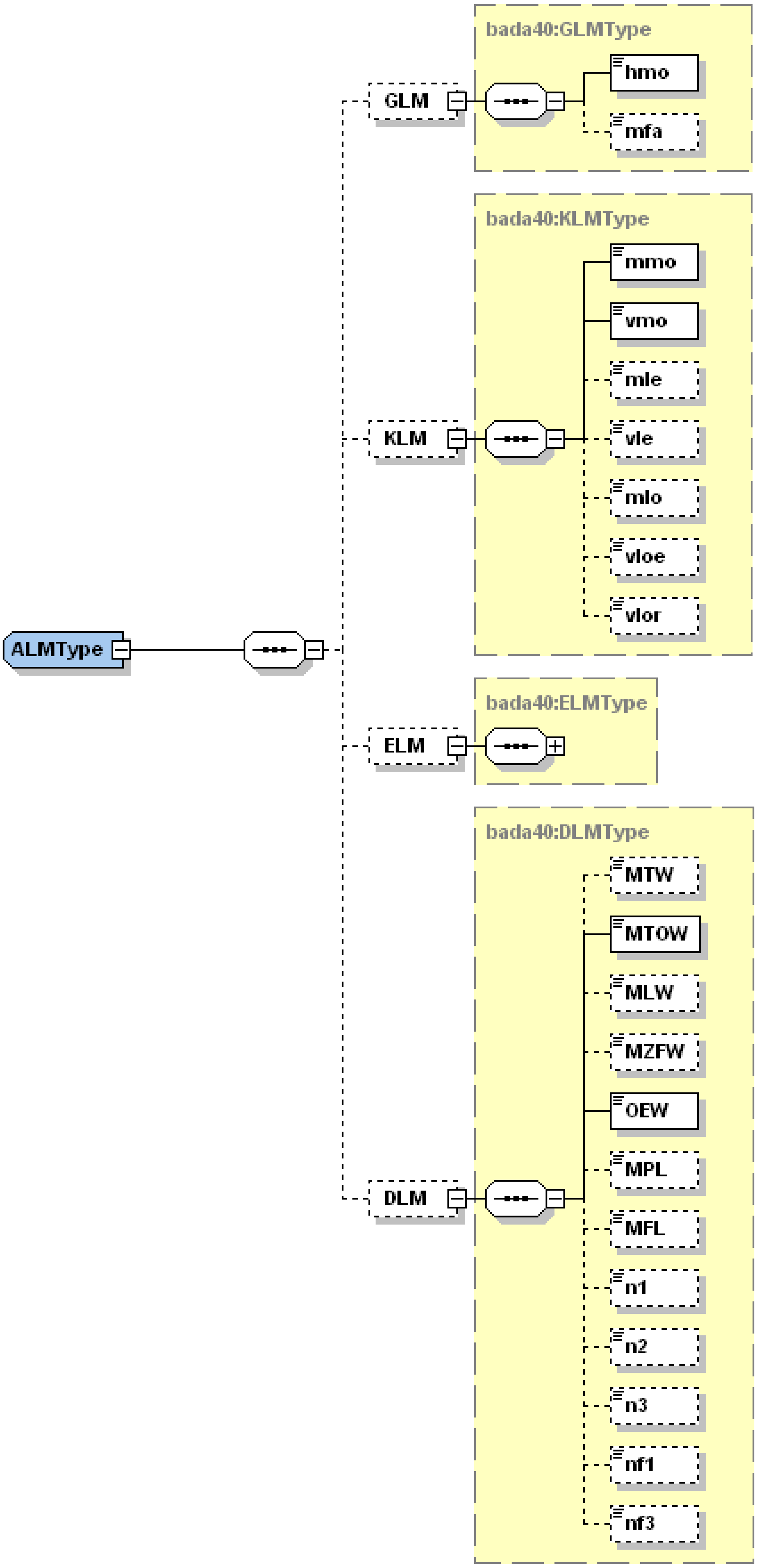

Figure 17 – XML schema of ACM file – ALM element type

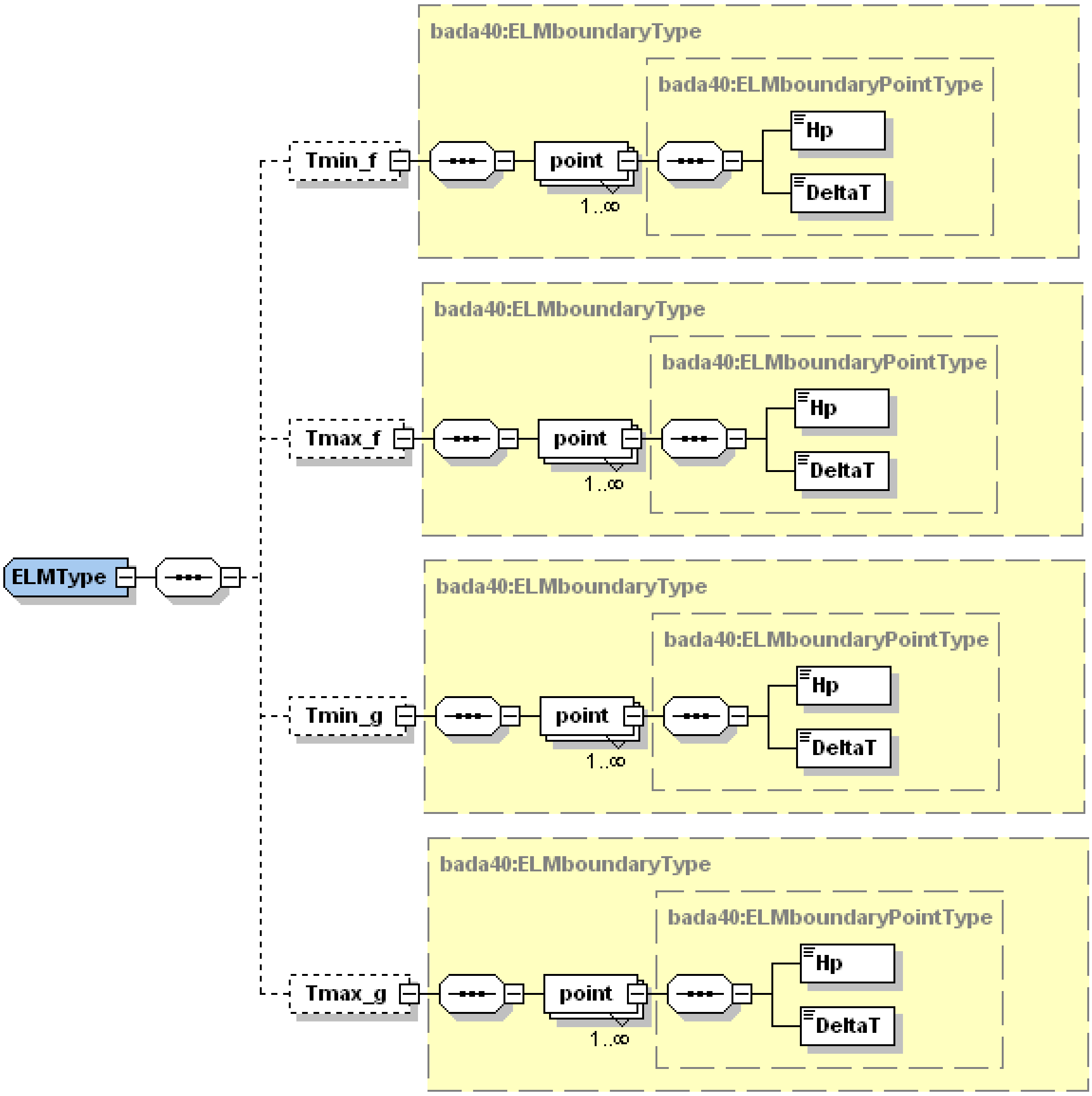

Figure 18 – XML schema of ACM file – ELM element type

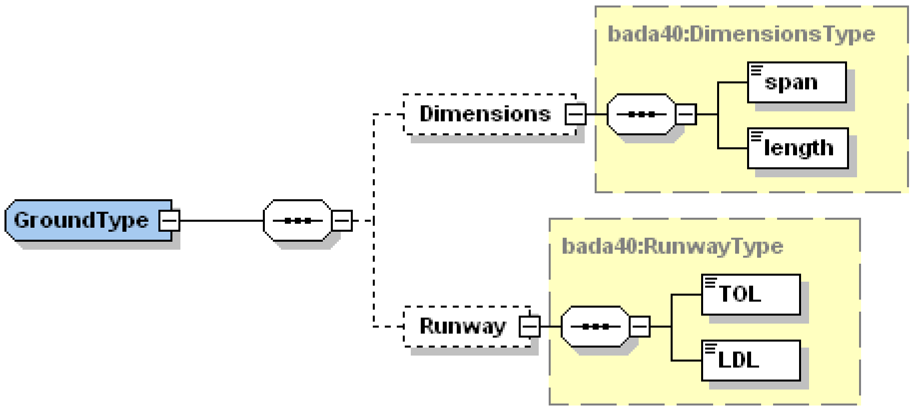

Figure 19 – XML schema of ACM file – Ground element type

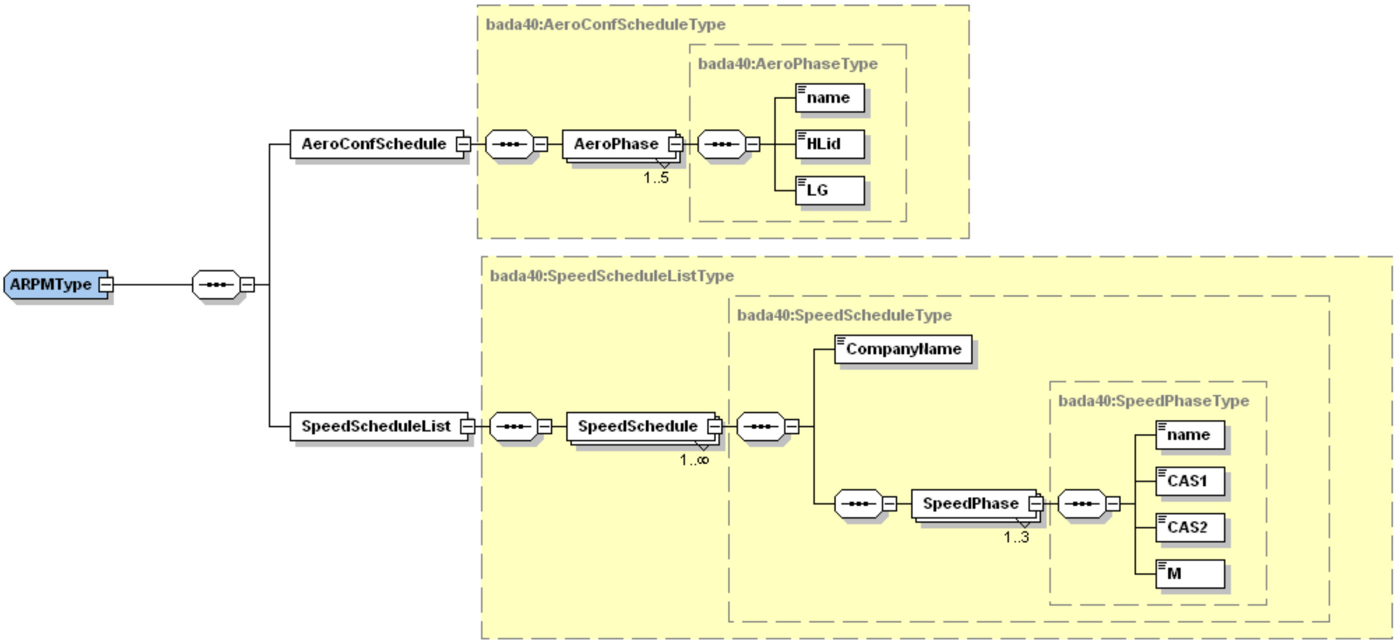

Figure 20 – XML schema of ACM file – ARPM element type

9.4.2. Example of ACM file

An example of ACM file for a dummy jet aircraft is presented below. For the sake of clarity, it has been split into its different submodels, and some items have been collapsed.

model	Dummy-TWIN		
type	JET		
engine	CFM56-XXX		
description			
ICAO			
	designator	DUM2	
	WTC	M	
Ground			
	Dimensions		
		span	33.4
		length	32.5
	Runway		
		TOL	1500
		LDL	1400

Figure 21 – Example of ACM file – Additional parameters sections

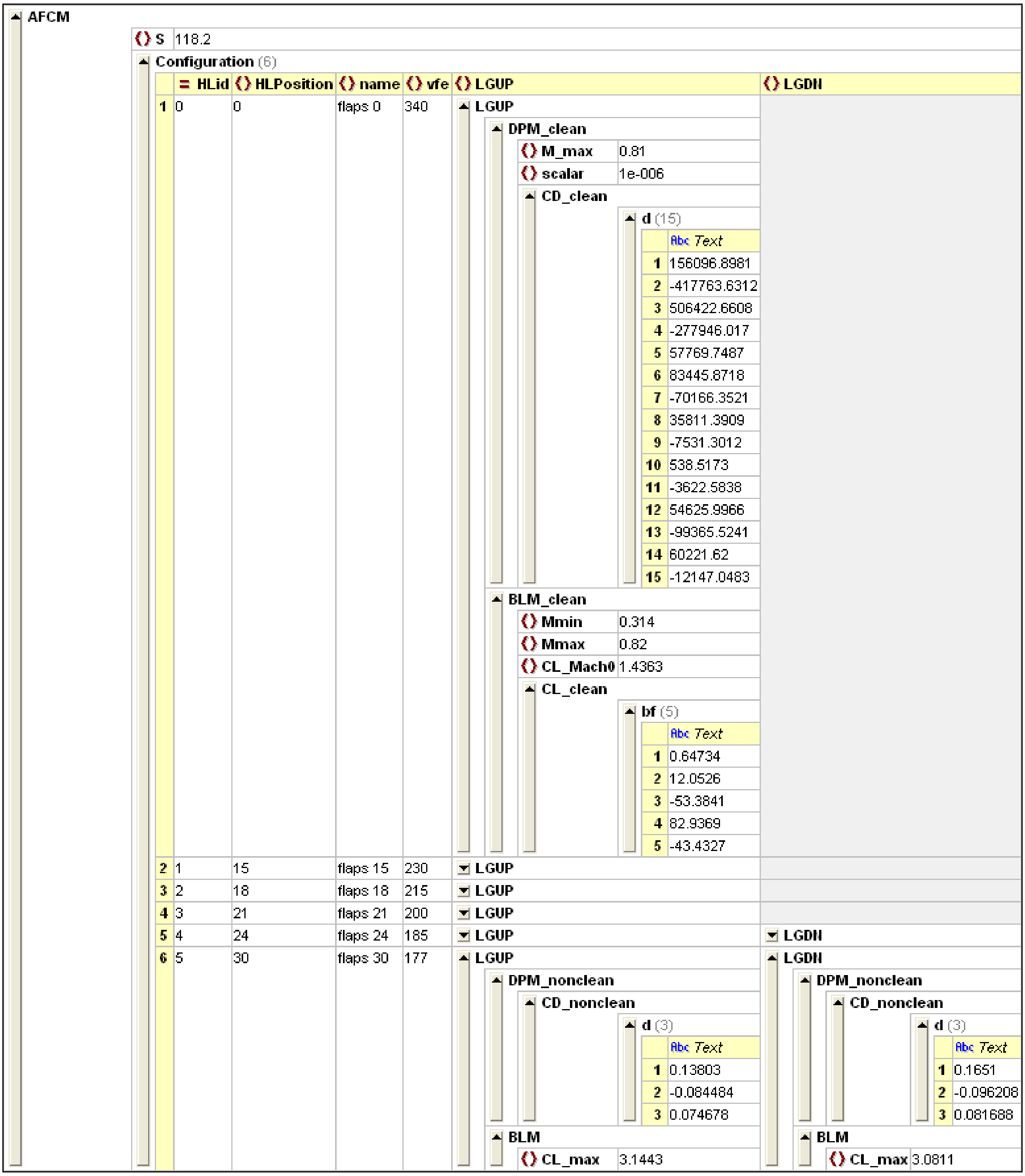

Figure 22 – Example of ACM file – AFCM section

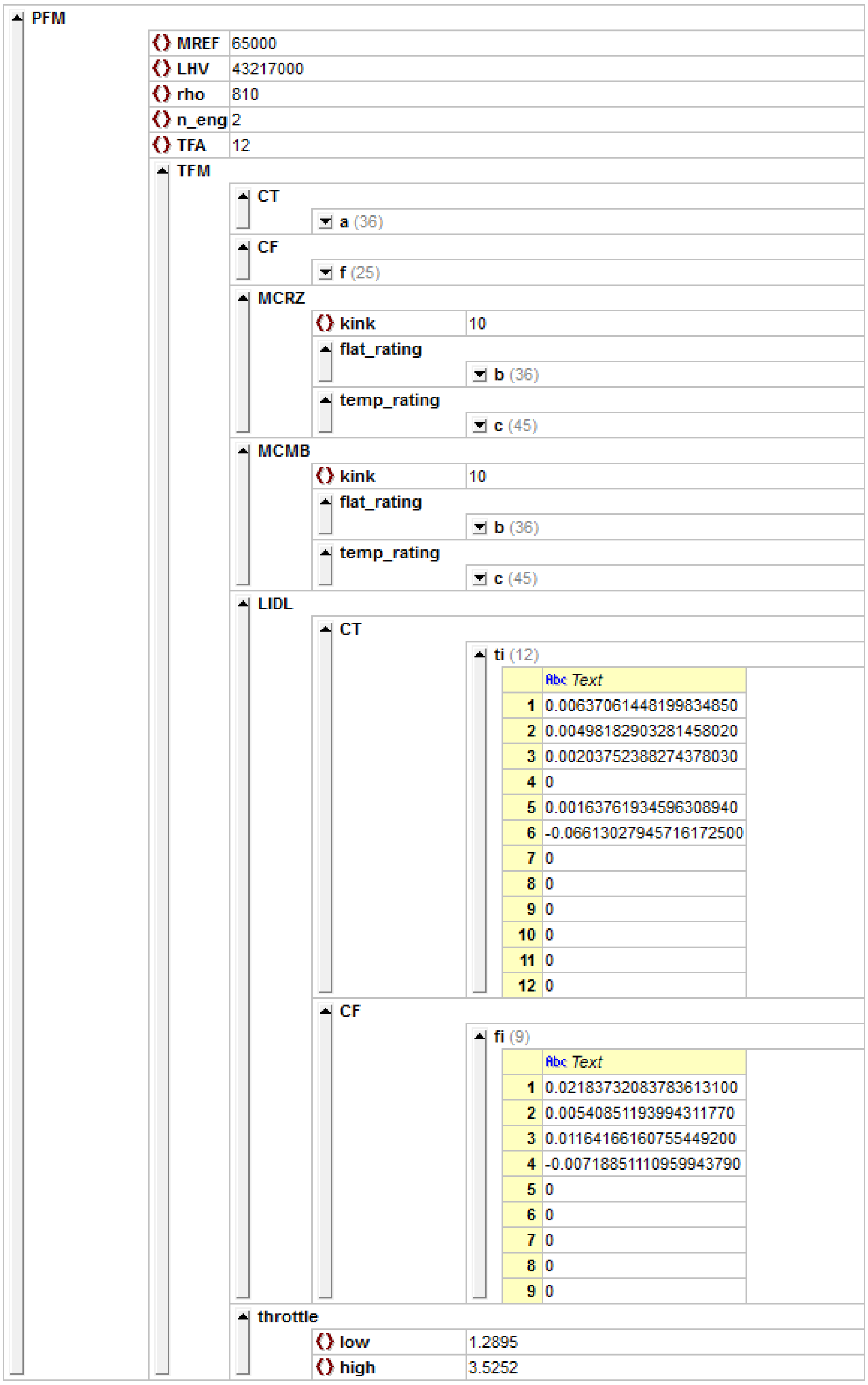

Figure 23 – Example of ACM file – PFM section

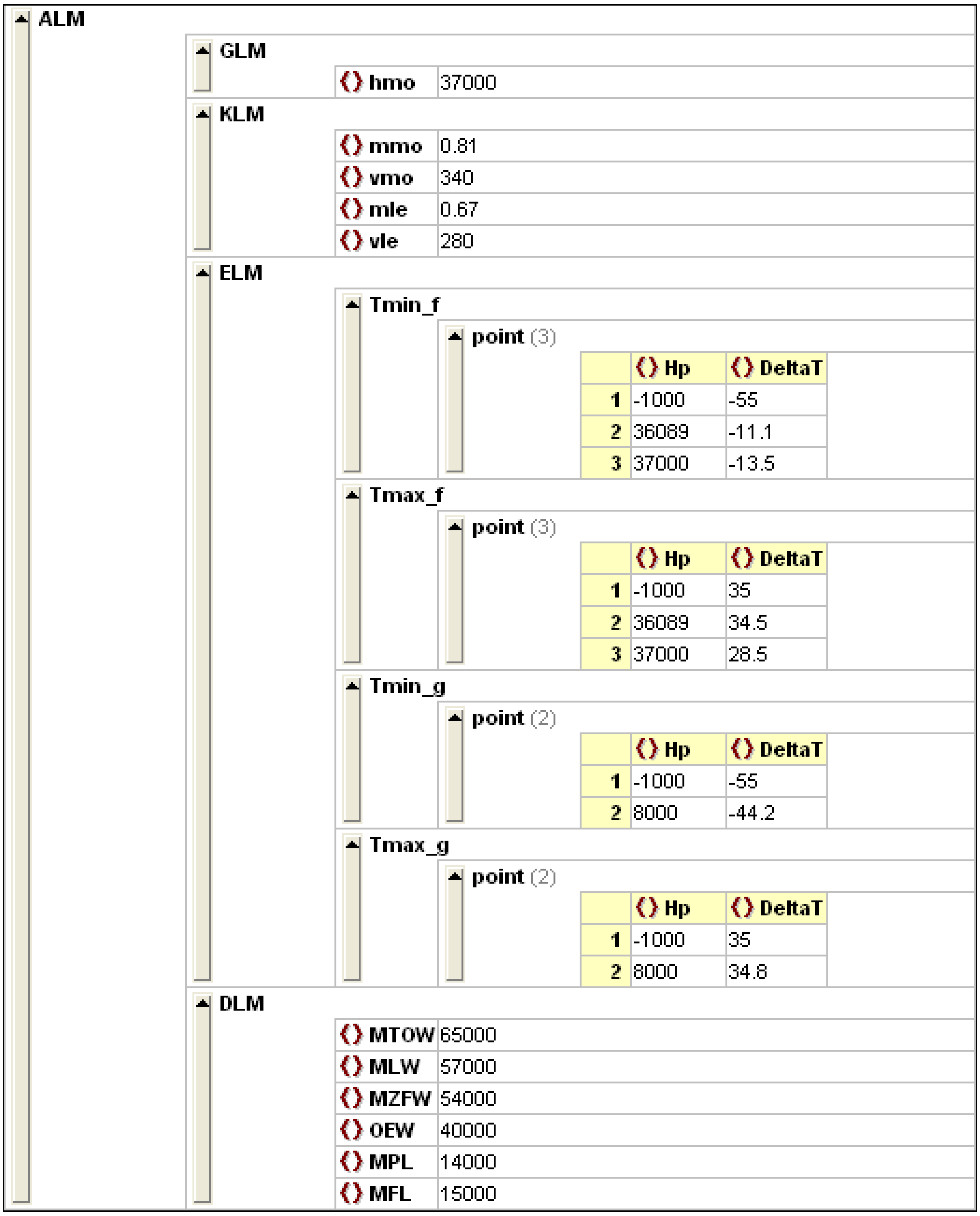

Figure 24 – Example of ACM file – ALM section

OPM

LGO

Transition (2)

	from	to	time
1	0	1	20
2	1	0	20

HLO

Transition (10)

	from	to	time
1	0	1	12
2	1	2	6.5
3	2	3	3.9
4	3	4	2.5
5	4	5	5.9
6	5	4	5.4
7	4	3	2.4
8	3	2	3.9
9	2	1	6.6
10	1	0	12.1

Figure 25 – Example of ACM file – OPM section

At the moment, within BADA all ARPM specify procedures for only one "default" company. The ACM file format, however, is designed so that the company list can be extended for the different companies which operate the aircraft and which may have different standard procedures.

ARPM

AeroConfSchedule

AeroPhase (5)

	name	HLid	LG
1	TO	2	UP
2	IC	1	UP
3	CR	0	UP
4	AP	3	UP
5	LD	5	DN

SpeedScheduleList

SpeedSchedule

CompanyName Default

SpeedPhase (3)

	name	CAS1	CAS2	M
1	Climb	310	310	0.78
2	Cruise	250	310	0.78
3	Descent	300	300	0.79

Figure 26 – Example of ACM file – ARPM section

9.5. ATF FILE FORMAT

The Accuracy Tables File (ATF) is an ASCII file which, for a particular aircraft type, presents summary tables of the accuracy of the BADA model compared to the manufacturer reference data, in terms of RMS error in vertical speed and fuel consumption, at various temperature deviations, altitudes, masses and speeds. An example of ATF file for a dummy jet aircraft is shown in Figure 27.

```
ERROR RESULTS SUMMARY for Dummy-TWIN:
====================================
                         Vertical speed  Fuel consumption     Points
                         [ fpm ]  [  % ]  [kg/min] [  % ]   [dimless]
=====================================================================
MCRZ FLAT CLIMBS     :    36.85    3.54      0.93    1.14       12516
MCRZ TEMP CLIMBS     :    58.42    5.94      1.44    1.95        8060
ALL MCRZ CLIMBS      :    46.51    4.63      1.16    1.51       20576
=====================================================================
MCMB FLAT CLIMBS     :    35.76    2.07      1.33    1.46       12494
MCMB TEMP CLIMBS     :    58.31    4.70      1.22    1.52        8362
ALL MCMB CLIMBS      :    46.14    3.38      1.28    1.49       20856
=====================================================================
ALL CLIMBS           :    46.32    4.05      1.22    1.50       41432
=====================================================================
ALL DESCENTS         :    31.05    1.46      0.44    5.43       16485
=====================================================================
ALL CRUISES          :     N/A     N/A       1.24    2.86        4720
=====================================================================
ALL CLIMBS & DESCENTS:    42.54    3.51      1.06    3.16       57917
=====================================================================
ALL POINTS           :     N/A     N/A       1.08    3.14       62637
=====================================================================
NUMBER OF COEFFICIENTS USED IN MODELS:
DRAG POLAR           :           15
FUEL CONSUMPTION     :            7
GENERALIZED THRUST   :            6
MCRZ FLAT RATING     :            7
MCRZ TEMP RATING     :            9
MCMB FLAT RATING     :            7
MCMB TEMP RATING     :            9
LIDL THRUST          :            5
LIDL FUEL CONSUMPTION:            4
=====================================================================


ERROR RESULTS by TEMPERATURE DEVIATION for Dummy-TWIN:
=====================================================
                         Vertical speed  Fuel consumption     Points
                         [ fpm ]  [  % ]  [kg/min] [  % ]   [dimless]
=====================================================================
MCRZ CLIMB  ISA-30   :    31.37    1.51      1.36    1.50        1702
MCRZ CLIMB  ISA-20   :    30.60    2.37      1.06    1.24        2312
MCRZ CLIMB  ISA-10   :    37.77    4.03      0.81    1.01        2833
MCRZ CLIMB  ISA      :    39.17    4.24      0.72    0.96        2833
MCRZ CLIMB  ISA+10   :    51.19    4.41      0.78    1.08        5672
MCRZ CLIMB  ISA+20   :    41.93    4.61      1.50    1.99        2725
MCRZ CLIMB  ISA+30   :    71.07    8.00      1.89    2.56        2499
ALL MCRZ CLIMBS      :    46.51    4.63      1.16    1.51       20576
=====================================================================
MCMB CLIMB  ISA-30   :    32.20    1.27      1.90    1.90        1702
MCMB CLIMB  ISA-20   :    27.79    1.21      1.59    1.74        2296
MCMB CLIMB  ISA-10   :    37.95    2.47      1.23    1.39        2830
MCMB CLIMB  ISA      :    37.02    2.31      1.04    1.21        2834
MCMB CLIMB  ISA+10   :    51.45    2.77      0.98    1.19        5664
MCMB CLIMB  ISA+20   :    41.82    3.94      1.06    1.34        2839
MCMB CLIMB  ISA+30   :    69.38    6.46      1.55    1.94        2691
ALL MCMB CLIMBS      :    46.14    3.38      1.28    1.49       20856
=====================================================================
DESCENT     ISA-30   :    31.68    1.67      0.59    7.49        1577
DESCENT     ISA-20   :    27.29    1.44      0.47    6.24        2160
DESCENT     ISA-10   :    28.44    1.35      0.38    4.98        2567
DESCENT     ISA      :    26.44    1.27      0.35    4.19        2565
DESCENT     ISA+10   :    27.10    1.28      0.37    4.26        2561
DESCENT     ISA+20   :    30.29    1.36      0.44    5.01        2558
DESCENT     ISA+30   :    42.96    1.88      0.54    6.12        2497
ALL DESCENTS         :    31.05    1.46      0.44    5.43       16485
=====================================================================
CRUISE      ISA      :     N/A     N/A       1.24    2.86        4720
ALL CRUISES          :     N/A     N/A       1.24    2.86        4720
=====================================================================
```

```
ERROR RESULTS by GEOPOTENTIAL PRESSURE ALTITUDE [kft] for Dummy-TWIN:
======================================================================
                          Vertical speed  Fuel consumption     Points
                          [ fpm ]  [ % ]  [kg/min] [ % ]   [dimless]
======================================================================
MCRZ CLIMB [ 0,  5]  :     43.34    1.79     1.79    1.89        3520
MCRZ CLIMB ] 5, 10]  :     34.58    1.48     1.15    1.32        3488
MCRZ CLIMB ]10, 15]  :     44.39    2.50     0.84    1.03        3380
MCRZ CLIMB ]15, 20]  :     47.04    3.58     0.93    1.30        3242
MCRZ CLIMB ]20, 25]  :     46.76    4.95     1.06    1.62        2720
MCRZ CLIMB ]25, 30]  :     46.94    7.81     0.97    1.65        2211
MCRZ CLIMB ]30, 35]  :     63.70    8.18     0.84    1.63        1536
MCRZ CLIMB ]35, 40]  :     76.89   10.80     0.89    1.88         479
ALL MCRZ CLIMBS      :     46.51    4.63     1.16    1.51       20576
======================================================================
MCMB CLIMB [ 0,  5]  :     41.40    1.25     1.78    1.58        3520
MCMB CLIMB ] 5, 10]  :     38.13    1.21     1.16    1.13        3488
MCMB CLIMB ]10, 15]  :     45.28    1.83     1.09    1.15        3376
MCMB CLIMB ]15, 20]  :     47.82    2.47     1.16    1.40        3234
MCMB CLIMB ]20, 25]  :     44.46    3.82     1.38    1.87        2746
MCMB CLIMB ]25, 30]  :     46.76    5.57     1.31    1.99        2213
MCMB CLIMB ]30, 35]  :     57.54    5.40     0.74    1.28        1722
MCMB CLIMB ]35, 40]  :     73.57    8.21     0.64    1.21         557
ALL MCMB CLIMBS      :     46.14    3.38     1.28    1.49       20856
======================================================================
DESCENT    [ 0,  5]  :     24.11    1.49     0.71    7.09        2765
DESCENT    ] 5, 10]  :     24.83    1.42     0.44    4.98        2765
DESCENT    ]10, 15]  :     26.39    1.40     0.37    4.75        2765
DESCENT    ]15, 20]  :     27.28    1.30     0.41    5.72        2744
DESCENT    ]20, 25]  :     33.00    1.21     0.34    5.22        2308
DESCENT    ]25, 30]  :     32.79    1.35     0.30    4.95        1712
DESCENT    ]30, 35]  :     56.64    2.33     0.23    3.96        1202
DESCENT    ]35, 40]  :     42.54    1.67     0.19    3.29         224
ALL DESCENTS         :     31.05    1.46     0.44    5.43       16485
======================================================================
CRUISE     [ 0,  5]  :      N/A      N/A     1.63    3.99         705
CRUISE     ] 5, 10]  :      N/A      N/A     1.39    3.07         603
CRUISE     ]10, 15]  :      N/A      N/A     1.32    2.83         631
CRUISE     ]15, 20]  :      N/A      N/A     1.13    2.51         635
CRUISE     ]20, 25]  :      N/A      N/A     1.04    2.34         688
CRUISE     ]25, 30]  :      N/A      N/A     1.21    2.67         719
CRUISE     ]30, 35]  :      N/A      N/A     0.88    2.33         577
CRUISE     ]35, 40]  :      N/A      N/A     0.61    2.00         162
ALL CRUISES          :      N/A      N/A     1.24    2.86        4720
======================================================================


ERROR RESULTS by MASS [ton] for Dummy-TWIN:
==========================================
                          Vertical speed  Fuel consumption     Points
                          [ fpm ]  [ % ]  [kg/min] [ % ]   [dimless]
======================================================================
MCRZ CLIMB [044:050[ :     60.47    3.79     1.18    1.63        3740
MCRZ CLIMB [050:055[ :     53.05    4.46     1.25    1.72        2028
MCRZ CLIMB [055:061[ :     45.08    3.97     1.15    1.52        3485
MCRZ CLIMB [061:066[ :     38.47    3.71     1.15    1.44        2932
MCRZ CLIMB [066:072[ :     39.50    4.97     1.14    1.48        3052
MCRZ CLIMB [072:077[ :     41.39    5.96     1.09    1.37        2718
MCRZ CLIMB [077:083] :     40.39    5.54     1.14    1.37        2621
ALL MCRZ CLIMBS      :     46.51    4.63     1.16    1.51       20576
======================================================================
MCMB CLIMB [044:050[ :     59.50    2.86     1.34    1.63        3749
MCMB CLIMB [050:055[ :     53.00    3.49     1.36    1.62        1955
MCMB CLIMB [055:061[ :     44.46    2.77     1.27    1.48        3500
MCMB CLIMB [061:066[ :     38.52    2.62     1.28    1.43        3023
MCMB CLIMB [066:072[ :     39.24    3.71     1.26    1.45        3096
MCMB CLIMB [072:077[ :     41.08    3.95     1.22    1.37        2798
MCMB CLIMB [077:083] :     41.66    4.26     1.28    1.39        2735
ALL MCMB CLIMBS      :     46.14    3.38     1.28    1.49       20856
======================================================================
DESCENT    [039:044[ :     37.93    1.69     0.45    5.55        2970
DESCENT    [044:050[ :     28.46    1.24     0.45    5.51        2950
DESCENT    [050:055[ :     24.71    1.10     0.45    5.41        2883
DESCENT    [055:061[ :     25.52    1.20     0.44    5.37        2683
DESCENT    [061:066[ :     27.85    1.45     0.43    5.36        2527
DESCENT    [066:072[ :     39.03    1.99     0.44    5.33        2472
ALL DESCENTS         :     31.05    1.46     0.44    5.43       16485
======================================================================
CRUISE     [039:044[ :      N/A      N/A     0.93    2.92         822
CRUISE     [044:050[ :      N/A      N/A     1.03    2.73         760
CRUISE     [050:055[ :      N/A      N/A     1.13    2.73         726
CRUISE     [055:061[ :      N/A      N/A     1.25    2.77         679
CRUISE     [061:066[ :      N/A      N/A     1.32    2.82         332
CRUISE     [066:072[ :      N/A      N/A     1.41    2.92         610
CRUISE     [072:077[ :      N/A      N/A     1.54    3.04         548
CRUISE     [077:083] :      N/A      N/A     1.64    3.18         243
ALL CRUISES          :      N/A      N/A     1.24    2.86        4720
======================================================================
```

```
ERROR RESULTS by SPEED for Dummy-TWIN:
=====================================
                          Vertical speed  Fuel consumption    Points
                          [ fpm ]  [  % ]  [kg/min] [  % ]   [dimless]
=====================================================================
MCRZ CLIMB [M00:M25[ :    78.96    2.57     1.89    2.00         80
MCRZ CLIMB [M25:M45[ :    44.12    1.92     1.40    1.52       4489
MCRZ CLIMB [M45:M60[ :    43.82    2.61     1.12    1.37       8229
MCRZ CLIMB [M60:M70[ :    43.74    4.65     0.91    1.35       4342
MCRZ CLIMB [M70:M75[ :    45.39    6.47     1.17    1.89       1461
MCRZ CLIMB [M75:M80[ :    61.25    9.85     1.15    1.99       1577
MCRZ CLIMB [M80:M85[ :    74.80   12.36     1.02    1.85        398
ALL MCRZ CLIMBS      :    46.51    4.63     1.16    1.51      20576
=====================================================================
MCMB CLIMB [M00:M25[ :    81.92    2.19     2.21    2.03         80
MCMB CLIMB [M25:M45[ :    43.68    1.53     1.51    1.45       4485
MCMB CLIMB [M45:M60[ :    44.43    2.01     1.19    1.30       8221
MCMB CLIMB [M60:M70[ :    42.26    2.73     1.17    1.46       4311
MCMB CLIMB [M70:M75[ :    45.22    5.05     1.32    1.82       1485
MCMB CLIMB [M75:M80[ :    58.74    7.24     1.33    1.96       1760
MCMB CLIMB [M80:M85[ :    66.51    7.85     1.07    1.72        514
ALL MCMB CLIMBS      :    46.14    3.38     1.28    1.49      20856
=====================================================================
DESCENT    [M00:M25[ :    14.90    1.51     0.83    7.31        245
DESCENT    [M25:M45[ :    23.44    1.70     0.49    5.26       5191
DESCENT    [M45:M60[ :    24.79    1.18     0.45    5.79       6150
DESCENT    [M60:M70[ :    31.61    1.17     0.41    5.57       2904
DESCENT    [M70:M75[ :    66.15    2.51     0.30    4.57        771
DESCENT    [M75:M80[ :    43.46    1.32     0.23    3.90        668
DESCENT    [M80:M85[ :    54.84    1.47     0.18    3.25        556
ALL DESCENTS         :    31.05    1.46     0.44    5.43      16485
=====================================================================
CRUISE     [M00:M25[ :     N/A     N/A      2.37    8.96         36
CRUISE     [M25:M45[ :     N/A     N/A      0.81    2.41       1156
CRUISE     [M45:M60[ :     N/A     N/A      1.27    2.80       1459
CRUISE     [M60:M70[ :     N/A     N/A      1.43    3.07        857
CRUISE     [M70:M75[ :     N/A     N/A      0.99    2.09        482
CRUISE     [M75:M80[ :     N/A     N/A      1.24    2.63        387
CRUISE     [M80:M85[ :     N/A     N/A      1.80    3.72        343
ALL CRUISES          :     N/A     N/A      1.24    2.86       4720
=====================================================================
```

Figure 27 – Example of ATF file

9.6. PTF FILE FORMAT

The Performance Table File (PTF) is an ASCII file which, for a particular aircraft type, specifies cruise, climb and descent performance at different flight levels. An example of PTF file for a dummy jet aircraft is shown in Figure 28.

```
BADA PERFORMANCE FILE                                        Apr 03 2014

AC/Type: Dummy-TWIN

 Speeds:   CAS(LO/HI)  Mach   Mass Levels [kg]         Temperature: ISA
 climb   - 250/310     0.78   low     -   48000
 cruise  - 250/310     0.78   nominal -   57500        Max Alt. [ft]:  37000
 descent - 250/300     0.79   high    -   65000
======================================================================================================
 FL |          CRUISE           |              CLIMB                |              DESCENT
    |  TAS         fuel         |  TAS         ROCD          fuel   |  TAS         ROCD          fuel
    | [kts]      [kg/min]       | [kts]        [fpm]      [kg/min]  | [kts]        [fpm]      [kg/min]
    |          lo   nom    hi   |          lo   nom    hi     nom   |          lo   nom    hi     nom
======================================================================================================
  0 |  170    26.6  31.9  36.8  |  139   3200  2678  2339   120.9   |  125    609   664   704    42.4
    |                           |                                   |
  5 |  171    26.6  31.9  36.8  |  146   3319  2787  2439   120.7   |  126    613   668   709    42.3
    |                           |                                   |
 10 |  172    26.5  31.9  36.7  |  147   3316  2779  2429   119.8   |  132    644   700   741    40.9
    |                           |                                   |
 15 |  174    26.5  31.8  36.6  |  148   3310  2770  2418   118.9   |  143    703   759   801    18.8
    |                           |                                   |
 20 |  175    26.5  31.8  36.6  |  168   4019  3434  3066   119.5   |  175    944   929   970    16.9
    |                           |                                   |
 30 |  230    31.5  34.5  37.2  |  171   4086  3421  3032   117.5   |  230   1450  1357  1321    10.1
    |                           |                                   |
 40 |  233    31.3  34.3  37.0  |  205   4709  3916  3425   117.0   |  233   1464  1371  1335     9.8
    |                           |                                   |
 60 |  272    35.8  38.2  40.3  |  272   5245  4246  3649   114.4   |  272   1915  1730  1638     9.2
    |                           |                                   |
 80 |  280    35.3  37.7  39.9  |  280   5046  4077  3495   109.6   |  280   1942  1757  1665     8.8
    |                           |                                   |
100 |  289    34.9  37.2  39.4  |  357   4558  3698  3184   107.1   |  345   2933  2561  2356     8.2
    |                           |                                   |
120 |  297    34.4  36.7  38.9  |  367   4311  3489  2997   102.8   |  356   2962  2587  2381     7.8
    |                           |                                   |
140 |  378    46.3  47.9  49.4  |  378   4054  3272  2804    98.6   |  366   2988  2612  2406     7.5
    |                           |                                   |
160 |  389    45.7  47.3  48.8  |  389   3790  3049  2604    94.5   |  377   3013  2635  2429     7.2
    |                           |                                   |
180 |  401    45.1  46.8  48.3  |  401   3519  2820  2399    90.5   |  388   3036  2657  2451     6.9
    |                           |                                   |
200 |  413    44.6  46.3  47.9  |  413   3243  2587  2190    86.6   |  400   3056  2677  2471     6.7
    |                           |                                   |
220 |  425    44.1  45.9  47.5  |  425   2962  2348  1975    82.8   |  412   3074  2695  2490     6.4
    |                           |                                   |
240 |  438    43.8  45.6  47.4  |  438   2670  2098  1748    79.0   |  425   3090  2714  2511     6.2
    |                           |                                   |
260 |  452    43.9  45.9  47.7  |  452   2347  1819  1495    75.3   |  438   3115  2742  2541     6.0
    |                           |                                   |
280 |  464    44.4  46.5  48.6  |  464   2779  2115  1704    71.6   |  452   3171  2799  2599     5.8
    |                           |                                   |
290 |  462    42.7  45.0  47.1  |  462   2695  2036  1627    69.3   |  459   3228  2849  2645     5.7
    |                           |                                   |
310 |  458    39.6  42.1  44.4  |  458   2504  1858  1455    64.7   |  464   4476  3960  3686     5.6
    |                           |                                   |
330 |  454    36.8  39.5  42.1  |  454   2290  1659  1262    60.2   |  459   4111  3677  3454     5.5
    |                           |                                   |
350 |  450    34.4  37.4  40.3  |  450   2057  1442  1052    55.7   |  455   3787  3430  3257     5.4
    |                           |                                   |
370 |  447    32.4  35.8  39.1  |  447   1670  1117   762    51.6   |  453   3226  2966  2852     5.4
    |                           |                                   |
======================================================================================================
```

Figure 28 – Example of PTF file

The ACM file is generated as a result of a modelling process using MatLab [RD4]. Once this file is generated, the PTF can be automatically generated. A brief summary of the format of this file is given below.

The header of each PTF file contains information as described below:

file creation date:	Presented in line 1, at the top-right corner
aircraft type:	Presented in line 3.
speeds:	The speed laws for climb, cruise and descent are specified in lines 6, 7 and 8, that is:

climb	min($V_{cl,1}$, 250kt) / $V_{cl,2}$	M_{cl}
cruise	min($V_{cr,1}$, 250kt) / $V_{cr,2}$	M_{cr}
descent	min($V_{des,1}$, 250kt) / $V_{des,2}$	M_{des}

mass Levels: The performance tables provide data for three different mass levels in lines 6, 7 and 8, that is:

low	1.2 · OEW
nominal	OEW + 2/3 · (MTOW – OEW)
high	MTOW

Temperature data:	The temperature deviation from ISA conditions is mentioned in line 5.
Maximum altitude:	The maximum altitude as specified in the ACM file, h_{MO}, is given in line 7.

The table of performance data within the file consists of 15 columns:

Column 1	Flight level
Column 2	cruise TAS (nominal mass) [knots]
Column 3	cruise fuel consumption (low mass) [kg/min]
Column 4	cruise fuel consumption (nominal mass) [kg/min]
Column 5	cruise fuel consumption (high mass) [kg/min]
Column 6	climb TAS (nominal mass) [knots]
Column 7	rate of climb (low mass) [ft/min]
Column 8	rate of climb (nominal mass) [ft/min]
Column 9	rate of climb (high mass) [ft/min]
Column 10	climb fuel consumption (nominal mass) [kg/min]
Column 11	descent TAS (nominal mass) [knots]
Column 12	rate of descent (low mass) [ft/min]
Column 13	rate of descent (nominal mass) [ft/min]
Column 14	rate of descent (high mass) [ft/min]
Column 15	descent fuel consumption (nominal mass) [kg/min]

Further explanatory notes on the data presented in the performance tables are given below:

(a) Performance data are specified up to a maximum flight level of 510 or to the highest level for which a positive rate of climb can be achieved at the low mass.

(b) True airspeed for climb, cruise and descent is determined based on the speed schedules and operational speed envelope limitations specified in Appendix B.

(c) Rates of climb/descent and fuel flows are calculated at each flight level assuming the energy share factors associated with constant CAS or constant Mach speed laws, the aerodynamic configurations specified in Appendix B, and the following thrust settings:

- climb: MCMB rating,
- descent:
  - when in CR configuration: idle rating
  - when in AP and LD configurations: adapted thrust to follow a 3° slope
- cruise: adapted thrust for level flight, limited by available thrust (the mention '(T)' is displayed instead of the fuel flow when MCRZ (or MCMB, if MCRZ is not defined) thrust would be exceeded)

In addition, the mention '(B)' is displayed instead of the cruise fuel flow when buffet limitations would be exceeded.

(d) The fuel consumption in climb is independent of the aircraft mass above 6,000 ft and thus only one value is given. There are three different climb rates however corresponding to low, nominal and high mass conditions.

(e) The fuel consumption in descent is independent of the aircraft mass above 3,000 ft and thus only one value is given. There are three different descent rates however corresponding to low, nominal and high mass conditions.

(f) Discontinuities in climb rate can occur for the following reasons:

- change in speed between flight levels (e.g. removal of 250 kt restriction above FL100),
- transition from constant CAS to constant Mach (typically around FL300),
- transition through the tropopause (FL360 for ISA).

(g) Discontinuities in descent rate can occur for the following reasons:

- transition through the tropopause (FL360 for ISA),
- transition from constant Mach to constant CAS,
- change to approach or landing aerodynamic configuration,
- change in speed between flight levels (e.g. application of 250 kt limit below FL100).

(h) The performance data presented in the table are computed by using 'point type' calculation, that is without performing integration over time: aircraft weight is constant and does not account for consumed fuel, and speed changes take place immediately.

9.7. PTD FILE FORMAT

In addition to the data provided in the PTF file, more detailed climb and descent performance data are presented in the PTD file. An example of a PTD file for a dummy jet aircraft is shown in Figure 29 (partial listing).

```
Low mass CLIMB
==============

 FL     T        p     rho       a      TAS      CAS      M      mass   Thrust   Drag    Fuel    ESF   ROCD   gamma Conf  Lim
[-]    [K]     [Pa]  [kg/m3]  [m/s]   [kt]     [kt]     [-]     [kg]     [N]     [N]    [kgm]    [-]   [fpm]  [deg]  [-]
  0  288.15  101325  1.225   340.3  127.04  127.04  0.192   48000  160127  40616  119.56  0.980  3200  14.40  TO
  5  287.16   99508  1.207   339.7  132.99  132.03  0.201   48000  158830  40177  119.38  0.978  3319  14.27  IC    C
 10  286.17   97717  1.190   339.1  133.97  132.04  0.203   48000  157882  40178  118.58  0.977  3316  14.15  IC    C
 15  285.18   95952  1.172   338.5  134.96  132.05  0.205   48000  156883  40180  117.75  0.977  3310  14.02  IC    C
 20  284.19   94213  1.155   337.9  153.06  148.69  0.233   48000  153824  28082  118.35  0.971  4019  15.03  CR    C
 30  282.21   90812  1.121   336.8  158.81  152.04  0.243   48000  150918  27411  116.65  0.968  4086  14.72  CR
 40  280.23   87511  1.088   335.6  192.88  182.04  0.296   48000  143831  24880  116.38  0.954  4709  13.95  CR
 60  276.26   81200  1.024   333.2  272.30  250.00  0.420   48000  127496  29465  114.43  0.913  5245  10.96  CR
 80  272.30   75262  0.963   330.8  280.34  250.00  0.436   48000  121594  29412  109.62  0.908  5046  10.24  CR
100  268.34   69682  0.905   328.4  356.65  310.00  0.559   48000  108163  39170  107.05  0.861  4558   7.25  CR
120  264.38   64441  0.849   326.0  367.06  310.00  0.579   48000  103108  39107  102.80  0.853  4311   6.66  CR
140  260.41   59524  0.796   323.5  377.86  310.00  0.601   48000   98100  39045   98.62  0.845  4054   6.08  CR
160  256.45   54915  0.746   321.0  389.07  310.00  0.623   48000   93156  38976   94.52  0.836  3790   5.52  CR
180  252.49   50600  0.698   318.5  400.70  310.00  0.647   48000   88290  38892   90.50  0.826  3519   4.97  CR
200  248.53   46563  0.653   316.0  412.77  310.00  0.672   48000   83513  38792   86.58  0.817  3243   4.45  CR
220  244.56   42791  0.610   313.5  425.28  310.00  0.698   48000   78834  38693   82.75  0.807  2962   3.94  CR
240  240.60   39271  0.569   311.0  438.26  310.00  0.725   48000   74258  38682   79.01  0.796  2670   3.45  CR
260  236.64   35989  0.530   308.4  451.70  310.00  0.754   48000   69788  39027   75.35  0.785  2347   2.94  CR
280  232.68   32932  0.493   305.8  463.64  308.55  0.780   48000   65470  39864   71.64  1.088  2779   3.39  CR
290  230.70   31485  0.475   304.5  461.66  302.03  0.780   48000   63515  38581   69.32  1.088  2695   3.30  CR
310  226.73   28745  0.442   301.9  457.68  289.21  0.780   48000   59591  36218   64.71  1.088  2504   3.10  CR
330  222.77   26201  0.410   299.2  453.66  276.67  0.780   48000   55684  34121   60.17  1.088  2290   2.86  CR
350  218.81   23842  0.380   296.5  449.61  264.42  0.780   48000   51833  32288   55.73  1.088  2057   2.59  CR
370  216.65   21663  0.348   295.1  447.38  252.49  0.780   48000   48075  30721   51.63  1.000  1670   2.11  CR

[...]

High mass DESCENT
=================

 FL     T        p     rho       a      TAS      CAS      M      mass   Thrust   Drag    Fuel    ESF   ROCD   gamma Conf  Lim
[-]    [K]     [Pa]  [kg/m3]  [m/s]   [kt]     [kt]     [-]     [kg]     [N]     [N]    [kgm]    [-]   [fpm]  [deg]  [-]
  0  288.15  101325  1.225   340.3  132.81  132.81  0.201   65000   58568  92682   48.17  0.978   704  -3.00  LD
  5  287.16   99508  1.207   339.7  133.78  132.81  0.203   65000   58563  92690   48.11  0.978   709  -3.00  LD
 10  286.17   97717  1.190   339.1  139.82  137.81  0.212   65000   55778  89977   46.51  0.975   741  -3.00  LD
 15  285.18   95952  1.172   338.5  151.06  147.81  0.230   65000   21419  55760   21.63  0.971   801  -3.00  AP
 20  284.19   94213  1.155   337.9  183.01  177.81  0.279   65000   14486  49280   18.74  0.959   970  -3.00  AP
 30  282.21   90812  1.121   336.8  229.62  220.00  0.351   65000   -4936  33722   10.09  0.937  1321  -3.26  CR
 40  280.23   87511  1.088   335.6  232.96  220.00  0.357   65000   -4857  33728    9.83  0.935  1335  -3.24  CR
 60  276.26   81200  1.024   333.2  272.30  250.00  0.420   65000   -6323  35143    9.22  0.913  1638  -3.41  CR
 80  272.30   75262  0.963   330.8  280.34  250.00  0.436   65000   -6059  35125    8.78  0.908  1665  -3.36  CR
100  268.34   69682  0.905   328.4  345.37  300.00  0.541   65000   -8261  41211    8.18  0.868  2356  -3.86  CR
120  264.38   64441  0.849   326.0  355.51  300.00  0.561   65000   -7851  41168    7.82  0.860  2381  -3.79  CR
140  260.41   59524  0.796   323.5  366.04  300.00  0.582   65000   -7432  41129    7.48  0.852  2406  -3.72  CR
160  256.45   54915  0.746   321.0  376.97  300.00  0.604   65000   -7007  41088    7.18  0.843  2429  -3.65  CR
180  252.49   50600  0.698   318.5  388.32  300.00  0.627   65000   -6576  41043    6.91  0.834  2451  -3.57  CR
200  248.53   46563  0.653   316.0  400.10  300.00  0.651   65000   -6142  40992    6.66  0.825  2471  -3.50  CR
220  244.56   42791  0.610   313.5  412.32  300.00  0.677   65000   -5705  40948    6.42  0.815  2490  -3.42  CR
240  240.60   39271  0.569   311.0  425.00  300.00  0.703   65000   -5267  40961    6.21  0.804  2511  -3.34  CR
260  236.64   35989  0.530   308.4  438.16  300.00  0.731   65000   -4828  41168    6.01  0.794  2541  -3.28  CR
280  232.68   32932  0.493   305.8  451.80  300.00  0.760   65000   -4390  41872    5.83  0.783  2599  -3.26  CR
290  230.70   31485  0.475   304.5  458.81  300.00  0.775   65000   -4172  42544    5.74  0.777  2645  -3.26  CR
310  226.73   28745  0.442   301.9  463.54  293.28  0.790   65000   -3556  42341    5.61  1.091  3686  -4.50  CR
330  222.77   26201  0.410   299.2  459.48  280.58  0.790   65000   -2818  40564    5.52  1.091  3454  -4.26  CR
350  218.81   23842  0.380   296.5  455.37  268.17  0.790   65000   -2131  39147    5.44  1.091  3257  -4.05  CR
```

Figure 29 – Example of PTD file

The performance values presented in the PTD file are a superset of the performance values presented in the PTF file. They are generated in the same conditions as the corresponding PTF file: same aircraft, same source ACM file, same speed laws, same mass values, same temperature and same flight levels. The purpose of this file is mainly to provide the user with a greater number of computed parameters, especially intermediate parameters used to compute the final TAS and ROCD, which may be useful to validate an implementation of the BADA model.

The files contains performance data consisting of 9 sections:

- low/nominal/high mass climb performance
- low/nominal/high mass descent performance
- low/nominal/high mass cruise performance

Each section contains a table that presents, for several flight levels, a set of performance parameters spread across 16 columns:

Column 1	Flight level [FL]
Column 2	Temperature [K]
Column 3	Pressure [Pa]
Column 4	Air density [kg/m^3]
Column 5	Speed of sound [m/s]
Column 6	TAS [kt]
Column 7	CAS [kt]
Column 8	Mach number [-]
Column 9	Mass [kg]
Column 10	Thrust [N]
Column 11	Drag [N]
Column 12	Fuel flow [kg/min]
Column 13	Energy share factor [-]
Column 14	Rate of climb/descent [ft/min] (absolute value)
Column 15	Flight path angle [degree]
Column 16	Aerodynamic configuration (TO, IC, CR, AP or LD)
Column 17	Flight envelope limitations checks: • T: available thrust exceeded (cruise only) • C: nominal speed from ARPM has been corrected to fit into the speed envelope • v: minimum speed exceeded • V: maximum speed exceeded

9.8. OPT FILES FORMAT

The Optimized Performance Tables (OPT) are ASCII files that contain tables of pre-computed optimum flight parameters (either speed schedule of cruise flight level) at various flight conditions.

Note:

- OPT files are not provided for aircraft types equipped with piston engines.
- The ECON OPT file is provided only for aircraft types equipped with turbofan engines.

These files share the same header:

- line 1: file description and date of generation
- line 3: engine type
- line 4: name of the aircraft model
- line 5: engine rating used for the computation (empty for cruise optimizations)
- line 6: number of tables contained in the file.

In addition, all the tables contained in each file are preceded by a subheader that provides information about their structure and content:

- line 1: table name
- line 2: table type, which defines the number of dimensions of the table (e.g. 1D or 2D)
- line 3: table variables, i.e. the parameters on which the table is indexed (with their respective units if any): those parameters correspond to the flight conditions on which the optimization result depends
- line 4: table dimension, i.e. the size of the table in each dimension (e.g. 10x1 for a 1D table, or 10x20 for a 2D table)
- line 5: table results, i.e. the kind of data provided as output of the table (with their respective units if any): those data correspond to the result of the optimization.

The following subsections provide the details specific to each OPT file, as well as an example of each such file.

9.8.1. Maximum range cruise

The maximum range cruise OPT file, named *MRC.opt*, contains some precomputed results of the optimization described in section 5.2.1.1, in the case of ISA conditions. Note that several input parameters to this optimization, namely the pressure altitude H_P and aircraft weight W, are combined in a single parameter, the weight coefficient C_W [-]:

$$C_W = W \cdot \delta^{-1} \cdot W_{MTOW}^{-1} \tag{9.8-1}$$

Where :

W is the weight force at current aircraft mass [N], see section 3.1

δ is the pressure ratio [-]

MTOW is the maximum take-off weight [kg], from the DLM

W_{MTOW} is the weight force at MTOW [N], see section 3.1

The data are organized into one 1D table whose structure is the following:

- **Input**: the weight coefficient C_W [-]
- **Output**: the maximum range cruise Mach number, M_{mrc} [-]

Figure 30 presents an example of maximum range cruise OPT file for a dummy jet aircraft.

```
BADA MAXIMUM RANGE CRUISE FILE                                  Jul 27 2011

Type: JET
AC: Dummy-JET
Engine rating:
Number of tables: 1

Table name: MRC
Table type: 1D
Table variables: CW
Table dimension: 21x1
Table results: Mmrc [-]

        CW|       Mmrc
=====================
  1.258019|  0.436925
  1.428988|  0.465496
  1.599957|  0.491738
  1.770925|  0.515938
  1.941894|  0.538320
  2.112863|  0.559073
  2.283831|  0.578370
  2.454800|  0.596384
  2.625769|  0.613301
  2.796737|  0.629330
  2.967706|  0.644708
  3.138675|  0.659716
  3.309644|  0.674696
  3.480612|  0.690059
  3.651581|  0.706223
  3.822550|  0.723109
  3.993518|  0.738838
  4.164487|  0.750713
  4.335456|  0.758610
  4.506424|  0.763755
  4.677393|  0.767113
```

Figure 30 – Example of MRC OPT file

9.8.2. Long range cruise

The long range cruise OPT file, named *LRC.opt*, contains some precomputed results of the optimization described in section 5.2.1.2, in the case of ISA conditions. Note that several input parameters to this optimization, namely the pressure altitude H_P and aircraft weight W, are combined in a single parameter, the weight coefficient C_W [-] (see section 9.8.1).

The data are organized into one 1D table whose structure is the following:

- **Input**: the weight coefficient C_W [-]
- **Output**: the long range cruise Mach number, M_{lrc} [-]

Figure 31 presents an example of long range cruise OPT file for a dummy jet aircraft.

```
BADA LONG RANGE CRUISE FILE                                       Jul 27 2011

Type: JET
AC: Dummy-JET
Engine rating:
Number of tables: 1

Table name: LRC
Table type: 1D
Table variables: CW
Table dimension: 21x1
Table results: Mlrc [-]

        CW|       Mlrc
=====================
  1.258019|  0.475747
  1.428988|  0.506129
  1.599957|  0.533804
  1.770925|  0.559093
  1.941894|  0.582254
  2.112863|  0.603516
  2.283831|  0.623099
  2.454800|  0.641237
  2.625769|  0.658187
  2.796737|  0.674240
  2.967706|  0.689719
  3.138675|  0.704982
  3.309644|  0.720374
  3.480612|  0.736022
  3.651581|  0.751253
  3.822550|  0.764172
  3.993518|  0.773299
  4.164487|  0.779075
  4.335456|  0.782634
  4.506424|  0.784769
  4.677393|  0.785933
```

Figure 31 – Example of LRC OPT file

9.8.3. Maximum endurance cruise

The maximum endurance cruise OPT file, named *MEC.opt*, contains some precomputed results of the optimization described in section 5.2.1.3, in the case of ISA conditions. Note that several input parameters to this optimization, namely the pressure altitude H_P and aircraft weight W, are combined in a single parameter, the weight coefficient C_W [-] (see section 9.8.1).

The data are organized into one 1D table whose structure is the following:

- **Input**: the weight coefficient C_W [-]
- **Output**: the maximum endurance cruise Mach number, M_{mec} [-]

Figure 32 presents an example of maximum range cruise OPT file for a dummy jet aircraft.

9.8.4. Optimum altitude

The optimum altitude OPT file, named *OPTALT.opt*, contains some precomputed results of the optimization described in section 5.2.1.4, in the case of ISA conditions.

The data are organized into one 2D table whose structure is the following:

- **Inputs**: the aircraft mass W [kg] and Mach number M [-]
- **Output**: the geopotential pressure altitude that maximizes the aircraft range, $H_{p,opt}$ [ft]

Figure 33 presents an example of optimum altitude OPT file for a dummy jet aircraft.

```
BADA MAXIMUM ENDURANCE CRUISE FILE                                  Jul 27 2011

Type: JET
AC: Dummy-JET
Engine rating:
Number of tables: 1

Table name: MEC
Table type: 1D
Table variables: CW
Table dimension: 41x1
Table results: Mmec [-]

        CW|      Mmec
=====================
  0.615385|  0.225439
  0.716935|  0.245153
  0.818485|  0.263598
  0.920035|  0.281006
  1.021585|  0.297548
  1.123136|  0.313352
  1.224686|  0.328519
  1.326236|  0.343127
  1.427786|  0.357240
  1.529336|  0.370910
  1.630887|  0.384180
  1.732437|  0.397086
  1.833987|  0.409659
  1.935537|  0.421925
  2.037088|  0.433904
  2.138638|  0.445615
  2.240188|  0.457076
  2.341738|  0.468299
  2.443288|  0.479297
  2.544839|  0.490082
  2.646389|  0.500663
  2.747939|  0.511051
  2.849489|  0.521253
  2.951039|  0.531281
  3.052590|  0.541145
  3.154140|  0.550854
  3.255690|  0.560422
  3.357240|  0.569864
  3.458790|  0.579195
  3.560341|  0.588435
  3.661891|  0.597608
  3.763441|  0.606742
  3.864991|  0.615873
  3.966542|  0.625042
  4.068092|  0.634302
  4.169642|  0.643719
  4.271192|  0.653379
  4.372742|  0.663389
  4.474293|  0.673882
  4.575843|  0.685012
  4.677393|  0.696882
```

Figure 32 – Example of MEC OPT file

```
BADA OPTIMAL ALTITUDE FILE                                   Jul 27 2011

Type: JET
AC: Dummy-JET
Engine rating:
Number of tables: 1

Table name: OPTALT
Table type: 2D
Table variables: W [kg], M
Table dimension: 11x21
Table results: Hpopt [ft]

         W|    40000    42500    45000    47500    50000    52500    55000    57500    60000    62500    65000
         M|
========================================================================================================================
  0.200000|     2000     2000     2000     2000     2000     2000     2000     2000     2000     2000     2000
  0.230500|     2000     2000     2000     2000     2000     2000     2000     2000     2000     2000     2000
  0.261000|     6771     5162     3628     2162     2000     2000     2000     2000     2000     2000     2000
  0.291500|    12374    10830     9358     7951     6603     5308     4062     2861     2000     2000     2000
  0.322000|    17100    15611    14191    12834    11533    10285     9083     7925     6807     5726     4679
  0.352500|    21148    19706    18331    17016    15757    14547    13384    12262    11179    10132     9119
  0.383000|    24654    23252    21916    20639    19415    18240    17109    16019    14967    13949    12964
  0.413500|    27715    26349    25047    23802    22609    21464    20362    19299    18274    17282    16322
  0.444000|    30406    29071    27799    26582    25416    24297    23220    22182    21180    20211    19273
  0.474500|    32781    31474    30228    29037    27895    26799    25744    24728    23746    22797    21879
  0.505000|    34888    33605    32383    31214    30093    29018    27982    26985    26022    25091    24189
  0.535500|    36766    35503    34301    33151    32050    30993    29975    28994    28048    27132    26246
  0.566000|    37000    37000    36016    34884    33800    32759    31757    30792    29859    28958    28086
  0.596500|    37000    37000    37000    36446    35376    34350    33362    32410    31491    30603    29743
  0.627000|    37000    37000    37000    37000    36817    35801    34827    33887    32980    32103    31254
  0.657500|    37000    37000    37000    37000    37000    37000    36198    35270    34374    33508    32670
  0.688000|    37000    37000    37000    37000    37000    37000    37000    36629    35743    34888    34060
  0.718500|    37000    37000    37000    37000    37000    37000    37000    37000    37000    36335    35517
  0.749000|    37000    37000    37000    37000    37000    37000    37000    37000    37000    37000    37000
  0.779500|    37000    37000    37000    37000    37000    37000    37000    37000    37000    37000    37000
  0.810000|    37000    37000    37000    37000    37000    37000    37000    37000    37000    37000    37000
```

Figure 33 – Example of optimum altitude OPT file

9.8.5. ECON cruise

The economy cruise OPT file, named *ECON.opt*, contains some precomputed results of the optimization described in section 5.2.2.1, in the case of ISA conditions. Note that several input parameters to this optimization are combined in single parameters:

- the pressure altitude H_P and aircraft weight W are combined in the weight coefficient C_W [-] (see section 9.8.1)
- the pressure altitude H_P and cost index CI are combined in the cost index coefficient C_{CI} [-]:

$$C_{CI} = CI \cdot L_{HV} \cdot \delta^{-1} \cdot W_{MTOW}^{-1} \cdot a^{-1} \tag{9.8-2}$$

Where :

CI is the cost index [kg/s][4], see section 5.2.2

L_{HV} is the fuel lower heating value [m^2/s^2], from the PFM

δ is the pressure ratio [-]

MTOW is the maximum take-off weight [kg], from the DLM

W_{MTOW} is the weight force at MTOW [N], see section 3.1

a is the speed of sound [m/s], see expression (2.2-22)

The data are organized into one 2D table whose structure is the following:

- **Inputs**: the weight coefficient C_W [-] and cost index coefficient C_{CI} [-]
- **Output**: the economy Mach number, M_{ECON} [-]

Figure 34 presents an example of economy cruise OPT file for a dummy jet aircraft.

[4] Unit and maximum value of the cost index differ between aircraft manufacturers and types, as well as FMS manufacturers. While section 5.2.2 defines the cost index in [kg/min], this sample OPT file uses a different unit.

```
BADA ECONOMY CRUISE FILE                                          Jul 27 2011

Type: JET
AC: Dummy-JET
Engine rating:
Number of tables: 1

Table name: ECON
Table type: 2D
Table variables: CCI, CW
Table dimension: 6x21
Table results: Mecon [-]

       CCI| -0.000466  0.093185  0.415354  0.737523  1.059692  1.766154
        CW|
=======================================================================
  1.258019|  0.436535  0.519369  0.715858  0.758916  0.776692  0.796337
  1.428988|  0.465139  0.539970  0.718658  0.760249  0.777652  0.796913
  1.599957|  0.491411  0.559241  0.721716  0.761752  0.778727  0.797548
  1.770925|  0.515639  0.577234  0.725025  0.763422  0.779912  0.798235
  1.941894|  0.538046  0.594031  0.728586  0.765253  0.781197  0.798963
  2.112863|  0.558822  0.609736  0.732402  0.767237  0.782568  0.799722
  2.283831|  0.578140  0.624482  0.736476  0.769356  0.784008  0.800497
  2.454800|  0.596174  0.638426  0.740805  0.771585  0.785492  0.801271
  2.625769|  0.613109  0.651752  0.745367  0.773886  0.786989  0.802028
  2.796737|  0.629153  0.664671  0.750107  0.776209  0.788465  0.802746
  2.967706|  0.644545  0.677427  0.754929  0.778494  0.789879  0.803406
  3.138675|  0.659564  0.690294  0.759683  0.780672  0.791191  0.803985
  3.309644|  0.674553  0.703552  0.764193  0.782676  0.792359  0.804460
  3.480612|  0.689922  0.717342  0.768287  0.784444  0.793343  0.804807
  3.651581|  0.706091  0.731231  0.771834  0.785922  0.794106  0.804999
  3.822550|  0.722989  0.743784  0.774763  0.787071  0.794615  0.805007
  3.993518|  0.738746  0.753604  0.777058  0.787861  0.794840  0.804796
  4.164487|  0.750653  0.760619  0.778738  0.788278  0.794754  0.804329
  4.335456|  0.758570  0.765464  0.779846  0.788315  0.794337  0.803564
  4.506424|  0.763727  0.768751  0.780437  0.787980  0.793578  0.802464
  4.677393|  0.767092  0.770914  0.780575  0.787299  0.792486  0.801003
```

Figure 34 – Example of economy cruise OPT file

9.9. GPF FILE FORMAT

The GPF file is an XML file which specifies the values of the global aircraft parameters (see Section 8). The format of this file is described in an XML schema presented in section 9.9.1, and the content of the file is shown in section 9.9.2.

9.9.1. Format of the GPF file

The format of the GPF file is described in an XML schema, which can be used to guide an XML parser in the reading of the GFP XML file. This XML schema is provided in a file named *GPF_BADA4.xsd*, and its content is presented in Figure 35.

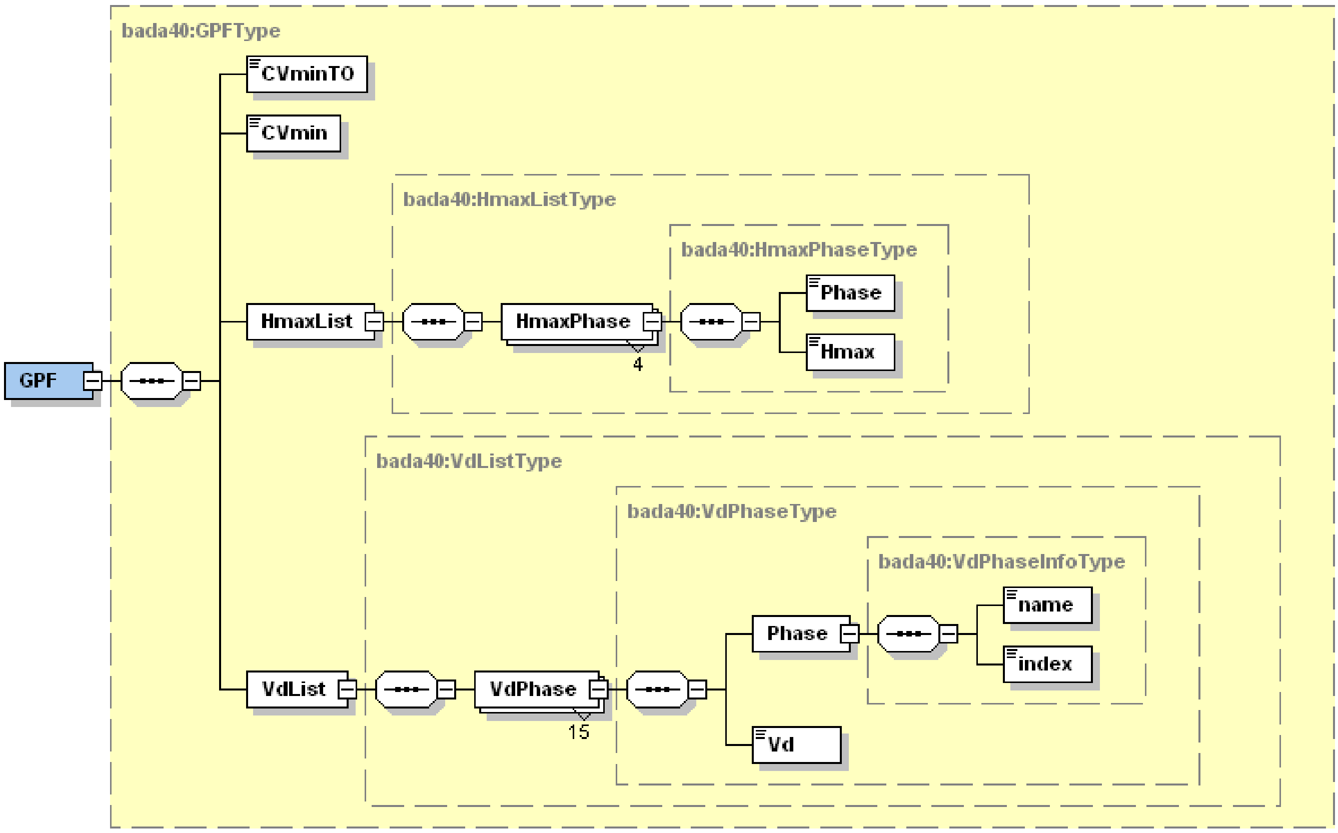

Figure 35 – XML schema of the GPF file

9.9.2. Content of the GPF file

The content of the GPF file is presented in Figure 36. For the sake of clarity, some items have been collapsed.

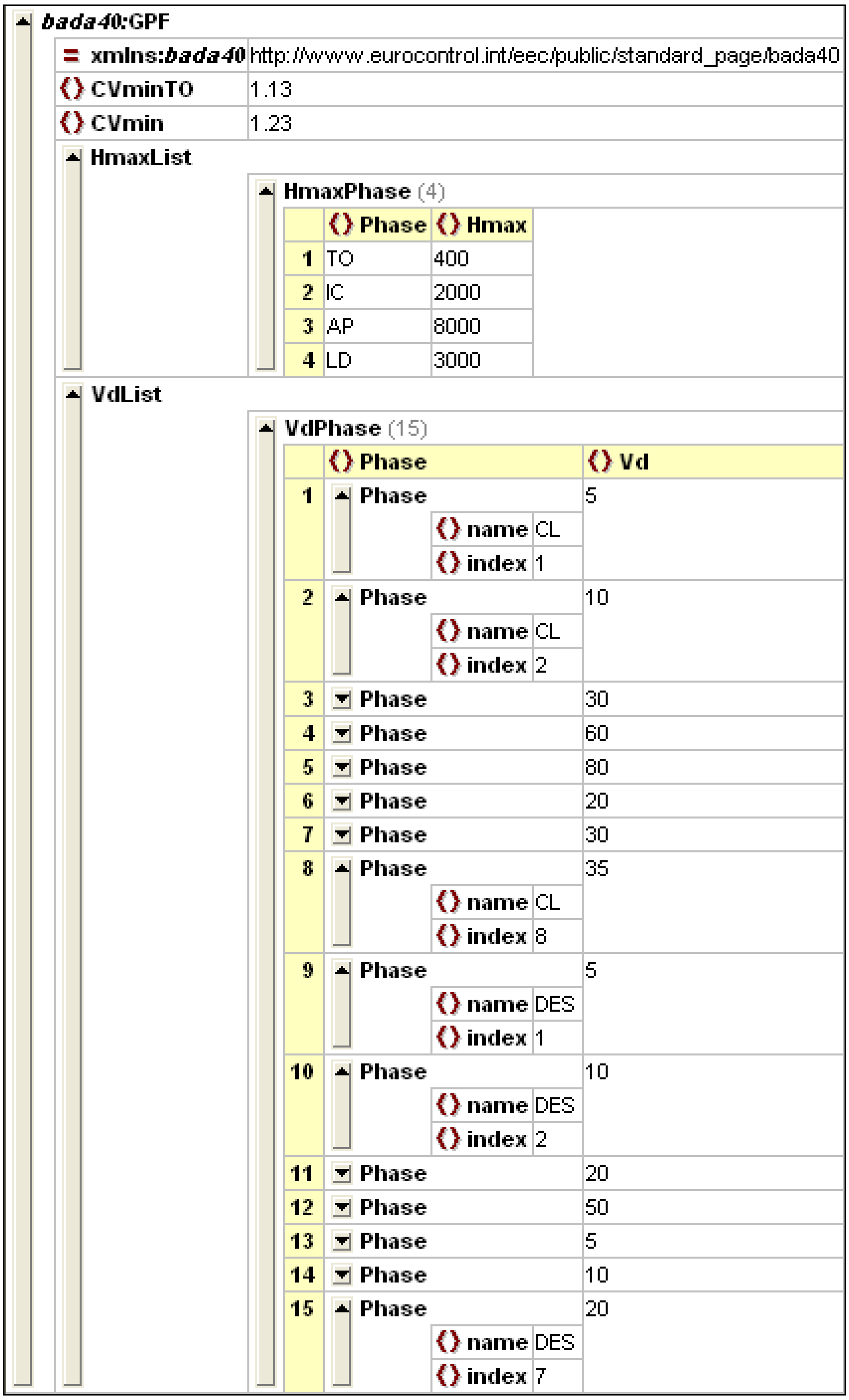

Figure 36 – Content of the GPF file

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APPENDIX A

SOLUTIONS FOR PROPELLER EFFICIENCY ALGORITHM

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A general solution for finding the roots of a cubic expression can be found in [RD6].

If we take expression (3.3-21), we can rewrite it to:

$$\eta^3 + \left(\frac{n_{eng}}{\dot{W}_P}\cdot\sigma\cdot\rho_0\cdot D_P^2\cdot\frac{\pi}{2}\cdot V_{TAS}^3\cdot\eta_{max}\right)\cdot\eta - \left(\frac{n_{eng}}{\dot{W}_P}\cdot\sigma\cdot\rho_0\cdot D_P^2\cdot\frac{\pi}{2}\cdot V_{TAS}^3\cdot\eta_{max}^2\right) = 0$$

Let:

$$a_1 = 0$$

$$a_2 = \frac{n_{eng}}{\dot{W}_P}\cdot\sigma\cdot\rho_0\cdot D_P^2\cdot\frac{\pi}{2}\cdot V_{TAS}^3\cdot\eta_{max}$$

$$a_3 = -\frac{n_{eng}}{\dot{W}_P}\cdot\sigma\cdot\rho_0\cdot D_P^2\cdot\frac{\pi}{2}\cdot V_{TAS}^3\cdot\eta_{max}^2$$

Now let:

$$Q = \frac{3\cdot a_2 - a_1^{\,2}}{9} = \frac{n_{eng}}{\dot{W}_P}\cdot\sigma\cdot\rho_0\cdot D_P^2\cdot\frac{\pi}{6}\cdot V_{TAS}^3\cdot\eta_{max}$$

$$R = \frac{9\cdot a_1\cdot a_2 - 27\cdot a_3 - 2\cdot a_1^{\,3}}{54} = \frac{n_{eng}}{\dot{W}_P}\cdot\sigma\cdot\rho_0\cdot D_P^2\cdot\frac{\pi}{4}\cdot V_{TAS}^3\cdot\eta_{max}^2$$

The discrimant is then:

$$D = Q^3 + R^2$$

In our case, we have always Q > 0 and R > 0 so D > 0, which means that one root is real and two complex conjugate.

Let:

$$S = \sqrt[3]{R + \sqrt{D}}$$

$$T = \sqrt[3]{R - \sqrt{D}}$$

The real root is then equal to:

$$\eta_1 = S + T - \frac{a_1}{3} = S + T$$

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APPENDIX B

AIRLINE PROCEDURE MODEL

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This section defines a set of standard airline procedures, parameterised by the BADA Airline Procedure Model, which can be used when no better knowledge is available to the user about the way the aircraft is operated. Definition of the standard airline procedures in BADA is driven by a requirement to provide means of simulating standard or nominal aircraft operations using different simulation and modelling tools for various ATM applications.

The BADA airline procedure model provides speed schedules for three separate flight phases (climb, cruise and descent), aerodynamic configurations for five flight segments (take-off, initial climb, en-route/cruise, approach and landing), and operational envelope limitations. For each flight phase and each aircraft model, the BADA airline procedure model requires the following information to determine the aircraft speed schedule:

1. BADA airline procedure default speeds provided in Airline Procedure Model (ARPM):

   V_1 - standard CAS [knots] below 10,000 ft;

   V_2 - standard CAS [knots] between 10,000 ft and Mach transition altitude;

   M - standard Mach number above Mach transition altitude;

   where the Mach transition altitude is defined in Section 2.2.2 k)
2. Stall speeds for take-off and landing configurations, computed from the corresponding maximum lift coefficients;
3. Coefficients provided in sections 8.3 and 8.4.

The process followed to define the BADA airline procedure default speeds and choose the aircraft configurations for each flight phase is described in [RD4]. The airline procedure model below 10,000 ft with corresponding coefficients (mentioned under item 3 above) has been defined taking into account aircraft manufacturer's performance reference data and aircraft operational data available at EUROCONTROL.

It is widely recognised that the way an aircraft is operated varies significantly depending on specific airspace procedures and operating policies of locally dominant airlines. For that reason, the resulting speed schedules of the BADA standard airline procedure model may differ from a geographical location or of an aerospace's specific aircraft operation.

To account for the local aircraft operation characteristics and improve conformance of the simulated aircraft behaviour with real operations, the user of BADA is given a possibility to modify the BADA default speeds (as provided in ACM file). The change of speed related ARPM parameters should be done in accordance with the BADA modelling procedure described in the Chapter 2.2.3 of [RD4]. The stall speeds (computed from maximum lift coefficients) and coefficients detailed in Section 8.3 and 8.4, however, are not subject to modification by the BADA user.

The altitude levels used for determination of CAS speed schedules and provided in the following chapters are expressed in terms of geopotential pressure altitude. Different reference datums for altitude measurement may however be applied depending on the user application and its functional design choices, such as use of standard operational pressure settings used in aviation: QNH for MSL pressure, QFE for pressure at the airport reference point or QNE corresponding to standard MSL 1013 hPa (these can be selected through the altimeter's pressure setting knob in the aircraft). The BADA Airline Procedure Model only identifies the possibility to introduce notion of different altimetry settings for calculation of the CAS speed schedules in the user application: the implementation decision is left to the application owner.

❖ Climb speeds

The following parameters are defined for each aircraft type to characterise the climb phase:

$V_{cl,1}$	-	standard climb CAS [knots] between 1,500/6,000 and 10,000 ft
$V_{cl,2}$	-	standard climb CAS [knots] between 10,000 ft and Mach transition altitude
M_{cl}	-	standard climb Mach number above Mach transition altitude

Note that the climb speed schedule shall determine an increasing speed from take-off to $V_{cl,1}$. To ensure that monotony, it is recommended to determine the speed schedule from the highest altitude to the lowest one, and to use at each step the speed of the higher altitude range as a ceiling value for the lower altitude range.

- For jet aircraft the following CAS schedule is assumed, based on the parameters mentioned above and the take-off stall speed:

from 0 to 1,499 ft	$C_{Vmin} \cdot (V_{stall})_{TO} + Vd_{CL,1}$	(B.1-1)
from 1,500 to 2,999 ft	$C_{Vmin} \cdot (V_{stall})_{TO} + Vd_{CL,2}$	(B.1-2)
from 3,000 to 3,999 ft	$C_{Vmin} \cdot (V_{stall})_{TO} + Vd_{CL,3}$	(B.1-3)
from 4,000 to 4,999 ft	$C_{Vmin} \cdot (V_{stall})_{TO} + Vd_{CL,4}$	(B.1-4)
from 5,000 to 5,999 ft	$C_{Vmin} \cdot (V_{stall})_{TO} + Vd_{CL,5}$	(B.1-5)
from 6,000 to 9,999 ft	$\min (V_{cl,1}, 250 \text{ kt})$	
from 10,000 ft to Mach transition altitude	$V_{cl,2}$	
above Mach transition altitude	M_{cl}	

- For turboprop and piston aircraft the following CAS schedule is assumed:

from 0 to 499 ft	$C_{Vmin} \cdot (V_{stall})_{TO} + Vd_{CL,6}$	(B.1-6)
from 500 to 999 ft	$C_{Vmin} \cdot (V_{stall})_{TO} + Vd_{CL,7}$	(B.1-7)
from 1,000 to 1,499 ft	$C_{Vmin} \cdot (V_{stall})_{TO} + Vd_{CL,8}$	(B.1-8)
from 1,500 to 9,999 ft	$\min (V_{cl,1}, 250 \text{ kt})$	
from 10,000 ft to Mach transition altitude	$V_{cl,2}$	
above Mach transition altitude	M_{cl}	

The take-off stall speed, $(V_{stall})_{TO}$, is computed as follows:

1. Determine, from the ARPM\AeroConfSchedule section of the ACM file, the aerodynamic configuration to be used for take-off (TO).
2. Compute the minimum Mach number in this aerodynamic configuration using section 6.3.3, taking into account the take-off mass and altitude.
3. Convert this Mach number to TAS using formula 2.2-26, then to CAS using formula 2.2-24.

The values for $Vd_{CL,i}$ can be found in Section 8.4.

❖ Cruise speeds

The following parameters are defined for each aircraft type to characterise the cruise phase:

$V_{cr,1}$	-	standard cruise CAS [knots] between 3,000 and 10,000 ft
$V_{cr,2}$	-	standard cruise CAS [knots] between 10,000 ft and Mach transition altitude
M_{cr}	-	standard cruise Mach number above Mach transition altitude

- For jet aircraft the following CAS schedule is assumed:

from 0 to 2,999 ft	min ($V_{cr,1}$, 170 kt)
from 3,000 to 5,999 ft	min ($V_{cr,1}$, 220 kt)
from 6,000 to 13,999 ft	min ($V_{cr,1}$, 250 kt)
from 14,000 ft to Mach transition altitude	$V_{cr,2}$
above Mach transition altitude	M_{cr}

- For turboprop and piston aircraft the following CAS schedule is assumed:

from 0 to 2,999 ft	min ($V_{cr,1}$, 150 kt)
from 3,000 to 5,999 ft	min ($V_{cr,1}$, 180 kt)
from 6,000 to 9,999 ft	min ($V_{cr,1}$, 250 kt)
from 10,000 ft to Mach transition altitude	$V_{cr,2}$
above Mach transition altitude	M_{cr}

❖ Descent speeds

The following parameters are defined for each aircraft type to characterise the descent phase:

$V_{des,1}$	-	standard descent CAS [knots] between 3,000/6,000 and 10,000 ft
$V_{des,2}$	-	standard descent CAS [knots] between 10,000 ft and Mach transition altitude
M_{des}	-	standard descent Mach number above Mach transition altitude

Note that the descent speed schedule shall determine a decreasing speed from $V_{des,1}$ to landing. To ensure that monotony, it is recommended to evaluate the speed schedule from the highest altitude to the lowest one, and to use at each step the speed of the higher altitude range as a ceiling value for the lower altitude range.

- For jet and turboprop aircraft the following CAS schedule is assumed, based on the above parameters and the landing stall speed:

from 0 to 999 ft	$C_{Vmin} \cdot (V_{stall})_{LD} + Vd_{DES,1}$	(B.3-1)
from 1,000 to 1,499 ft	$C_{Vmin} \cdot (V_{stall})_{LD} + Vd_{DES,2}$	(B.3-2)
from 1,500 to 1,999 ft	$C_{Vmin} \cdot (V_{stall})_{LD} + Vd_{DES,3}$	(B.3-3)
from 2,000 to 2,999 ft	$C_{Vmin} \cdot (V_{stall})_{LD} + Vd_{DES,4}$	(B.3-4)
from 3,000 to 5,999 ft	$\min(V_{des,1}, 220)$	
from 6,000 to 9,999 ft	$\min(V_{des,1}, 250)$	
from 10,000 ft to Mach transition altitude	$V_{des,2}$	
above Mach transition altitude	M_{des}	

- For piston aircraft the following CAS schedule is assumed:

from 0 to 499 ft	$C_{Vmin} \cdot (V_{stall})_{LD} + Vd_{DES,5}$	(B.3-5)
from 500 to 999 ft	$C_{Vmin} \cdot (V_{stall})_{LD} + Vd_{DES,6}$	(B.3-6)
from 1000 to 1,499 ft	$C_{Vmin} \cdot (V_{stall})_{LD} + Vd_{DES,7}$	(B.3-7)
from 1,500 to 9,999 ft	$V_{des,1}$	
from 10,000 ft to Mach transition altitude	$V_{des,2}$	
above Mach transition altitude	M_{des}	

The landing stall speed, $(V_{stall})_{LD}$, is computed as follows:

1. Determine, from the ARPM\AeroConfSchedule section of the ACM file, the aerodynamic configuration to be used for landing (LD).
2. Compute the minimum Mach number in this aerodynamic configuration using section 6.3.3, taking into account the landing mass and altitude.
3. Convert this Mach number to TAS using formula 2.2-26, then to CAS using formula 2.2-24.

The values for $Vd_{DES,i}$ can be found in Section 8.4.

❖ Aerodynamic configurations

The ARPM defines five different flight segments and four associated threshold altitudes, $H_{max,i}$ (see Section 8.2), and proposes in the ACM file an aerodynamic configuration to be used for each segment. Table 2 sums up the conditions of application of each flight segment.

Table 2 – ARPM aerodynamic configurations

Id	Segment	Conditions of application
TO	Take-off	in climb up to $H_{max,TO}$ AGL
IC	Initial climb	in climb between $H_{max,TO}$ and $H_{max,IC}$ AGL
CR	Cruise	in climb above $H_{max,IC}$ AGL, in descent above $H_{max,AP}$ AGL, in descent below $H_{max,AP}$ AGL when $V \geq V_{min,CR} + 10$ kt
AP	Approach	in descent between $H_{max,AP}$ AGL and $H_{max,LD}$ AGL when $V < V_{min,CR} + 10$ kt, in descent below $H_{max,LD}$ AGL when $V_{min,CR} + 10\text{ kt} > V \geq V_{min,AP} + 10$ kt
LD	Landing	in descent below $H_{max,LD}$ AGL when $V < V_{min,AP} + 10$ kt

Figure 37 illustrates how both altitude and speed conditions are combined to determine the aerodynamic configuration during the descent.

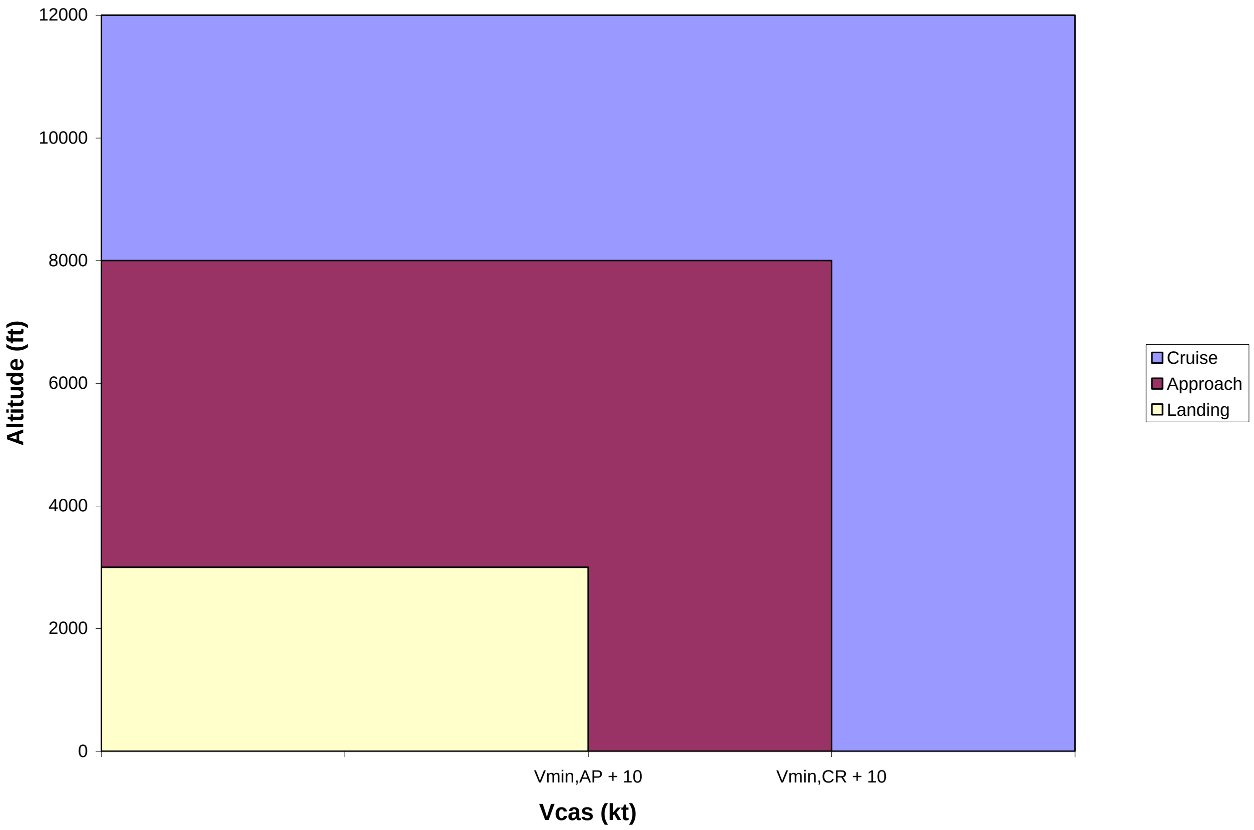

Figure 37 - Determination of aerodynamic configuration in descent

The threshold altitudes are expressed in terms of geopotential pressure altitude. However, when aircraft operations close to the ground are considered, one has to account for airport/runway elevation (Measured from MSL). The pressure altitude thresholds provided above correspond to geopotential pressure altitude Above Ground Level (AGL).

The threshold speeds are based on the operational minimum speed V_{min}, determined for each aerodynamic configuration using the formulas from the next section “Operational envelope limitations”. Please note that the operational minimum speed in the cruise configuration $V_{min,CR}$ is computed using the clean $C_{L,max}$ model when available, and as such may vary with the aircraft altitude.

❖ Operational envelope limitations

The limitations described in section 6 correspond to aircraft design or certification limits that shall never be exceeded. Actual operations of an aircraft, however, are usually restricted to a smaller flight envelope to provide adequate margin in case of turbulence or maneuver. While such safety margins may vary according to many factors (e.g. regulations, phase of flight, airline), the BADA airline procedure model considers the following margins:

- Stall margin

  The minimum operational speed in non-clean configurations, and clean configuration when no clean $C_{L,max}$ model is provided, is defined by:

$$V_{min} = \begin{cases} C_{V\,min,TO} \cdot V_S & \text{in the take-off phase (TO)} \\ C_{V\,min} \cdot V_S & \text{in all other phases (IC, CR, AP, LD)} \end{cases} \qquad \text{(B.5-1)}$$

  Where :

  $C_{Vmin,TO}$ and C_{Vmin} are minimum speed coefficients [-], see section 8.3

  V_S is the stall speed expressed in CAS [kt], determined by converting the minimum Mach number from section 6.3.3 into CAS (see section 2.2.2)

- Buffet margin

  Operational limitations computed using the clean $C_{L,max}$ model (e.g. minimum and maximum speeds, maximum altitude) are determined using a 1.2g buffet margin, i.e. assuming a load factor n = 1.2 (see section 6.3.4).

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