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Development of a mathematical model for the Bs Prime

Relatori:
pof.ssa Manuela Battipede

Candidati:
Diego Orio 275698

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Nomenclature

Abbreviations

Symbol	Description	Units
$c.g.$	center of gravity	
CAS	calibrated airspeed	$\mathrm{m\,s^{-1}}$
EAS	equivalent airspeed	$\mathrm{m\,s^{-1}}$
IAS	indicated airspeed	$\mathrm{m\,s^{-1}}$
mac	mean aerodynamic chord	m
MAP	manifold pressure	”Hg
MTO	take-off weight	kg
$MTOW$	maximum take-off weight	kg
ROC	rate of climb	$\mathrm{m\,s^{-1}}$
TAS	true airspeed	$\mathrm{m\,s^{-1}}$

Greek Symbols

Symbol	Description	Units
α	angle of attack	rad
β	sideslip angle	rad
χ	azimuth angle	rad
δ_a	deflection of ailerons	rad
δ_e	deflection of elevator	rad
δ_f	deflection of flaps	rad
δ_r	deflection of rudder	rad
η	propeller efficiency	–
Γ	angle between X_{body}-axis and the ground	rad
Γ	dihedral angle	deg

γ	flight-path angle	rad
γ	advancement ratio	–
λ	taper ratio	–
λ_a	longitudinal plan eigenvector	–
λ_b	lateral-directional plan eigenvector	–
λ_i	eigenvalue	–
μ	dynamic viscosity	$\mathrm{kg\,m^{-1}\,s^{-1}}$
ω_e	propeller speed	$\mathrm{rad\,s^{-1}}$
ω_n	natural frequency of damped system	$\mathrm{rad\,s^{-1}}$
Φ	bank angle	rad
ϕ	roll angle	rad
Π	engine power	W
Ψ	it takes into account the change in altitude	–
ψ	yaw angle	rad
ρ	density	$\mathrm{kg\,m^{-3}}$
θ	pitch angle	rad
ξ	damping	–
ξ	throttle percentage	%

Roman Symbols

Symbol	**Description**	**Units**
$\bar{c}$	mean aerodynamic chord	m
C	torque	N m
$C_p,\ C_t$	respectively power coefficient and thrust coefficient	–
D	propeller diameter	m
Fc	fuel consumption	$\mathrm{kg\,s^{-1}}$
fc	fuel consumption	$\mathrm{L\,s^{-1}}$
T	thrust	N
$(XYZ)_{datum}$	cartesian axes centered in the datum	
$(XYZ)_{body}$	cartesian axes centered in the aricraft center of gravity	
A	state matrix of the linearized system	

A_a	longitudinal plan state matrix of the linearized system	
A_b	lateral-directional plan state matrix of the linearized system	
b	wing span	m
C_l, C_m, C_n	respectively non-dimensional aerodynamic moments: rolling moment, pitching moment and yawing moment	−
C_x, C_y, C_z	respectively non-dimensional aerodynamic forces along: X_{body}, Y_{body} and Z_{body} axes	−
$F_{x_a}, F_{y_a}, F_{z_a}$	aerodynamic forces respectively along: X_{body}, Y_{body} and Z_{body} axes	N
$F_{x_p}, F_{y_p}, F_{z_p}$	propeller forces respectively along: X_{body}, Y_{body} and Z_{body} axes	N
H	altitude	m
h	height of a generic geometric figure	m
$I_{xx_i}, I_{yy_i}, I_{zz_i}$	the i-th element moment of inertia respectively along:X_{body} ,Y_{body} and Z_{body} axes	$\mathrm{kg\,m^2}$
$I_{xx_{tot}}$	aircraft moment of inertia along X_{body}-axis	$\mathrm{kg\,m^2}$
$I_{yy_{tot}}$	aircraft moment of inertia along Y_{body}-axis	$\mathrm{kg\,m^2}$
$I_{zz_{tot}}$	aircraft moment of inertia along Z_{body}-axis	$\mathrm{kg\,m^2}$
$J_{xy_i}, J_{xz_i}, J_{yz_i}$	i-th element product of inertia respectively in the planes X_{body} Y_{body}, X_{body} Z_{body} and Y_{body} Z_{body}	$\mathrm{kg\,m^2}$
$J_{xy_{tot}}$	aircraft product of inertia in X_{body} Y_{body}-plane	$\mathrm{kg\,m^2}$
$J_{xz_{tot}}$	aircraft product of inertia in X_{body} Z_{body}-plane	$\mathrm{kg\,m^2}$
$J_{yz_{tot}}$	aircraft product of inertia in Y_{body} Z_{body}-plane	$\mathrm{kg\,m^2}$
L_a, M_a, N_a	aerodynamic moments respectively along: X_{body}, Y_{body} and Z_{body} axes	N
L_p, M_p, N_p	propeller moments respectively along: X_{body}, Y_{body} and Z_{body} axes	N m
n	engine speed	RPM
p, q, r	respectively angular rate of: roll, pitch, yaw	$\mathrm{rad\,s^{-1}}$
ps	ambient or free-stream pressure	Pa
pz	manifold pressure	”Hg
S	wing surface area	$\mathrm{m^2}$
S_i	area of a generic geometric figure	$\mathrm{m^2}$

T	temperature	°C
u, v, w	speed components respectively along: X_{body}, Y_{body} and Z_{body} axes	m
V	true airspeed	$\mathrm{m\,s^{-1}}$
X	state vector	
x_e, y_e	respectively: X-coordinate and Y-coordinate in Earth-fixed reference frame	m
$x_{c.g_i}, y_{c.g_i}, z_{c.g_i}$	i-th element center of gravity position with respect to aircraft c.g., respectively along the: X_{body} ,Y_{body} and Z_{body} axes	m
$x_{c.g\%}$	position of the center of gravity along the X_{body}-axis as a percentage of the mean aerodynamic chord	%
$x_{c.g}, y_{c.g}, z_{c.g}$	aircraft center of gravity position with respect to the datum, respectively along the X_{datum}, Y_{datum} and Z_{datum} axes	m
$x_{d_i}, y_{d_i}, z_{d_i}$	i-th element center of gravity coordinates respectively along the: X_{datum}, Y_{datum} and Z_{datum} axes	m

Abstract

This master's thesis work was carried out abroad at the EURO FLIGHT TEST company, based in Winningen, Germany. The main purpose is to create a mathematical model in MATLAB / Simulink that allows to simulate the behavior of the BS PRIME aircraft. The company, which has recently chartered a BS PRIME aircraft, has provided its willingness to support the project, both during the development phase and during the validation phase. It was in this second phase that the possibility of performing flight tests with a specialized pilot was of fundamental importance. This thesis focuses on model development, from geometry to aerodynamic derivatives. It was necessary to separate the modeling discussion from the validation one for bureaucratic reasons. It is noted that in order to have a global understanding it is necessary to consider the two parts as a single work. **The work and all the necessary activities, net of the contribution provided by EURO FLIGHT TEST, were carried out by both candidates: Andrea Corino and Diego Orio**.

Chapter 1

Blackshape Prime

1.1 Bs Prime aircraft

The Blackshape Prime is an aircraft developed by the Italian company Blackshape, based in Monopoli. It is an ultralight, single-engine, monoplane and two-seater aircraft. The structure is entirely in carbon fiber. It is characterized by a high level of safety thanks to the installation of a ballistic parachute, an advanced diagnostic module, stall entry warning and four-point safety belts. The engine installed is a 4-cylinder Rotax 912 ULS, delivering 100 HP (74.6 kW). The BS Prime under consideration is characterized by a MVT-33-1A variable pitch hydraulically controlled two-blade propeller. In 2013 it won the Flieger Magazine Award for best ultralight aircraft. The following table shows the main features.

	Symbol	Value	Unit
Total Length	L_{tot}	7.178	[m]
Total Height	H_{tot}	2.41	[m]
EmptyWeight	EW	390	[kg]
Maximum Take-OffWeight	MTOW	600	[kg]
Maximum Engine Power	Π_{max}	100	[HP]
Propeller Radius	d	1.75	[m]
Wing Surface Area	S	9.51	[m^2]
Wing Span	b	7.94	[m]
Never Exceed Speed	V_{NE}	305	[km/h]
Maximum Structural Cruising Speed	V_{NO}	250	[km/h]

Table 1.1: BS Prime General Characteristics

Chapter 2

Digital DATCOM

2.1 Introduction

Digital DATCOM is a program developed by the United States Air Force that allows you to calculate the characteristics of static stability, dynamic stability, and control, using suitable numerical methods. The program requests as input a file containing a long series of geometric data of the aircraft, and returns as output the non-dimensional derivatives relating to the specified flight conditions.

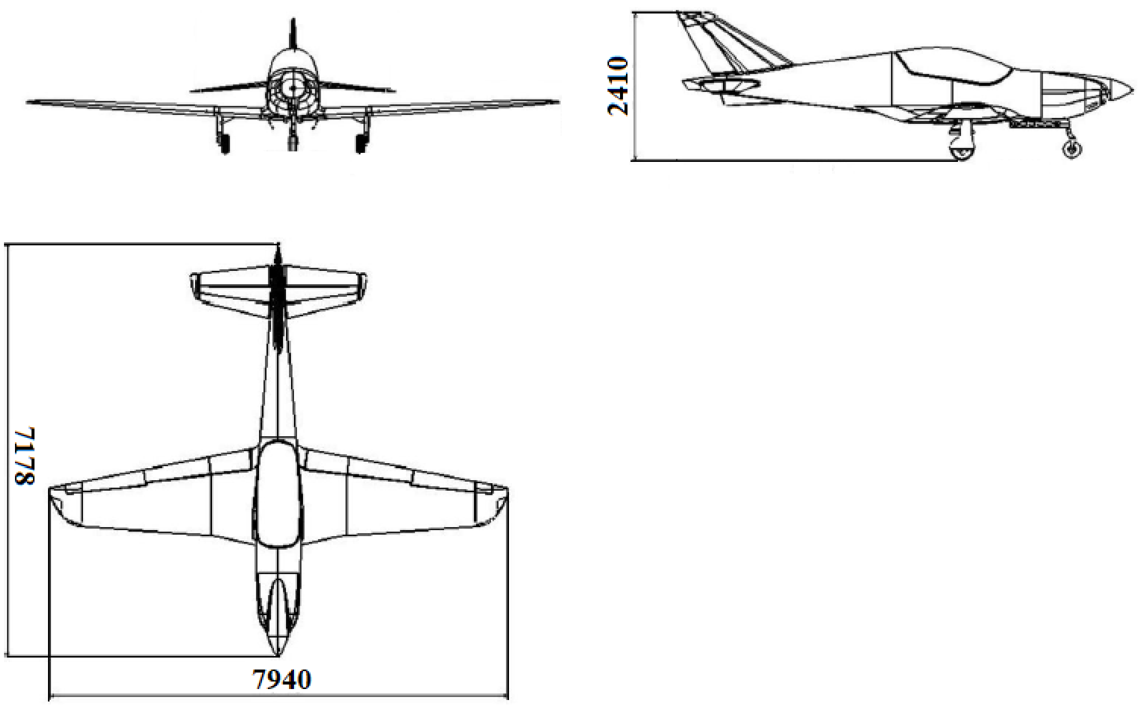

Figure 2.1.1: Three view drawing (all dimensions in mm)

This program was used instead of creating a CAD model as the time required was much shorter, without affecting the reliability of the results obtained. Also, having no data source other than the flight manual, making a sufficiently faithful CAD model would have been unnecessarily laborious. The drawings in the flight manual were used to obtain the necessary geometric data, [1, p. 21]. Almost all of the values entered in the input file were obtained by measuring the quantities directly from the drawings with a caliper (or, if necessary, a protractor) and multiplying the values obtained by the appropriate scale ratios. Note that it was decided to build the aerodynamic model centered in the datum (see section 2.5), as the position of the center of gravity is variable. The aerodynamic derivatives obtained from the output file do not take into account the effect of the flow produced by the propeller.

2.2 Reference systems

It should be noted that the reference system used by DATCOM for the construction of the geometric model originates in the tip of the propeller of the aircraft, with axis X pointing towards the tail, axis Z pointing upwards, axis Y facing the driver's right, as shown in Figure 2.2.1. Conversely, all outputs are provided with respect to a different reference system, which

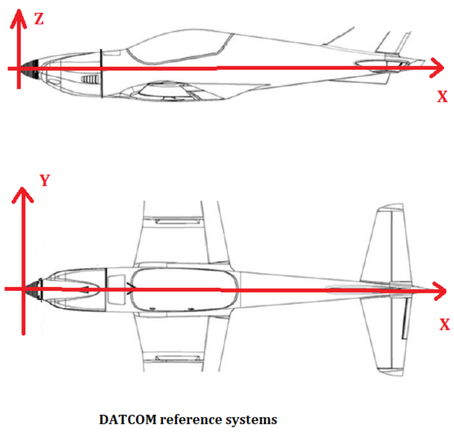

Figure 2.2.1: Datcom reference systems

in this discussion will be defined as the datum reference system. It is centered in the datum, with the X_{datum} axis facing the nose of the aircraft, the Z_{datum} axis facing down, and the Y_{datum} axis as in Figure 2.2.2 on the left. The Body reference system is also defined as a set of three

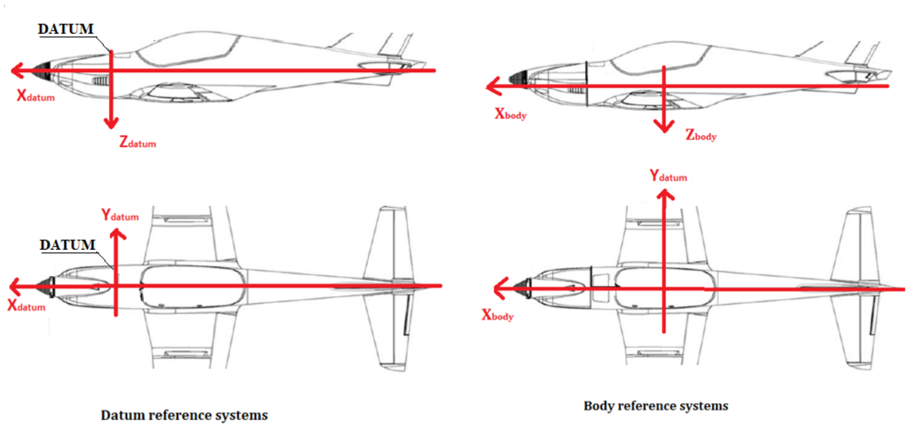

Figure 2.2.2: Datum and Body reference systems

arbitrary Cartesian axes centered in the center of gravity of the aircraft. In this discussion they are defined parallel to those of the datum, but centered in the center of gravity as in Figure 2.2.2 on the right.

2.3 DATCOM modeling: input file

Flight Condition

The weight corresponds to the MTOW in *lb*. The altitude in ft is chosen arbitrarily, and the Mach is calculated assuming the maximum cruising speed at the indicated altitude. The choice of these last two values has an absolutely negligible influence on the results, both because they are dimensionless and because the flight regime is abundantly subsonic and the effects of compressibility are also negligible ($M < 0.3$). In fact, the program takes them into account only starting from Mach values higher than 0.6. The altitude was chosen as an intermediate value between sea level and the maximum altitude [1, p. 100]. The values of α listed are those for which the output coefficients will be calculated. They are limited between $-8°$ and $20°$ (see chapter 5).

```
**********************
*  Flight Conditions *
**********************
*   WT       Vehicle Weight
*   LOOP     Program Looping Control
*               1 = vary altitude and mach together, default)
*               2 = vary Mach, at fixed altitude
*               3 = vary altitude, at fixed Mach
*   NMACH    Number of Mach numbers or velocities to be run, max of 20
*            Note: This parameter, along with NALT, may affect the
*            proper setting of the LOOP control parameter.
*   MACH     Array(20) Values of freestream Mach number
*   VINF     Array(20) Values of freestream speed (unit: l/t)
*   NALPHA   Number of angles of attack to be run, max of 20
*   ALSCHD   Array(20) Values of angles of attack, in ascending order
*   RNNUB    Array(20) Reynolds number per unit length
*            Freestream Reynolds numbers. Each array element must
*            correspond to the respective Mach number/freestream
*            speed input, use LOOP=1.0
*   NALT     Number of atmospheric conditions to be run, max of 20
*            input as either altitude or pressure and temperature
*            Note: This parameter, along with NMACH, may affect the
*            proper setting of the LOOP control parameter.
*   ALT      Array(20) Values of geometric altitude
*            Number of altitude and values. Note, Atmospheric conditions
*            are input either as altitude or pressure and temperature. (MAX
   20)
*   PINF     Array(20) Values of freestream Static Pressure
*   TINF     Array(20) Values of freestream Temperature
*   HYPERS   =.true.  Hypersonic analysis at all Mach numbers > 1.4
*   STMACH   Upper limit of Mach numbers for subsonic analysis
*            (0.6<STMACH<0.99), Default to 0.6 if not input.
*   TSMACH   Lower limit of Mach number for Supersonic analysis
*            (1.01<=TSMACH<=1.4)  Default to 1.4
*   TR       Drag due to lift transition flag, for regression analysis
*            of wing-body configuration.
*            = 0.0 for no transition (default)
*            = 1.0 for transition strips or full scale flight
*   GAMMA    Flight path angle WT=1322.77,

    $FLTCON WT = 1322.77, LOOP=2.0, RNNUB = 1000000.0,
```

```
NMACH=1.0, MACH(1)=0.206,
NALT=1.0, ALT(1)=4000.0,
NALPHA=16.0,
ALSCHD(1)= -8.0, -6.0, -4.0, -2.0,
0.0, 1.0, 2.0, 4.0, 6.0, 8.0, 10.0, 12.0, 14.0,16.0,17.0,20.0,
STMACH=0.6, TSMACH=1.4, TR=1.0$
```

Reference parameters

The data reported in this section are taken from the flight manual, with the exception of the wingspan value, which is about 4 cm lower than the real value (an error of about 0.5 %). This is due to the impossibility of modeling the curved section corresponding to the wing tip in DATCOM, which made it necessary to slightly modify the parameters of the wing planform (for more details see the section on the wing planform)

```
*************************
*  Reference Parameters *   pg 29
*************************
*   SREF    Reference area value of theoretical wing area used by program
*           if not input
*   CBARR   Longitudinal reference length value of theoritcal wing
*           Mean Aerodynamic Chord used by program if not input
*   BLREF   Lateral reference length value of wing span used by program
*   ROUGFC  Surface roughness factor, equivalent sand roughness, default
*           to 0.16e-3 inches (Natural sheet metal)
*           0.02/0.08E-3 - Polished metal or wood
*           0.16E-3  - Natural sheet metal
*           0.25E-3  - Smooth matte paint, carefully applied
*           0.40E-3  - Standard camouflage paint, average application

    $OPTINS SREF=102.36, CBARR=4.11, BLREF=25.4531, ROUGFC=0.16E-3$
```

Synthesis Parameters

The coordinates of the center of gravity actually correspond to those of the datum. It follows that all the results obtained, in particular the moment coefficients, will be intended with respect to the datum. This is because the actual position of the center of gravity is variable, it depends on the load condition and fuel consumption. Therefore it was preferred to refer everything with respect to the datum and subsequently, within the mathematical model, to transport forces and moments with respect to the position of the center of gravity. The positions of the "apex" of wing, vertical and horizontal tail, and ventral fin were measured assuming the shape that the surfaces have inside the fuselage, as shown in the drawings in the following sections.

```
**************************************
* Group II     Synthesis Parameters  *  pg 33
**************************************
*   XCG     Longitudinal location of cg (moment ref. center)
*   ZCG     Vertical location of CG relative to reference plane
*   XW      Longitudinal location of theoretical wing apex (where
*           leading edge would intersect long axis)
*   ZW      Vertical location of theoretical wing apex relative to
*           reference plane
*   ALIW    Wing root chord incident angle measured from reference plane
*   XH      Longitudinal location of theoretical horizontal tail apex.
*           If HINAX is input, XH and ZH are evaluated at zero incidence.
*   ZH      Vertical location of theoretical horizontal tail apex
```

```
*            relative to reference plane. If HINAX is input, XH and ZH
*            are evaluated at zero incidence.
*   ALIH     Horizontal tail root chord incidence angle measured from
*            reference plane
*   XV       Longitudinal location of theoretical vertical tail apex
*   XVF      Longitudinal location of theoretical ventral fin apex
*   ZV       Vertical location of theoretical vertical tail apex
*            This kinda makes sense only for twin tails that are canted
*   ZVF      Vertical location of theoretical ventral fin apex
*            This kinda makes sense only for twin tails that are canted
*   SCALE    Vehicle scale factor (multiplier to input dimensions)
*   VERTUP   Vertical panel above reference plane (default=true)
*   HINAX    Longitudinal location of horizontal tail hinge axis.
*            Required only for all-moveable horizontal tail trim option.
*XCG=4.7458, ZCG=-0.0,  datum coordinates

    $SYNTHS XCG=4.7458, ZCG=-0.0,
     XW=5.9531,  ZW=-1.148,    ALIW=0.0,
     XH=19.0507, ZH=0.1916,    ALIH=-2.0,
     XV=17.313,  ZV=0.0,
     XVF=17.313, ZVF=-0.656,
     SCALE=1.0, VERTUP=.TRUE.$
```

Body Configuration Parameters

The fuselage was modeled considering 9 different sections, specifying for each its x coordinate (with respect to the construction reference system), its width, and the corresponding values of ZU and ZL, the meaning of which is illustrated in the code reported. The remaining parameters are not used in the subsonic field, and can therefore be ignored. The selected sections are shown in Figure 2.3.1.

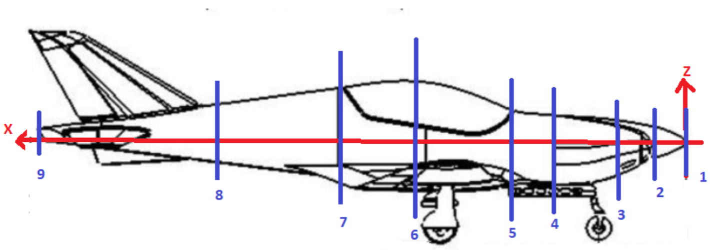

Figure 2.3.1: Body section position

```
**********************************
*  Body Configuration Parameters *  pg 36
**********************************
*  Here is an error message output by DIGDAT concerning body geometry:
*   IN NAMELIST BODY, ONLY THE FOLLOWING COMBINATIONS OF VARIABLES CAN BE
   USED
*   FOR A CIRCULAR BODY, SPECIFY X AND R OR X AND S
*   FOR AN ELLIPTICAL BODY, SPECIFY X AND R OR X AND S, AND THE VARIABLE
   ELLIP
*   FOR OTHER BODY SHAPES X, R, S, AND P MUST ALL BE SPECIFIED
*
*   NX       Number of longitudinal body stations at which data is
*            specified, max of 20
*   X        Array(20) Longitudinal distance measured from arbitray location
*   S        Array(20) Cross sectional area at station. See note above.
```

```
*    P         Array(20) Periphery at station Xi. See note above.
*    R         Array(20) Planform half width at station Xi. See note above.
*    ZU        Array(20) Z-coordinate at upper body surface at station Xi
*              (positive when above centerline)
*              [Only required for subsonic asymmetric bodies]
*    ZL        Array(20) Z-coordinate at lower body surface at station Xi
*              (negative when below centerline)
*              [Only required for subsonic asymmetric bodies]
*    BNOSE     Nosecone type  1.0 = conical (rounded), 2.0 = ogive (sharp point)
*              [Not required in subsonic speed regime]
*    BTAIL     Tailcone type  1.0 = conical, 2.0 = ogive, omit for lbt = 0
*              [Not required in subsonic speed regime]
*    BLN       Length of body nose
*              Not required in subsonic speed regime
*    BLA       Length of cylindrical afterbody segment, =0.0 for nose alone
*              or nose-tail configuration
*              Not required in subsonic speed regime
*    DS        Nose bluntness diameter, zero for sharp nosebodies
*              [Hypersonic speed regime only]
*    ITYPE     1.0 = straight wing, no area rule
*              2.0 = swept wing, no area rule (default)
*              3.0 = swept wing, area rule
*    METHOD    1.0 = Use existing methods (default)
*              2.0 = Use Jorgensen method

     $BODY NX=9.0,
      X(1)=0.0,1.1349,2.4131,4.7458,6.1415,9.5479,12.3490,16.5023,23.2829,
      R(1)=0.0,0.5735,1.1541,1.2304,1.3343,1.3174,1.0404,0.7102,0.3001,
      ZU(1)=0.0,0.5119,0.7985,0.9417,1.2491,2.2922,2.0025,1.3217,0.4829,
      ZL(1)=0.0,-0.5119,-1.1365,-1.6050,-1.7852,-2.0153,-1.6838,-0.8854,0.0,
      BNOSE=1.0, BLN=4.7458,
      BTAIL=1.0, BLA=0.0,
      ITYPE=1.0, METHOD=1.0$
```

Wing planform variables

As mentioned above, DATCOM does not allow to model the curved leading edge that characterizes the wing tip of the Blackshape Prime. The best that can be done is to model the wing as consisting of two trapezoids. This made it necessary to make an approximation of the wing planform, which was then remodeled as shown in Figure 2.3.3. This led to a surface loss of 0.066 m^2, thus committing an error of 0.7% on the surface.

```
**********************************
*         Wing planform variables   pg 37-38
**********************************
*    CHRDR     Chord root
*    CHRDBP    Chord at breakpoint. Not required for straight
*              tapered planform.
*    CHRDTP    Tip chord
*    SSPN      Semi-span theoretical panel from theoretical root chord
*    SSPNE     Semi-span exposed panel, See diagram on pg 37.
*    SSPNOP    Semi-span outboard panel. Not required for straight
*              tapered planform.
*    SAVSI     Inboard panel sweep angle
*    SAVSO     Outboard panel sweep angle
*    CHSTAT    Reference chord station for inboard and outboard panel
*              sweep angles, fraction of chord
*    TWISTA    Twist angle, negative leading edge rotated down (from
*              exposed root to tip)
*    SSPNDD    Semi-span of outboard panel with dihedral
```

```
*    DHDADI   Dihedral angle of inboard panel
*    DHDADO   Dihedral angle of outboard panel. If DHDADI=DHDADO only
              input DHDADI
*    TYPE     1.0 - Straight tapered planform
              2.0 - Double delta planform (aspect ratio <= 3)
              3.0 - Cranked planform (aspect ratio > 3)

   $WGPLNF CHRDR=6.2283,          CHRDTP=2.1093, CHRDBP=4.5,
    SSPN=12.956,   SSPNE=11.4824, SSPNOP=9.0891,
    SAVSI=10.0, SAVSO=3.0,
    CHSTAT=0.25, TWISTA=-1.0,
    DHDADI=4.0,
    TYPE=1.0$
```

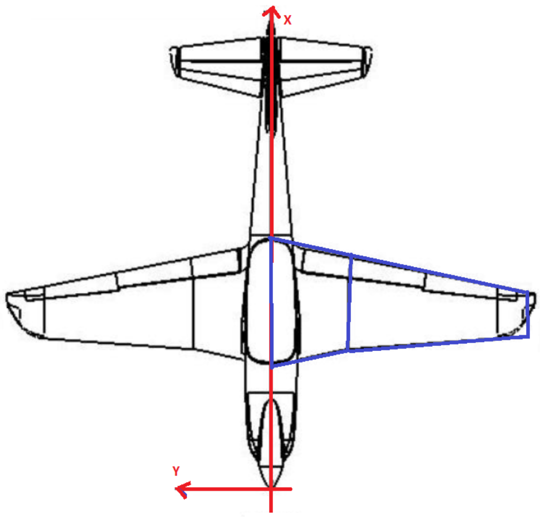

Figure 2.3.2: Wing planform

Wing sectional characteristics parameters

The sectional characteristics of the wing can be entered as input on DATCOM in two different ways: by entering a list of parameters listed in the guide, or by directly entering a NACA profile. In this discussion, the second solution was chosen, using a NACA 651-212 profile. This is quite an important approxima-

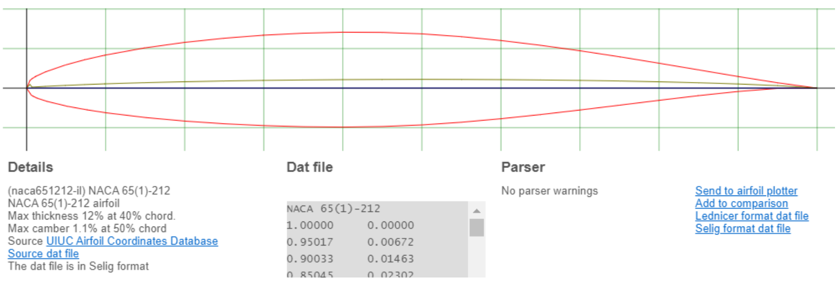

Figure 2.3.3: Wing airfoil: NACA 651-212

tion, as the wing has different profiles along its span, and nothing guarantees that NACA profiles were actually used for the real aircraft. However, DATCOM does not allow to insert any more profiles for different sections, so this approximation is inevitable. In order to choose the profile, the data relative to the DATCOM modeling of the BS115 were used as a starting point [6, p. 81]: thickness 8 %, $C_{L_{max}} = 1.6$,

C_{m_0} = - 0.051. These data, entered in the airfoiltools search engine, initially led to the profile that comes closest to them: NACA 2408. Following tests performed on the trim conditions, it was necessary to correct this choice, and consequently the aerodynamic derivatives , to ensure that the test results coincide with those reported in the flight manual. With a trial and error approach, we then arrived, as mentioned, at the NACA 651-212 profile. In any case, as it will be seen later in the thesis "Validation of mathematical model for the Bs Prime", some coefficients have been further modified, to make the aerodynamic model sufficiently faithful to the real aircraft. Consequently, the choice of the profile to be inserted on DATCOM is indicative, in the absence of more precise information from the flight manual.

```
**********************************************
*  Wing Sectional Characteristics Parameters *  pg 39-40
**********************************************
*  The section aerodynamic characteristics for these surfaces are
*  input using either the sectional characteristics namelists WGSCHR,
*  HTSCHR, VTSCHR and VFSCHR and/or the NACA control cards. Airfoil
*  characteristics are assummed constant for each panel of the planform.
*
*  To avoid having to input all the airfoil sectional characteristics,
*  you can specify the NACA airfoil designation. Starts in Column 1.
*
* NACA x y zzzzzz
*
*   where:
*      column 1-4   NACA
*             5     any deliminator
*             6     W, H, V, or F  Planform for which the airfoil
*                                  designation applies:  Wing, Horizontal
*                                  tail, Vertical tail, or Ventral fin.
*             7     any deliminator
*             8     1,4,5,6,S      Type of airfoil section: 1-series,
*                                  4-digit, 5-digit, 6-series, or Supersonic
*             9     any deliminator
*            10-80  Designation, columns are free format, blanks are ignored

   NACA-W-6-651-212

   SAVE
```

Horizontal tail sectional characteristics and planform variables

The measurements relating to this surface were measured with a caliper and a protractor after having approximated the shape to two trapezoids as in Figure 2.3.4. NACA 63-010 was chosen as the profile, which is the same used for the analogous surface of the Blackshape 115 [6, p. 81].

```
*************************************************
*       Horizontal Tail Sectional Characteristics   pg 39-40
*************************************************
NACA-H-5-63-010

*********************************************
*         Horizontal Tail planform variables    pg 37-38
*********************************************
*   CHRDTP  Tip chord
*   SSPNOP  Semi-span outboard panel. Not required for straight
*           tapered planform.
*   SSPNE   Semi-span exposed panel
*   SSPN    Semi-span theoretical panel from theoretical root chord
*   CHRDBP  Chord at breakpoint
```

```
*    CHRDR    Chord root
*    SAVSI    Inboard panel sweep angle
*    CHSTAT   Reference chord station for inboard and outboard panel
*             sweep angles, fraction of chord
*    TWISTA   Twist angle, negative leading edge rotated down (from
*             exposed root to tip)
*    SSPNDD   Semi-span of outboard panel with dihedral
*    DHDADI   Dihedral angle of inboard panel
*    DHDADO   Dihedral angle of outboard panel. If DHDADI=DHDADO only
*             input DHDADI
*    TYPE     1.0 - Straight tapered planform
*             2.0 - Double delta planform (aspect ratio <= 3)
*             3.0 - Cranked planform (aspect ratio > 3)
*    SHB      Portion of fuselage side area that lies between Mach lines
*             originating from leading and trailing edges of horizontal
*             tail exposed root chord (array 20).
*             Only required for supersonic and hypersonic speed regimes.
*    SEXT      Portion of extended fueslage side area that lies between
*              Mach lines originating from leading and trailing edges of
*              horizontal tail exposed root chord (array 20)
*             Only required for supersonic and hypersonic speed regimes.
*    RLPH     Longitudinal distance between CG and centroid of Sh(B)
*             positive aft of CG
*             Only required for supersonic and hypersonic speed regimes.

  $HTPLNF CHRDR=3.178, CHRDTP=1.411,
   SSPN=4.95, SSPNE=4.552,
   SAVSI=10.0,
   CHSTAT=0.25, TWISTA=-1.0,
   DHDADI=-3.0,
   TYPE=1.0$
```

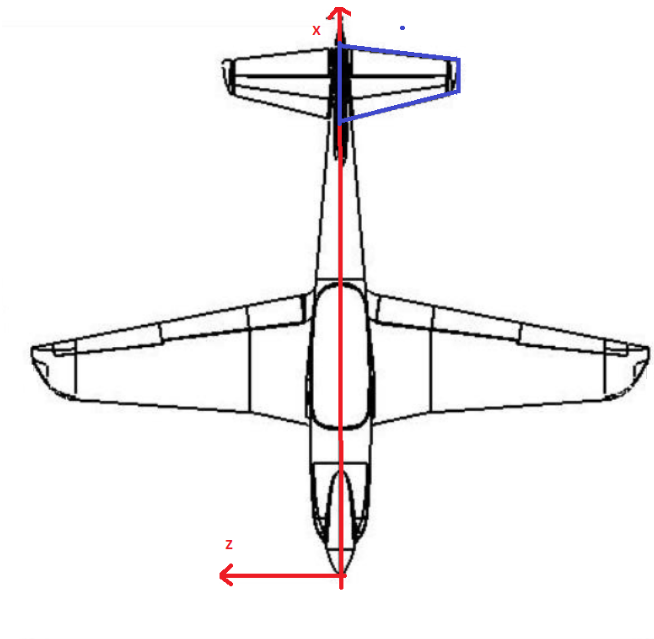

Figure 2.3.4: Horrizontal Tail planfor

Vertical Tail planform

The measurements relating to this surface were measured with a caliper and a protractor after having approximated the shape to a trapezoid as in Figure 2.3.4. NACA 63-010 was chosen as the profile, which is the same used for the analogous surface of the Blackshape 115 [6, p. 81].

```
*******************************************
*          Vertical Tail planform variables    pg 37-38
*******************************************
*    CHRDTP  Tip chord
*    SSPNOP  Semi-span outboard panel
*    SSPNE   Semi-span exposed panel
*    SSPN    Semi-span theoretical panel from theoretical root chord
*    CHRDBP  Chord at breakpoint
*    CHRDR   Chord root
*    SAVSI   Inboard panel sweep angle
*    SAVSO   Outboard panel sweep angle
*    CHSTAT  Reference chord station for inboard and outboard panel
*            sweep angles, fraction of chord
*    TYPE    1.0 - Straight tapered planform
*            2.0 - Double delta planform (aspect ratio <= 3)
*            3.0 - Cranked planform (aspect ratio > 3)
*    SVWB    Portion of exposed vertical panel area that lies between
*            Mach lines emanating from leading and trailing edges of
*            wing exposed root chord (array 20)
*            Only required for supersonic and hypersonic speed regimes.
*    SVB     Area of exposed vertical panel not influenced by wing or
*            horizontal tail (array 20)
*            Only required for supersonic and hypersonic speed regimes.
*    SVHB    Portion of exposed vertical panel area that lies between Mach
*            lines emanating from leading and and trailing edges of
*            horizontal tail exposed root chord (array 20)
*            Only required for supersonic and hypersonic speed regimes.

   $VTPLNF CHRDTP=1.672, SSPNE=3.21,  SSPN=3.871, CHRDR=5.42,
    SAVSI=55.0,  CHSTAT=0.25, TYPE=1.0$
```

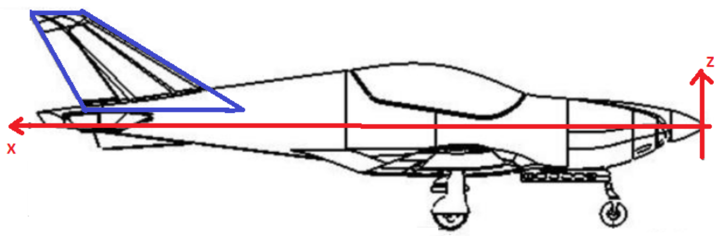

Figure 2.3.5: Vertical Tail planform

Vertical Fin planform

The measurements relating to this surface were measured with a caliper and a protractor after having approximated the shape to a triangle as in Figure 2.3.4. NACA 63-010 is the profile, which is the same used for the analogous surface of the Blackshape 115 [6, p. 81]

```
*******************************************
*          Vertical Fin planform variables     pg 37-38
```

```
*******************************************
*   CHRDR   Chord root
*   CHRDBP  Chord at breakpoint
*   CHRDTP  Tip chord
*   SSPNOP  Semi-span outboard panel
*   SSPNE   Semi-span exposed panel
*   SSPN    Semi-span theoretical panel from theoretical root chord
*   SAVSI   Inboard panel sweep angle
*   CHSTAT  Reference chord station for inboard and outboard panel
*           sweep angles, fraction of chord
*   DHDADO  Dihedral angle of outboard panel. If DHDADI=DHDADO only
*           input DHDADI
*   DHDADO  Dihedral angle of outboard panel. If DHDADI=DHDADO only
*           input DHDADI
*   TYPE    1.0 - Straight tapered planform
*           2.0 - Double delta planform (aspect ratio <= 3)
*           3.0 - Cranked planform (aspect ratio > 3)
*   SVWB    Portion of exposed vertical panel area that lies between
*           Mach lines emanating from leading and trailing edges of
*           wing exposed root chord (array 20)
*           Only required for supersonic and hypersonic speed regimes.
*   SVB     Area of exposed vertical panel not influenced by wing or
*           horizontal tail (array 20)
*           Only required for supersonic and hypersonic speed regimes.
*   SVHB    Portion of exposed vertical panel area that lies between Mach
*           lines emanating from leading and and trailing edges of
*           horizontal tail exposed root chord (array 20)
*           Only required for supersonic and hypersonic speed regimes.

    $VFPLNF CHRDR=4.075, CHRDTP=4.075, CHSTAT=0.5, DHDADO=0.0,
     SAVSI=-26.0, SSPN=0.73472,  SSPNE=0.36736, TYPE=1.0$

***********************************************
*       Ventral Fin Sectional Characteristics  pg 39-40
***********************************************
    NACA-F-5-63-010
```

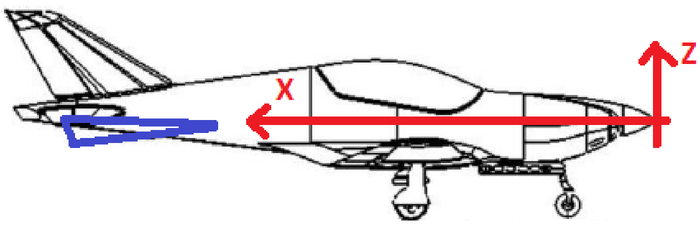

Figure 2.3.6: Ventral Fin planform

Flap Deflection parameter

The simplest shape was chosen for this control surface: plain. The deflections are the same as reported in the flight manual [1, p. 91].

```
*******************************************
*    Symetrical Flap Deflection parameters
*******************************************
*
*   FTYPE   Flap type
```

```
*                1.0  Plain flaps
*                2.0  Single slotted flaps
*                3.0  Fowler flaps
*                4.0  Double slotted flaps
*                5.0  Split flaps
*                6.0  Leading edge flap
*                7.0  Leading edge slats
*                8.0  Krueger
*    NDELTA   Number of flap or slat deflection angles, max of 9
*
*    DELTA    Flap deflection angles measured streamwise
*             (NDELTA values in array)
*    PHETE    Tangent of airfoil traile edge angle based on ordinates at
*             90 and 99 percent chord
*    PHETEP   Tangent of airfoil trailing edge angle based on ordinates at
*             95 and 99 percent chord
*    CHRDFI   Flap chord at inboard end of flap, measured parallel to
*             longitudinal axis
*    CHRDFO   Flap chord at outboard end of flap, measured parallel to
*             longitudinal axis
*    SPANFI   Span location of inboard end of flap, measured perpendicular
*             to vertical plane of symmetry
*    SPANFO   Span location of outboard end of flap, measured perpendicular
*             to vertical plane of symmetry
*    CPRMEI   Total wing chord at inboard end of flap (translating devices
*             only) measured parallel to longitudinal axis
*             (NDELTA values in array)
*                Single-slotted, Fowler, Double-slotted, leading-edge
*                slats, Krueger flap, jet flap
*    CPRMEO   Total wing chord at outboard end of flap (translating devices
*             only) measured parallel to longitudinal axis
*             (NDELTA values in array)
*                Single-slotted, Fowler, Double-slotted, leading-edge
*                slats, Krueger flap, jet flap
*    CAPINS                           (double-slotted flaps only)
*    CAPOUT                           (double-slotted flaps only)
*    DOSDEF                           (double-slotted flaps only)
*    DOBCIN                           (double-slotted flaps only)
*    DOBCOT                           (double-slotted flaps only)
*    SCLD     Increment in section lift coefficient due to
*             deflecting flap to angle DELTA[i]      (optional)
*             (NDELTA values in array)
*    SCMD     Increment in section pitching moment coefficient due to
*             deflecting flap to angle DELTA[i]      (optional)
*             (NDELTA values in array)
*    CB       Average chord of the balance    (plain flaps only)
*    TC       Average thickness of the control at hinge line
*                                             (plain flaps only)
*    NTYPE    Type of nose
*                1.0  Round nose flap
*                2.0  Elliptic nose flap
*                3.0  Sharp nose flap
*    JETFLP   Type of flap
*                1.0  Pure jet flap
*                2.0  IBF
*                3.0  EBF
*    CMU      Two-dimensional jet efflux coefficient
*    DELJET   Jet deflection angle
*             (NDELTA values in array)
*    EFFJET   EBF Effective jet deflection angle
*             (NDELTA values in array)
```

```
$SYMFLP FTYPE=1.0,     NDELTA=3.0,
 DELTA(1)=0.0,10.0,30.0,
 PHETE=0.052, PHETEP=0.0391,
 CHRDFI=1.222,   CHRDFO=0.76,
 SPANFI=1.66,  SPANFO=7.7,
 NTYPE=1.0$
```

Assimmetrical Control Deflection parameters : Ailerons

The simplest shape was chosen for this control surface: plain. The deflections are the same as reported in [5, p. 6].

```
*************************************************************
*    Asymmetrical Control Deflection parameters : Ailerons
*************************************************************
*   STYPE   Type
*              1.0  Flap spoiler on wing
*              2.0  Plug spoiler on wing
*              3.0  Spoiler-slot-deflection on wing
*              4.0  Plain flap aileron
*              5.0  Differentially deflected all moveable horizontal tail
*   NDELTA  Number of control deflection angles, required for all controls,
*           max of 9
*   DELTAL  Defelction angle for left hand plain flap aileron or left
*           hand panel all moveable horizontal tail, measured in
*           vertical plane of symmetry
*   DELTAR  Defelction angle for right hand plain flap aileron or right
*           hand panel all moveable horizontal tail, measured in
*           vertical plane of symmetry
*   SPANFI  Span location of inboard end of flap or spoiler control,
*           measured perpendicular to vertical plane of symmetry
*   SPANFO  Span location of outboard end of flap or spoiler control,
*           measured perpendicular to vertical plane of symmetry
*   PHETE   Tangent of airfoil trailing edge angle based on ordinates
*           at x/c - 0.90 and 0.99
*   CHRDFI  Aileron chord at inboard end of plain flap aileron,
*           measured parallel to longitudinal axis
*   CHRDFO  Aileron chord at outboard end of plain flap aileron,
*           measured parallel to longitudinal axis
*   DELTAD  Projected height of deflector, spoiler-slot-deflector
*           control, fraction of chord
*   DELTAS  Projected height of spoiler, flap spoiler, plug spoiler and
*           spoiler-slot-deflector control; fraction of chord
*   XSOC    Distance from wing leading edge to spoiler lip measured
*           parallel to streamwise wng chord, flap and plug spoilers,
*           fraction of chord
*   XSPRME  Distance from wing leading edge to spoiler hinge line
*           measured parallel to streamwise chord, flap spoiler,
*           plug spoiler and spoiler-slot-deflector control, fraction
*           of chord
*   HSOC    Projected height of spoiler measured from and normal to
*           airfoil mean line, flap spoiler, plug spoiler and spoiler-
*           slot-reflector, fraction of chord

  $ASYFLP STYPE=4.0, NDELTA=7.0,
   DELTAL(1)=-28.0,-18.0,-8.0,0.0,3.0,13.0,23.0,
   DELTAR(1)=28.0,18.0,8.0,0.0,-3.0,-13.0,-23.0,
   SPANFI=7.7, SPANFO=12.217,
   PHETE=0.05228,
```

```
CHRDFI=0.94, CHRDFO=0.57$
```

Elevator Deflection parameters

The simplest shape was chosen for this control surface: plain. The deflections are the same as reported in [5, p. 6].

```
***********************************
*    Elevator Deflection parameters
***********************************
*   FTYPE   Flap type
*              1.0  Plain flaps
*              2.0  Single slotted flaps
*              3.0  Fowler flaps
*              4.0  Double slotted flaps
*              5.0  Split flaps
*              6.0  Leading edge flap
*              7.0  Leading edge slats
*              8.0  Krueger
*   NDELTA  Number of flap or slat deflection angles, max of 9
*   DELTA   Flap deflection angles measured streamwise
*           (NDELTA values in array)
*   PHETE   Tangent of airfoil trailine edge angle based on ordinates at
*           90 and 99 percent chord
*   PHETEP  Tangent of airfoil trailing edge angle based on ordinates at
*           95 and 99 percent chord
*   CHRDFI  Flap chord at inboard end of flap, measured parallel to
*           longitudinal axis
*   CHRDFO  Flap chord at outboard end of flap, measured parallel to
*           longitudinal axis
*   SPANFI  Span location of inboard end of flap, measured perpendicular
*           to vertical plane of symmetry
*   SPANFO  Span location of outboard end of flap, measured perpendicular
*           to vertical plane of symmetry
*   CPRMEI  Total wing chord at inboard end of flap (translating devices
*           only) measured parallel to longitudinal axis
*           (NDELTA values in array)
*              Single-slotted, Fowler, Double-slotted, leading-edge
*              slats, Krueger flap, jet flap
*   CPRMEO  Total wing chord at outboard end of flap (translating devices
*           only) measured parallel to longitudinal axis
*           (NDELTA values in array)
*              Single-slotted, Fowler, Double-slotted, leading-edge
*              slats, Krueger flap, jet flap
*   CAPINS  (double-slotted flaps only) (NDELTA values in array)
*   CAPOUT  (double-slotted flaps only) (NDELTA values in array)
*   DOSDEF  (double-slotted flaps only) (NDELTA values in array)
*   DOBCIN  (double-slotted flaps only)
*   DOBCOT  (double-slotted flaps only)
*   SCLD    Increment in section lift coefficient due to
*           deflecting flap to angle DELTA[i]      (optional)
*           (NDELTA values in array)
*   SCMD    Increment in section pitching moment coefficient due to
*           deflecting flap to angle DELTA[i]      (optional)
*           (NDELTA values in array)
*   CB      Average chord of the balance    (plain flaps only)
*   TC      Average thickness of the control at hinge line
*           (plain flaps only)
*   NTYPE   Type of nose
*              1.0  Round nose flap
*              2.0  Elliptic nose flap
```

```
*              3.0   Sharp nose flap
*   JETFLP   Type of flap
*              1.0   Pure jet flap
*              2.0   IBF
*              3.0   EBF
*   CMU      Two-dimensional jet efflux coefficient
*   DELJET   Jet deflection angle(NDELTA values in array)
*   EFFJET   EBF Effective jet deflection angle(NDELTA values in array)

  $SYMFLP FTYPE=1.0,
   NDELTA=9.0, DELTA(1)=-29.0,-24.0,-19.0,-14.0,-9.0,-4.0,0.0,4.0,8.0,
   HETE=0.0522, PHETEP=0.0523, CHRDFI=1.21,  CHRDFO=0.69,
   SPANFI=1.09, SPANFO=9.0, CB=0.40, TC=0.061, NTYPE=1.0$
```

2.4 DATCOM: OUTPUT FILE

As output DATCOM returns a long file, partially included in figure 2.4.2. In addition to all the main aerodynamic derivatives, the lift, drag and moment contributions due to the possible deflection of moving surfaces are also provided.

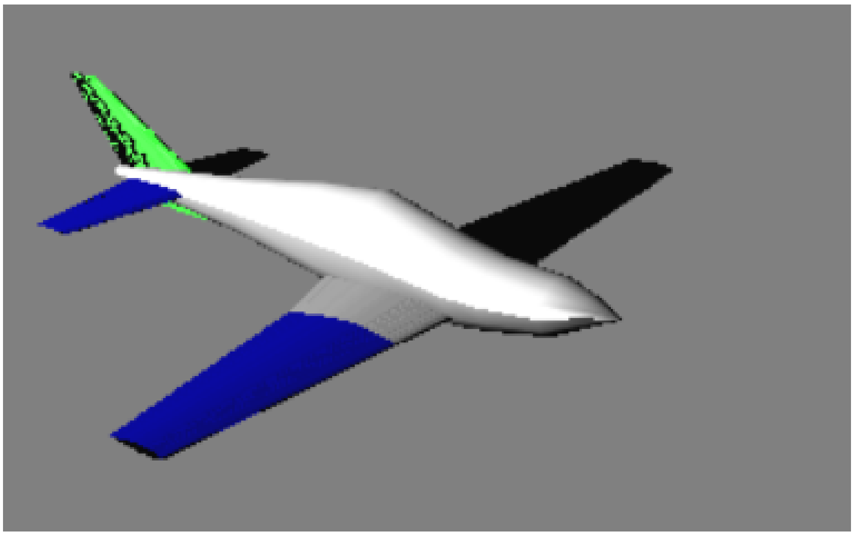

Figure 2.4.1: BS PRIME on Digital Datcom

It should be noted that all the reported coefficients refer to the reference system centered in the datum, with axis X_{datum} towards the nose, axis Z_{datum} downwards, and axis Y_{datum} towards the right of the pilot. The coefficients C_L and C_D are exceptions, since lift and drag are taken with a positive direction, respectively perpendicular and parallel to the speed (therefore in wind axes). The coefficients CN and CA are also an exception, as they represent the coefficients of the overall forces in the normal and axial direction, assumed with a positive direction respectively upwards and towards the tail of the aircraft,therefore in a reference system opposite to the one used for the remaining coefficients.

.

```
 NUMBER                                                          NUMBER                AREA       LONG.     LAT.     HORIZ      VERT
                FT        FT/SEC      LB/FT**2        DEG R         1/FT                 FT**2        FT        FT        FT         FT
0   .206    4000.00      226.78    1.8277E+03      504.408      1.0000E+06             102.360      4.110    25.453     4.746      .000
0                                                                  -------------------DERIVATIVE (PER RADIAN)-------------------
0 ALPHA       CD        CL        CM        CN        CA        XCP        CLA          CMA          CYB          CNB          CLB
0
   -8.0     .040     -.653     .6304    -.652    -.052     -.967   5.738E+00   -6.552E+00   -4.271E-01    1.316E-01   -3.044E-02
   -6.0     .027     -.455     .4125    -.455    -.021     -.907   5.616E+00   -6.126E+00                             -3.226E-02
   -4.0     .019     -.261     .2028    -.262     .001     -.775   5.467E+00   -5.888E+00                             -3.390E-02
   -2.0     .015     -.073     .0014    -.073     .013     -.019   5.315E+00   -5.671E+00                             -3.531E-02
     .0     .016      .110    -.1931     .110     .016    -1.753   5.320E+00   -5.578E+00                             -3.652E-02
    1.0     .017      .204    -.2905     .204     .014    -1.424   5.403E+00   -5.602E+00                             -3.721E-02
    2.0     .020      .299    -.3886     .299     .009    -1.299   5.489E+00   -5.643E+00                             -3.795E-02
    4.0     .028      .493    -.5871     .494    -.006    -1.189   5.642E+00   -5.767E+00                             -3.958E-02
    6.0     .042      .693    -.7912     .693    -.031    -1.141   5.793E+00   -5.961E+00                             -4.137E-02
    8.0     .060      .898   -1.0033     .897    -.065    -1.118   5.808E+00   -6.073E+00                             -4.326E-02
   10.0     .083     1.098   -1.2152    1.096    -.109    -1.109   5.264E+00   -5.774E+00                             -4.475E-02
   12.0     .107     1.265   -1.4063    1.260    -.158    -1.116   4.353E+00   -5.057E+00                             -4.480E-02
   14.0     .132     1.402   -1.5682    1.392    -.211    -1.126   3.322E+00   -3.800E+00                             -4.388E-02
   15.0     .143     1.455   -1.6272    1.442    -.239    -1.128   2.857E+00   -3.598E+00                             -4.296E-02
   16.0     .154     1.502   -1.6938    1.486    -.266    -1.140   2.528E+00   -4.034E+00                             -4.157E-02
0                                            ALPHA      Q/QINF     EPSLON   D(EPSLON)/D(ALPHA)
0
                                              -8.0      1.000      -3.606          .521
                                              -6.0      1.000      -2.564          .525
                                              -4.0      1.000      -1.506          .529
                                              -2.0      1.000       -.447          .532
                                                .0      1.000        .623          .548
                                               1.0      1.000       1.177          .561
                                               2.0      1.000       1.745          .571
                                               4.0      1.000       2.898          .568
                                               6.0      1.000       4.018          .551
                                               8.0      1.000       5.103          .528
                                              10.0      1.000       6.131          .473
                                              12.0      1.000       6.996          .381
                                              14.0       .961       7.653          .291
                                              15.0       .919       7.925          .252
                                              16.0       .908       8.157          .232
1                                   AUTOMATED STABILITY AND CONTROL METHODS PER APRIL 1976 VERSION OF DATCOM
                                                               DYNAMIC DERIVATIVES
                                      WING-BODY-HORIZONTAL TAIL-VERTICAL TAIL-VENTRAL FIN CONFIGURATION
                                                     TOTAL: AILERONS: BS PRIME Aircraft

 ----------------------  FLIGHT CONDITIONS  ------------------------          --------------  REFERENCE DIMENSIONS  ------------
  MACH    ALTITUDE   VELOCITY    PRESSURE    TEMPERATURE     REYNOLDS              REF.     REFERENCE LENGTH   MOMENT REF. CENTER
 NUMBER                                                       NUMBER               AREA       LONG.     LAT.     HORIZ      VERT
                FT        FT/SEC      LB/FT**2        DEG R         1/FT                 FT**2        FT        FT        FT         FT
0   .206    4000.00      226.78    1.8277E+03      504.408      1.0000E+06             102.360      4.110    25.453     4.746      .000
                                                          DYNAMIC DERIVATIVES (PER RADIAN)
0           -------PITCHING-------     -----ACCELERATION------    --------------ROLLING--------------    --------YAWING--------
0    ALPHA      CLQ          CMQ           CLAD         CMAD         CLP          CYP          CNP          CNR          CLR
0
    -8.00   1.694E+01   -3.135E+01      3.407E+00   -1.290E+01   -5.034E-01   -1.248E-01    6.962E-02   -3.577E-01   -7.195E-02
    -6.00                               3.432E+00   -1.299E+01   -4.875E-01   -1.169E-01    5.035E-02   -3.522E-01   -4.330E-02
    -4.00                               3.460E+00   -1.310E+01   -4.706E-01   -1.089E-01    3.117E-02   -3.495E-01   -1.568E-02
    -2.00                               3.480E+00   -1.318E+01   -4.543E-01   -1.007E-01    1.197E-02   -3.495E-01    1.063E-02
      .00                               3.582E+00   -1.356E+01   -4.549E-01   -9.244E-02   -7.168E-03   -3.517E-01    3.591E-02
     1.00                               3.668E+00   -1.389E+01   -4.633E-01   -8.831E-02   -1.685E-02   -3.537E-01    4.894E-02
     2.00                               3.730E+00   -1.412E+01   -4.719E-01   -8.420E-02   -2.655E-02   -3.563E-01    6.229E-02
     4.00                               3.717E+00   -1.407E+01   -4.869E-01   -7.600E-02   -4.610E-02   -3.635E-01    8.981E-02
     6.00                               3.606E+00   -1.365E+01   -4.990E-01   -6.787E-02   -6.585E-02   -3.736E-01    1.182E-01
     8.00                               3.453E+00   -1.307E+01   -4.952E-01   -5.985E-02   -8.661E-02   -3.868E-01    1.472E-01
    10.00                               3.094E+00   -1.172E+01   -4.375E-01   -5.199E-02   -1.109E-01   -4.021E-01    1.747E-01
    12.00                               2.489E+00   -9.422E+00   -3.511E-01   -4.380E-02   -1.355E-01   -4.152E-01    1.949E-01
    14.00                               1.825E+00   -6.911E+00   -2.687E-01   -3.529E-02   -1.585E-01   -4.255E-01    2.094E-01
    15.00                               1.513E+00   -5.730E+00   -2.200E-01   -3.093E-02   -1.696E-01   -4.290E-01    2.140E-01
    16.00                               1.379E+00   -5.219E+00   -1.677E-01   -2.622E-02   -1.783E-01   -4.310E-01    2.168E-01
```

Figure 2.4.2: DATCOM results

2.5 GEOMETRY

Below are the fundamental quantities obtained from measurements with caliper and protractor and calculated by DATCOM

	Symbol	Value	Unit
Wing Surface Area	S	9.44	[m2]
Wing Span	b	7.9	[m]
Mean Aerodynamic Chord	$\bar{c}$	1.296	[m]
Aspect Ratio	A	6.61	[-]
Wing Incidence Angle	iw	0	[deg]
Wing Dihedral Angle	Γ	4	[deg]
Taper Ratio	λ	0.339	[-]
Root Chord	c_r	1.9	[m]
Tip Chord	c_t	0.643	[m]

Table 2.5.1: BS Prime Wing Characteristics on DATCOM

	Symbol	Value	Unit
HT Surface Area	S_h	2.11	[m2]
HT Span	b_h	3.02	[m]
Mean Aerodynamic Chord	$\bar{c_h}$	0.735	[m]
Aspect Ratio	A_h	4.315	[-]
HT Taper Ratio	λ_h	0.444	[-]
Root Chord	c_{rh}	0.969	[m]
Tip Chord	c_{th}	0.43	[m]
Dihedral angle	Γ	-3	[Deg]

Table 2.5.2: BS Prime HT Characteristics on DATCOM

	Symbol	Value	Unit
VT Surface Area	S_v	1.28	[m2]
VT Span	b_v	1.18	[m]
Mean Aerodynamic Chord	$\bar{c_v}$	1.18	[m]
Aspect Ratio	A_v	1.1	[-]
HT Taper Ratio	λ_v	0.308	[-]
Root Chord	c_{rv}	1.65	[m]
Tip Chord	c_{th}	0.51	[m]

Table 2.5.3: BS Prime VT Characteristics on DATCOM

	Symbol	Value	Unit
Elevator Span	b_e	2.74	[m]
Root Chord	c_{re}	0.369	[m]
Tip Chord	c_{te}	0.21	[m]

Table 2.5.4: BS Prime Elevator Characteristics on DATCOM

	Symbol	Value	Unit
Aileron Span	b_a	1.38	[m]
Root Chord	c_{ra}	0.287	[m]
Tip Chord	c_{ta}	0.174	[m]

Table 2.5.5: BS Prime Aileron Characteristics on DATCOM

	Symbol	Value	Unit
Flap Span	b_f	1.84	[m]
Root Chord	c_{rf}	0.373	[m]
Tip Chord	c_{tf}	0.232	[m]

Table 2.5.6: BS Prime Flap Characteristics on DATCOM

	Symbol	Up	Down	Unit
Flap To	δ_f	-	10	[Deg]
Flap Landing	δ_f	-	30	[Deg]
Elevatore	δ_e	29	8	[Deg]
Right Aaileron	δ_a	23	28	[Deg]
left aileron	δ_a	28	23	[Deg]

Table 2.5.7: BS Prime Surface deflections on DATCOM

Chapter 3

Mass distribution

3.1 Introduction

To calculate the z coordinate of the center of gravity and inertias, it is necessary to estimate a distribution of the masses for the various components of the aircraft. The procedure that led to the results reported here will be explained in the following. The basic empty weight of 390 kg [1, p. 114] has been split into the following components:

- wing = 55 kg
- horizontal tail = 15 kg
- vertical tail = 13 kg
- engine = 67.7 kg
- propeller = 18 kg
- main landing gear = 24 kg
- nose landing gear = 10 kg
- body = 187.3 kg

3.2 Mass distribution

Since in the flight manual there are no documents with the mass distribution, we opted as a first attempt to follow the reasoning proposed by Raymer [10] , that is to use percentages to attribute the mass to each element. Later it is used the analogy with the Bs 115 mass distribution.

3.2.1 Mass distribution by Raymer

A first estimate was made using the percentages found in the Raymer [10, p. 569], which however provide not entirely reliable results, as we later verified. In fact, the mass distributions reported by the Raymer are very generic, for a wide range of aircraft. In addition, it is observed that the subdivision into masses is carried out starting from the MTOW.

Element	Raymer Percentage	Mass [kg]
Wing	0.16	96
Vertical tail	0.033	19.8
Horizontal tail	0.043	25.8
Landing gear	0.055	33

Table 3.2.1: Mass distribution by Raymer's coefficients

3.2.2 Mass distribution starting from BS115

To better estimate this distribution, the data found on "Handling qualities criteria for training effectiveness assessment of the BS115 aircraft" [see 6], which refers to Blackshape 115, were used as a starting point. Since the Blackshape 115 has a higher empty weight, we calculated the mass percentages of the various elements of the BS 115 as:

$$\% = mass/empty\ weight \tag{3.2.1}$$

These percentages were then used to define the masses of the elements listed for the Blackshape Prime. Below is the table with the mass distribution of the BS 115 [6, p. 77].

	Weight [kg]	X_{CG} [mm]
Fuel Tank	10.8	45
Fuselage	75.5	1003
Tail Cover	2.5	4460
Horizontal Stabilizer	8.6	4030
Elevators	6.9	4313
Rudder	3.1	4528
Wing	47.8	553
Main Landing Gear	31.5	432
Nose Landing Gear	12.5	-752
Aileron Control	3.0	295
Rudder Control	11.9	417
Control Column	4.7	365
Elevator Control	2.0	2419
Flaps Control	2.9	743
Engine Mount	10.4	-1046
Propeller	24.1	-1920
Firewall	3.2	-1001
Engine	145.5	-1430
Oil Cooler	1.3	-909
Avionics	33.3	2122
Engine Cowling	5.0	-1300
Exhaust	4.2	-1350
Instrument Panel	12.7	-260
Fuel System	7.5	-450
Landing Gear Doors	1.3	150
Seat Upholstery Forward	2.5	507
Seat Upholstery Rearward	2.5	1282
Canopy	13.5	534
Miscellaneous	18.5	273
Unusable Fuel	11.0	45
Total	520.0	19.0 %

Figure 3.2.1: Complete mass distribution of BS 115

In this case it was not possible to use such a detailed list, both because it was not present in the flight manual, and because it would have been impossible with the data available to calculate the positions along XYZ of all the listed components. The components have therefore been grouped as follows:

- Wing: fuel tank, wing, aileron control, flap control.
- Body: fuselage, firewall, oil cooler, avionics, engine cowling, exhaust, instrument panel, fuel system, landing gerar doors, set upholstery Forward and Reaeward, canopy, engine mount, control column, miscellaneous, unusable fuel.
- Vertica tail: tail cover, rudder, rudder control.
- Horizontal tail: horizzontal stabilizer, elevators, elevator controller
- Main landing gear: main landing gear
- Nose landing gear: nose landing gear
- Engine: engine
- Propeller: propeller (with nozzle)

Element	Mass $[kg]$	%
Wing	64.5	12.4
Vertical tail	17.5	3.37
Horizontal tail	17.5	3.37
Body	207	39.88
Engine	145.5	27.98
Propeller	24.1	4.63
Nose landing gear	12.5	2.40
Main landing gear	31.5	6.06

Table 3.2.2: Mass distribution of BS 115

Element	%	Mass $[kg]$
Wing	12.4	48.36
Vertical tail	3.37	13.14
Horizontal tail	3.37	13.14
Body	39.88	155.19
Engine	27.98	109.12
Propeller	4.63	18.06
Nose landing gear	2.40	9.36
Main landing gear	6.06	23.63

Table 3.2.3: First mass distribution of BS Prime

As a result of this division, the percentages in the table have been calculated in Table 3.2.2. Starting from these percentages, the mass distribution for the BS Prime is determined , as in Table 3.2.3. It was necessary to make corrections due to the weight of the BS Prime engine: through the technical sheet it was possible to find the real mass of the engine, and verify that it was not in line with the one calculated as a percentage reported in Table 3.2.3. In fact, the actual weight of the engine with all its components is 67.7 kg [2], i.e. 40 kg lower compared to the value in the table.

Element	Mass $[kg]$
Wing	55
Vertical tail	13
Horizontal tail	15
Body	187.3
Engine	67.7
Propeller	18
Nose landing gear	10
Main landing gear	24

Table 3.2.4: Final mass distribution of BS Prime

The idea was to make some corrections to the other masses so that the total basic empty weight was 390 kg, as reported in the flight manual. To do this, however, we tried to stay within the range of the values shown in Table 3.2.3. In general, the masses have rounded up, in particular the 40 kg difference has been mainly attributed to the Body.

3.3 Evaluation of C.G. position

For the evaluation of the center of gravity it is used the flight manual and the mass distribution defined above.

3.3.1 Position of $x_{c.g}$ with respect to the datum

As misured from the flight manual, the datum is located at $x_d = 1.45\ m$ with respect to the nose of the aircraft and belongs to the X axis. To calculate the position of the center of gravity along X, the procedure described in the flight manual [1, p. 114] is used, where the distances and moments with respect to the datum of luggage, fuel, passenger and pilot are provided . Remember that the two fuel tanks are symmetrical with respect to the Y axis and they are positioned in the wing. In the flight manual the position of the center of gravity is calculated as a percentage of the mean aerodynamic chord according to the following formula:

$$x_{c.g\%} = \frac{\frac{Moment}{Weight} - 0.68}{1.252} \cdot 100 \tag{3.3.1}$$

	ITEM	ARM [m]	WEIGHT [kg]	MOMENT [kg*m]	C.G. [% MAC]
1	**Basic empty weight**		390	/	
2	**Pilot**	/	90(+)	/	
3	**Passenger**	/	0(+)	/	
4	**Baggage**	/	15(+)	/	
5	**Fuel**	/	40(+)	/	
	Take off condition	/	535	503.75	21%

Figure 3.3.1: BS Prime table for c.g. computation

To find the position of the center of gravity along the X axis with respect to the position of the datum ($x_{c.g}$), take the equation 3.3.1. Since the result is a percentage of the mean aerodynamic chord, it has to be multiplied by the value of the MAC and referenced with respect to the datum:

$$x_{c.g} = 680 + \frac{\frac{Moment}{Weight} - 0.68}{1.252} \cdot \bar{c} \tag{3.3.2}$$

Where $680mm$ is the distance along the X axis between the datum and the leading edge at the mean aerodynamic chord, as shown in 3.3.2. Before starting the calculation of the trim conditions with FDC, it will therefore be necessary to enter the weight of the baggage, the fuel on board, the pilot and any passenger.

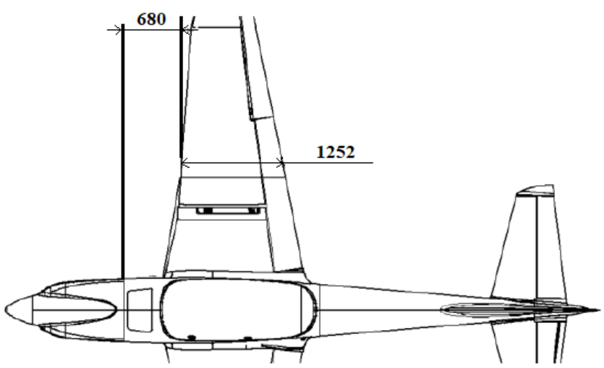

Figure 3.3.2: BS Prime plan wiev

3.3.2 Elements position along X_{datum} axis

It was still necessary to determine the positions along X axis an initial approximation of the elements shown in the table 3.3.1 for the calculations of the inertias, with the hypothesis of concentrated masses. In the table, "computed" means that in the case of the wing and tail, it has been assumed that the

Element	x_d $[m]$	Evaluating method
propeller	1.450	caliper measurment
engine	0.5400	caliper measurment
wing	-1.1182	computed
Horizontal tail	-4.9	computed
Vertical tail	-4.8	computed
pilot	-1.0	from flight manual
passenger	- 1.8	from flight manual
main landing gear	-1.2380	from flight manual
nose landing gear	0.5	from flight manual
fuel	-0.75	from flight manual
baggage	-2.25	from flight manual
body	-1.4851	computed

Table 3.3.1: Elements position along Z_{datum}

position x_d of the element is at one third of their mean aerodynamic chord. For the body, the x_d position of the body of the BS 115 was calculated first, as all the necessary data was available. Then, a value was obtained and was rescaled for the length of the fuselage itself, as the BS Prime one has different dimensions.

3.3.3 Elements position along Y_{datum}

The plane is assumed to be symmetrical about the X_{datum} axis, so many elements will have a zero position along the Y axis. It follows that the overall center of gravity of the aircraft is on the X axis, $y_d = y_{c.g} = 0$ and it is therefore not necessary to evaluate this coordinate.

Element	y_d $[m]$	Evaluation method
propeller	0	hypothesis
engine	0	hypothesis
right half wing	0.9337	computed
left half wing	-0.933	computed
right horizontal tail	0.6579	computed
left horizontal tail	-0.6579	computed
vertical tail	0	hypothesis
pilot	0	hypothesis
passenger	0	hypothesis
main landing gear	0	hypothesis
nose landing gear	0	hypothesis
fuel a destra	0.732	hypothesis
fuel a sinistra	-0.732	hypothesis
baggage	0	hypothesis
body	0	hypothesis

Table 3.3.2: Elements position along Y_{datum}

However, it was necessary to calculate the position along the Y_{datum} axis for: the center of gravity of the half-wings, the half-horizontal tail, and the fuel. These values are used to calculate the moments of inertia. The Table 3.3.2 shows the values found for all the other components (such as engine, fuselage, vertical tail, etc.). It is assumed that their center of gravity falls on the X_{datum} axis. It should be noted that left and right in the Table 3.3.2 are meant with respect to the pilot. An indicative Y coordinate value was assumed for the position of the fuel, based on the drawings shown in the flight manual, [1, p. 128]. For the calculation of the center of gravity, each of the two half-wings was considered to be formed by two trapezoids: one from the root of the wing to the breakpoint, the one in blue in the Figure 3.3.3; the other up to the tip, the one in green in the Figure 3.3.3. Exactly as it was done on DATCOM.

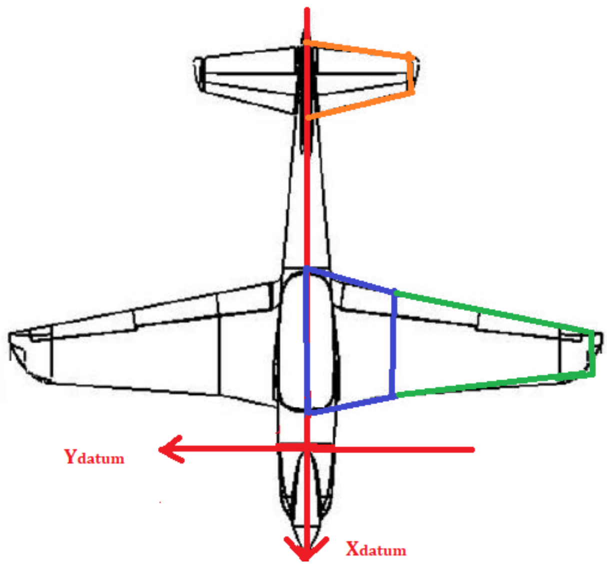

Figure 3.3.3: Wing and Tail approximation for c.g. computation

The same measures already considered for the geometric model were used. For the wing we calculated the center of gravity of each trapezoid and subsequently the overall center of gravity, making a weighted average on the surfaces of the individual trapezoids. The geometric centers of gravity of the trapezoids along Y are calculated as follows:

$$y_{d_{semiwing\ left\ external}} = -h_e/3(c_{breakpoint} + 2 \cdot c_{tip})/(c_{breakpoint} + c_{tip}) + h_i \tag{3.3.3}$$

$$y_{d_{semiwing\ left\ internal}} = -h/3(c_{root} + 2 \cdot c_{breakpoint})/(c_{breakpoint} + c_{root}) \tag{3.3.4}$$

Where c indicates the chord, h the height of the single trapezoid intended as the distance between the two bases. The center of gravity of the wing is defined as the weighted average on the surfaces as follows:

$$y_{d_{semiwing\ left}} = \frac{y_{d_{semiwing\ left\ external}} \cdot S_{tapezoid\ external} + y_{d_{semiwing\ left\ internal}} \cdot S_{tapezoid\ internal}}{S_{tapezoid\ external} + S_{tapezoid\ internal}} \tag{3.3.5}$$

Where S indicates the surface of the trapezoid calculated as

$$S = (Base\ major + Base\ minor) \cdot height/2$$

The same reasoning applies to the right wing, but since they are identical, the only difference will be the sign of the final result, given the different position with respect to the reference system. So $y_{d_{wing\ right}} = -\ y_{d_{wing\ left}}$. A similar process was used for the horizontal tail; in this case, however, the structure was assimilated to a single trapezoid:

$$y_{d_{semitail\ left}} = -h/3(c_{root} + 2 \cdot c_{tip})/(c_{root} + c_{tip}) \tag{3.3.6}$$

So for the same reasoning described above $y_{d_{semitail\ right}} = -\ y_{d_{semitail\ left}}$. All the results of the equations described are present in the matlab file "primo_run_fdc.m".

3.3.4 Elements position along Z_{datum}

For what concerns the position of the center of gravity along the Z axis, in the complete absence of precise data, only the measurements with calipers on the drawings in the flight manual were carried out. In this way the value of z_d was estimated as a first approximation. Note that the measure of the z coordinate of the vertical tail was calculated using the formulas and the reasoning implemented for the position of the y coordinate of the horizontal tail. That is by approximating the vertical tail to a trapezoid, so the coordinate along Z-axis is determined by calculating the geometric center of gravity as follows:

$$z_{d_{tail\ vertical}} = h/3(c_{root} + 2 \cdot c_{tip})/(c_{root} + C_{tip}) + d \tag{3.3.7}$$

Where d is the distance between the c_{root} of the vertical tail and the X_{datum} axis. As for the wing and the horizontal tail, they are characterized by a certain dihedral angle, therefore the position of the coordinate of the center of gravity along the Z axis must take this into account. In fact it is determined as

$$z_{d_{tail\ horizontal}} = z_{c_{root\ tail}} + |y_{semitail\ left}| \cdot \tan(\Gamma_{tail}) \tag{3.3.8}$$

Where $z_{c_{root\ tail}}$ indicates the position along the Z axis of the chord at the root of the vertical tail, and Γ_{tail} represents the dihedral angle of the tail, in this case negative for the BS Prime. For the position of the center of gravity of the wing along the z coordinate, the same procedure is valid as for the tail, that is:

$$z_{g_{wing}} = z_{c_{root\ wing}} + |y_{semiwing\ left}| \cdot \tan(\Gamma_{wing}) \tag{3.3.9}$$

The table 3.3.3 shows the positions of the various elements along the Z_{datum} axis.

Element	z_d $[m]$	Evaluation method
propeller	0	measure
engine	0.1	measure
wing	0.28	computed
Right horizontal tail	-0.073	computed
Left horizontal tail	-0.073	computed
Vertical tail	-0.7	computed
pilot	0	measure
passenger	-0.1	measure
main landing gear	0.28	measure
nose landing gear	0.3	measure
fuel right	0.3	measure
fuel left	0.3	measure
baggage	0.1	measure
body	0.1	measure

Table 3.3.3: Elements position along Z_{datum}

Subsequently the position of the center of gravity of the aircraft was calculated with respect to the datum as:

$$z_{c.g} = \frac{\sum_{i=1}^{N} Mass_{element_i} \cdot z_d}{TOW} \tag{3.3.10}$$

3.3.5 Example of load configuration

In the previous section, only the formulas for determining the center of gravity of the aircraft have been reported and not the results, as they are influenced by other factors that may vary depending on the load condition, namely: pilot mass, passenger mass, fuel mass , baggage mass. Consider the following load condition:

	Mass $[kg]$	Momentum $[kg\ m]$
pilot	85	-
passenger	85	-
baggage	0	-
fuel	40	-
empty weight	390	-
TOW	600	648.17

Table 3.3.4: Load condition

From this diagram the position of the center of gravity of the aircraft with respect to the datum appears to be:

$$x_{c.g} = -1.028\ m \quad y_{c.g} = 0\ m \quad z_{c.g} = 0.069\ m$$

3.3.6 Positions referred to the center of gravity of the aircraft

To take into account the variation of the center of gravity during the flight phases and to be able to calculate the inertias with respect to the center of gravity of the aircraft, all the positions were referred to the center of gravity, and the body axes were considered instead of the datum axes. For example:

$$x_{c.g_{engine}} = -(x_{c.g} - x_{d_{engine}})$$

$$y_{c.g_{engine}} = -(z_{c.g} - z_{d_{engine}})$$

In this way the position of the center of gravity of each element is defined, referring it to the center of gravity of the aircraft and in accordance with the triad of Body axes, which differs in direction with respect to X_{datum} and Z_{datum}. Note that the aircraft's center of gravity position varies during flight due to fuel consumption. This excursion occurs most importantly for the $x_{c.g}$ component. In fact there is no variation of the y_g coordinate, as the fuel tanks are equal and symmetrical with respect to the axis of symmetry of the aircraft [1, p. 128] and the fuel is taken equally between the two tanks. As for the variation of the $z_{c.g}$ coordinate, refer to the following test shown in the Table 3.3.5:

Pilot $[kg]$	Pilot $[kg]$	Baggage $[kg]$	Fuel $[kg]$	TOW $[kg]$	% $\bar{c}$	$z_{c.g}$
85	85	0	40	600	27.97	-0.0736
85	85	0	10	570	28.96	-0.0612

Table 3.3.5: Variation of $z_{c.g}$ due to fuel consumption

The tests were conducted by manually varying the amount of fuel in the program "primo_run_fdc.m". As can be seen, under the same conditions, an initial fuel value of 40 kg was passed to a quarter, ie 10 kg. Following this consumption there was a variation of $z_{c.g}$ of 1.24 cm upwards. A very small variation, which in reality would be even more contained, since in the model described here the fuel is seen as a concentrated mass. In reality, however, when the fuel is fished by the pumps on the bottom of the tanks, the center of gravity of the fuel mass is lowered. The variation of fuel related to consumption is defined in chapter 6.

3.3.7 Weight and Moment envelope and C.G. envelope

Note that the load diagram with the relative moments must lead to take-off weight and total moment values that fall within the Weight and Moment envelope [1, p. 115] , as shown in the Figure 3.3.5. At the same time, the position of the center of gravity along the X axis associated with the take-off weight must fall within the C.G. envelope. [1, p. 116], as shown in Figure3.3.4. These controls are implemented in the spreadsheet "primo_run_fdc.m". The control on the percentage of center of gravity with respect

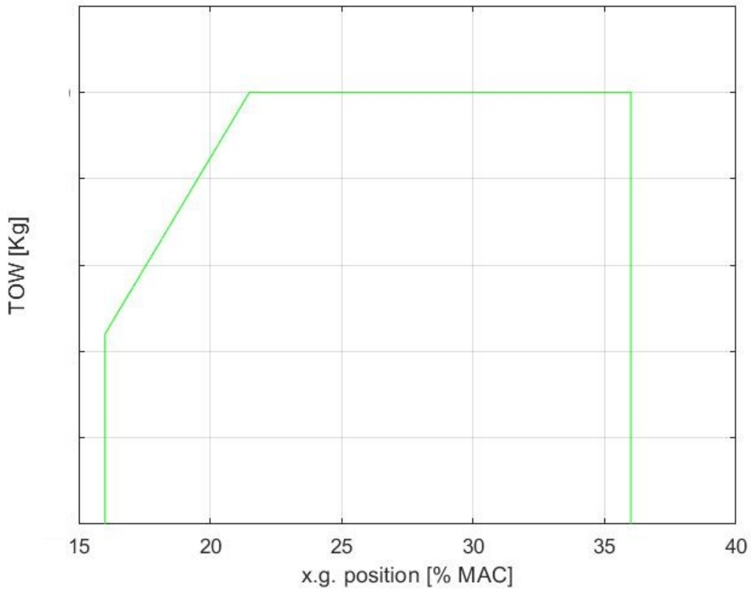

Figure 3.3.4: BS Prime c.g. envelope

to the mean aerodynamic chord is carried out using the Figure 3.3.4. If the weight is less than 460 kg the C.G. in % of mac must be between 16 % and 36 %. If the weight is more than 460kg and the C.G. is between 16 % of mac and 21.5 % of mac, then since there is a linear section, starting from the weight of the aircraft, the position of the center of gravity in % of mac which belongs to the straight line is calculated. The latter is compared with the position of the C.G. % of mac of the aircraft, which must be greater. When the weight is over 460kg and the C.G. % of mac is greater than 21.5 % of mac, then it is sufficient that it meets the condition of RWD.

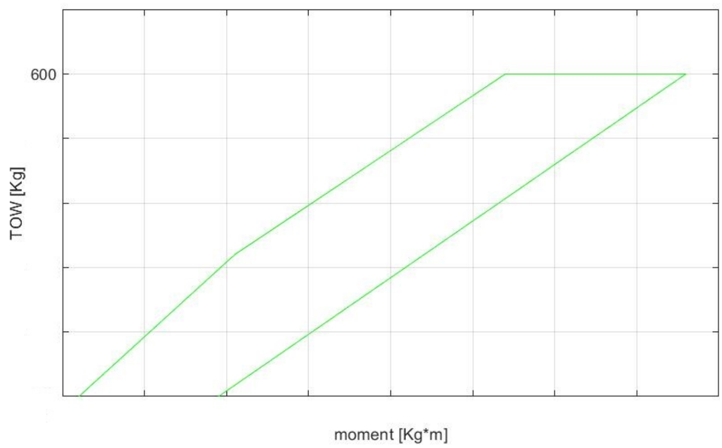

Figure 3.3.5: BS Prime Weight and Moment envelope

The mass and total moment are checked using Figure 3.3.5.

3.3.8 Fuel

It should also be noted that the mass value of fuel has some limitations, as the tanks contain a maximum of 31 liters each, therefore it is possible to embark a maximum of 62 liters. Of these 62 liters, those actually usable can be 58 liters [1, p. 11]. Possible fuels are:

- Min. RON 95 density$(15°C)$ = 720÷775 kg/m^3, [12]
- EN 228 Super density$(15°C)$ = 720÷775kg/m^3, [14]
- EN 228 Super plus density$(15°C)$ = 720÷775 kg/m^3, [15]
- AVGAS 100 LL density$(15°C)$ = 720÷768 kg/m^3, [18]

In this discussion an average value of 737.4 kg/m^3 is used. So with this density and a maximum volume of 62 liters, the maximum mass of fuel that can be loaded would be

$$fuel_{max} = 0.0062\ m^3 \cdot 737.4\ kg/m^3 = 45.7\ kg$$

In the matlab code a control on the mass of the fuel has been inserted.

3.4 Inertia computation

Given the difficulty of creating a sufficiently faithful CAD model, due to the almost total absence of data relating to geometry and mass distribution, the estimation of inertias was made in an approximate way. The starting point was a concentrated mass model, in which the inertia of each component with respect to the Body axes was calculated as the product of the component's mass times the square of its distance from the center of gravity:

$$I_{xx_i} = m_i \cdot (z_{c.g_i}^2 + y_{c.g_i}^2) \tag{3.4.1}$$

$$I_{yy_i} = m_i \cdot (z_{c.g_i}^2 + x_{c.g_i}^2) \tag{3.4.2}$$

$$I_{zz_i} = m_i \cdot (x_{c.g_i}^2 + y_{c.g_i}^2) \tag{3.4.3}$$

$$J_{xz_i} = m_i \cdot x_{c.g_i} \cdot z_{c.g_i} \tag{3.4.4}$$

Where $x_{c.g_i}$, $y_{c.g_i}$ and $z_{c.g_i}$ are the coordinates of the center of gravity of the nth mass with respect to the Body reference system. However, this model leads to unreliable results because, considering each body as a point mass, the "own" inertia is completely ignored, ie that due to the rotation of the body around its own center of gravity. These contributions were calculated as a first approximation by assimilating the various components to geometric solids, for which it was possible to find the formulas relating to the moments of inertia. The components considered, the geometric figures to which they have been assimilated, and the relative formulas for the moments of inertia are listed below. For the components not listed this contribution has been neglected.

- **Wing**: each wing was approximated to a rod for rotations around its X and Z axis, and to a cylinder for rotations around its Y axis (the radius equal to half of the mean aerodynamic chord):

$$I_{xx_{wing}} = (1/12) \cdot mass_{wing} \cdot (b/2)^2 \tag{3.4.5}$$

$$I_{zz_{wing}} = I_{xx_{wing}} \tag{3.4.6}$$

$$I_{yy_{wing}} = 1/2 \cdot mass_{wing} \cdot (\bar{c}/2)^2 \tag{3.4.7}$$

- **Horizontal tail**: each half wing of the tail was approximated to a rod for rotations around its X and Z axis, and to a cylinder for rotations around its Y axis (with a radius equal to half of the mean aerodynamic chord of the horizontal tail). Then the same formulas of the wing have been implemented, but using the data of the horizontal tail.

- **Vertical tail**: the surface was approximated to a rod for rotations around its X and Y axis, and to a cylinder for rotations around its Z axis (with a radius equal to half the mean aerodynamic chord of the vertical tail):

$$I_{xx_{tail_v}} = (1/12) \cdot mass_{tail_v} \cdot (h/2)^2 \tag{3.4.8}$$

$$I_{yy_{tail_v}} = I_{xx_{tail_v}} \tag{3.4.9}$$

$$I_{zz_{wing}} = 1/2 \cdot mass_{tail_v} \cdot (\bar{c}/2)^2 \tag{3.4.10}$$

- **Body**: the body has been approximated for the rotations around its own axis X, Y and Z with two cones (of base radius equal to the intermediate value between the width and height of the maximum section). In addition, it was assumed by analogy that the anterior cone has a greater mass, since there were more elements in the mass distribution of the BS 115. Therefore it is attributed to the anterior cone 2/3 of the mass of the body, and to the posterior cone 1/3

$$I_{xx_{body}} = (3/10) \cdot mass_{body} \cdot 2/3 \cdot (R_{average})^2 + (3/10) \cdot massa_{body} \cdot 1/3 \cdot (R_{average})^2 \tag{3.4.11}$$

$$I_{xx_{body}} = (1/12) \cdot mass_{body} \cdot (length_{body})^2 \tag{3.4.12}$$

$$I_{zz_{body}} = I_{yy_{body}} \tag{3.4.13}$$

- **Engine**: it has been approximated to a cylinder for rotations around its X, Y, and Z axis. The radius (R_m) was taken as the average between the height and width of the station where the engine is located. The length is half of the distance between the station where the engine is located and the firedoor. The gyroscope moment are neglected.

$$I_{xx_{engine}} = (1/2) \cdot mass_{engine} \cdot (R_m)^2 \tag{3.4.14}$$

$$I_{yy_{engine}} = (1/12) \cdot mass_{engine} \cdot (length_{engine})^2 \tag{3.4.15}$$

$$I_{zz_{engine}} = I_{yy_{engine}} \tag{3.4.16}$$

To summarize the procedure just described, it can be stated that the overall moments of inertia have been calculated as the sum of the “proper” moments of inertia of the components, plus the transport components with respect to the center of gravity. The final values, related to the specified load conditions, are shown below:

$$I_{xx_{tot}} = 290.88\ kg \cdot m^2 \quad I_{yy_{tot}} = 1641.0\ kg \cdot m^2 \quad I_{zz_{tot}} = 1859.31\ kg \cdot m^2$$
$$J_{xz_{tot}} = -58.65\ kg \cdot m^2 \quad J_{xy_{tot}} = 0\ kg \cdot m^2 \quad J_{zy_{tot}} = 0\ kg \cdot m^2$$

$J_{xy_{tot}} = 0\ kg \cdot m^2$ e $J_{zy_{tot}} = 0\ kg \cdot m^2$ for simmetry reasons. Having obtained the inertia values reported above, to verify these values it was decided to compare them as a first approximation with the inertias of aircraft present in the literature. Searching among the main texts, the smallest aircraft that comes closest in terms of MTOW to the BS Prime is the CESSNA 182.

Aircaft	Mass $[kg]$	$I_{xx_{tot}}$ $[kg\ m^2]$	$I_{yy_{tot}}$ $[kg\ m^2]$	$I_{zz_{tot}}$ $[kg\ m^2]$	$I_{xz_{tot}}$ $[kg\ m^2]$
CESSNA 182	1202.02	1274.5	1824.9	2666.8	0

Table 3.4.1: CESSNA 182 moment of inertia

The inertias computed for the Bs Prime $I_{yy_{tot}}$ *and* $I_{zz_{tot}}$ are similar to CESSNA 182, the others differ. Naturally this aspect does not affect the values of the inertias calculated for the BS Prime, as they strongly depend on the geometry and mass distribution of the aircraft. In fact, the CESSNA 182, in addition to having a double MTOW, is a high-wing aircraft, so in general it has a different geometry.

3.5 Spreadshit for c.g. and inertias

As already said, all the calculations for determining the position of the center of gravity and inertias, as the load condition varies, are carried out in the "primo_ run _ fdc.m" code on matlab. It should be noted that controls are implemented within it: on the MTOW, to prevent it from being too hight by changing the load condition; on fuel, to avoid entering a greater quantity than the maximum of the tanks, or forgetting to insert fuel; checks are carried out on the center of gravity, to confirm that it is confined within the range imposed by the flight manual; checks are inserted to verify that the weight and the overall moment are within the Weight and Moment Envelope diagram. If all the checks are satisfied you can proceed to the trim conditions, otherwise "not ready for trim condition" is printed. Below is an example of the output on the command window of "primo_ run _ fdc.m". The same code also allows you to implement the lookup tables relating to the efficiency and thrust coefficient of the propeller, see chapter 6

```
========================
Controls
========================

-> TOW = 600.0 Kg ok,  FUEL =40.0 Kg ok

-> Gravity center coordinates given from datum:

Xg=-1.028 m  Yg=0 m   Zg=0.069 m

-> Xg given as a percentage of main aerodynamic corde: 27.79
21.50< xg  MAC <36 satisfies limits

-> Weight and moment(616.79 Kgm) are in the range of Weight and Moment
    Envelope
Moment satisfies the condition 570.09 <Moment< 680.90

-> Moment of inertia:

Ixx=290.88  Iyy=1641.00 Izz=1859.31 Jxz =-58.65  in Kgm^2

Ready for trim condition
========================
```

Chapter 4

FDC: Flight dynamics and control toolbox

4.1 Introduction

FDC is a toolbox by matlab/simulink that allows you to easily implement the equations to model the behavior of the Beaver aircraft, i.e. the default model [9]. With appropriate modifications it is possible to implement the behavior of other aircrafts. To start, just run the FDC.m script. Subsequently it is possible to call the routines from the command window, or to use the "fdclib" command to open the simulink library and operate directly from there. As part of this project, the toolbox proved to be very useful as a starting point for the mathematical model, and to already have routines available that would allow for many calculations necessary for validation. It was of course necessary to adapt the model and the various routines to the BS prime. The main features of the toolbox will be illustrated below.

4.2 Structure of "BEAVER" model

In the toolbox there is a default simulink model related to the Beaver aircraft, an overview of which is shown. The set of equations is contained in the central block, which can be used in various ways. The

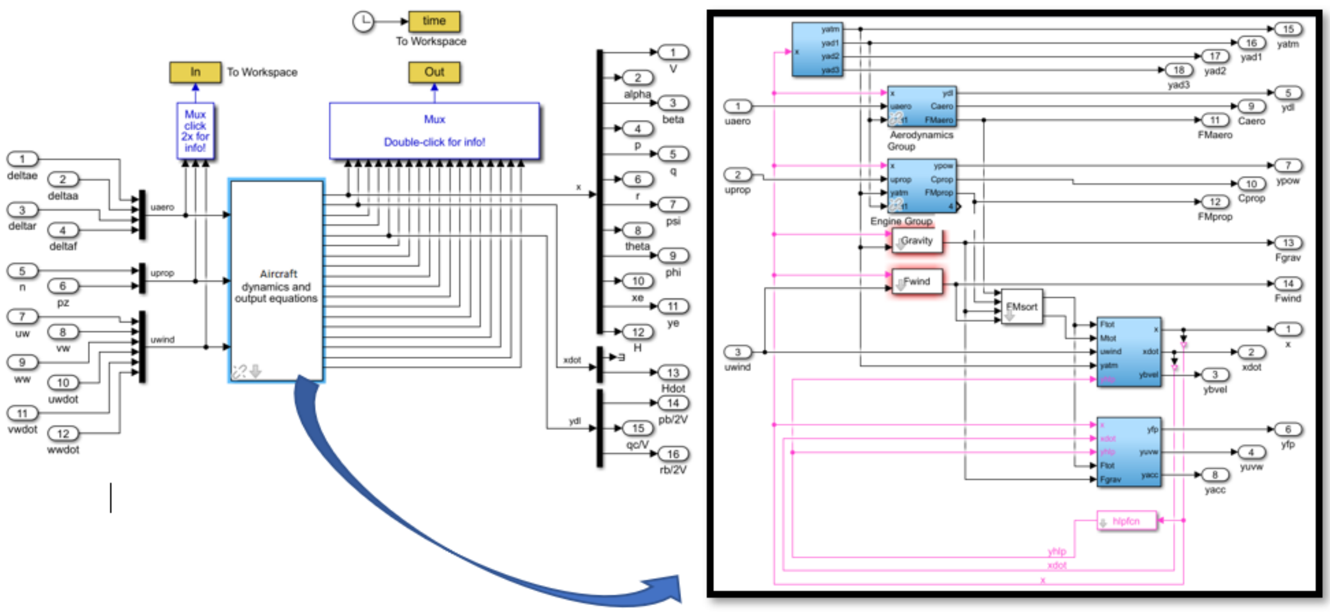

Figure 4.2.1: FDC simulik model

example shows a generic diagram that receives the pilot commands as input, along the wind speed and turbulence components. All the vectors shown in the right panel are provided as an output (for more information on the outputs it is possible to double click on the appropriate blue box inside simulink). However, the center block can be used in various ways such as to calculate trim conditions, analyze dynamic response, or to perform open loop and closed loop simulations. For the purposes of this project it was necessary to modify in particular the parts of the model that concern aerodynamics and propulsion. Therefore, a more detailed description of those original blocks, namely those intended for modeling the Beaver aircraft, will be given here. The changes made will be explained in the following chapters.

4.2.1 Aerodynamic model of "BEAVER"

The aerodynamic model in FDC is developed on three blocks: the first, Dimless, where the dimensionless components $q\bar{c}/V$, $bp/2V$, $br/2V$ are calculated; the second, Aeromod, where the angles are multiplied by the AM matrix containing the aerodynamic derivatives and their interpolation coefficients; the third, FMdims, where the aerodynamic forces are calculated.

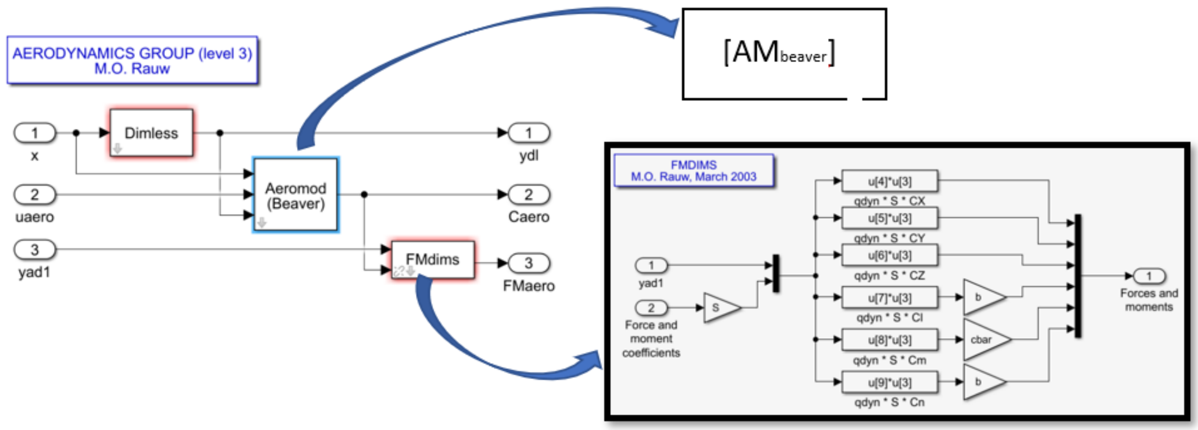

Figure 4.2.2: FDC Beaver aerodynimics simulik grup

The equations of motion used for the FMdims block are the three translation equations and the three rotation equations, considering the aircraft as a rigid body. Each one is characterized by its own force or moment coefficient. In the FDC model used for the Beaver, the equations of motion are calculated as follows:

$$\begin{aligned} F_{x_a} &= \frac{1}{2}\rho V^2 S C_x \\ F_{y_a} &= \frac{1}{2}\rho V^2 S C_y \\ F_{z_a} &= \frac{1}{2}\rho V^2 S C_z \\ L_a &= \frac{1}{2}\rho V^2 S b C_l \\ M_a &= \frac{1}{2}\rho V^2 S \bar{c} C_m \\ N_a &= \frac{1}{2}\rho V^2 S b C_n \end{aligned} \tag{4.2.1}$$

The force and moment coefficients reported above derive from the interpolation of the aerodynamic derivatives (Aeromod block, AM matrix), they are defined as follows:

$$
\begin{aligned}
C_x &= C_{x_0} + C_{x_\alpha}\alpha + C_{x_{\alpha^2}}\alpha^2 + C_{x_{\alpha^3}}\alpha^3 + C_{x_q}\frac{q\bar{c}}{V} + C_{x_{\delta_r}}\delta_r + C_{x_{\delta_f}}\delta_f + C_{x_{\alpha\delta_f}}\alpha\delta_f \\
C_y &= C_{y_0} + C_{y_\beta} + C_{y_p}\frac{pb}{2V} + C_{y_r}\frac{rb}{2V} + C_{y_{\delta_a}}\delta_a + C_{y_{\delta_r}}\delta_r + C_{y_{\delta_a\alpha}}\delta_a\alpha + C_{y_{\dot\beta}}\frac{\dot\beta b}{2V} \\
C_z &= C_{z_0} + C_{z_\alpha}\alpha + C_{z_{\alpha^3}}\alpha^3 + C_{z_q}\frac{\bar{c}q}{V} + C_{z_{\delta_e}}\delta_e + C_{z_{\delta_f}}\delta_f + C_{z_{\alpha\delta_f}}\alpha\delta_f + C_{z_{\delta_e\beta^2}}\delta_e\beta^2 \\
C_l &= C_{l_0} + C_{l_\beta}\beta + C_{l_p}\frac{bp}{2V} + C_{l_r}\frac{br}{2V} + C_{l_{\delta_a}}\delta_a + C_{l_{\delta_r}}\delta_r + C_{l_{\delta_a\alpha}}\delta_a\alpha \\
C_m &= C_{m_0} + C_{m_\alpha}\alpha + C_{m_{\alpha^2}}\alpha^2 + C_{m_q}\frac{\bar{c}q}{V} + C_{m_{\delta_e}}\delta_e + C_{m_{\delta_f}}\delta_f + C_{m_{\beta^2}}\beta^2 + C_{m_r}\frac{br}{2V} \\
C_n &= C_{n_0} + C_{n_\beta}\beta + C_{n_p}\frac{bp}{2V} + C_{n_r}\frac{br}{2V} + C_{n_{\delta_a}}\delta_a + C_{n_{\delta_r}}\delta_r + C_{n_q}\frac{\bar{c}q}{v} + C_{n_{\beta^3}}\beta^3
\end{aligned}
\tag{4.2.2}
$$

The Beaver's aerodynamic derivatives have been found by several experiments in the wind tunnel [16].

4.2.2 Engine model of "BEAVER"

The engine model for the Beaver, already present in FDC, is developed on three blocks: the first, Power, in which the power is calculated; the second, Engmod, in which the force coefficients are calculated, exploiting suitable derivatives relating to tests on the engine [16]; the third, FMdims, in which the dimensional forces are calculated.

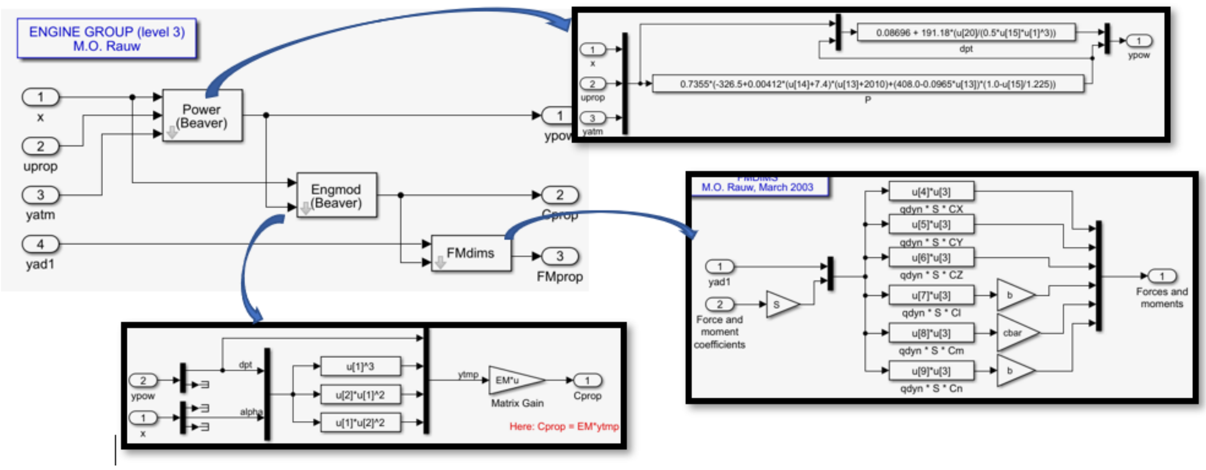

Figure 4.2.3: FDC Beaver engine simulik grup

4.2.3 Input Vector

Note that the input vectors have not been modified in the BS Prime model; therefore, they are the same as the model included in the FDC, which made integration easier.

- State vector: $X = [V\ \alpha\ \beta\ p\ q\ r\ \psi\ \theta\ \phi\ x_e\ y_e\ H]$.
- Engine controls vector: $uprop = [n\ pz]$, where pz is the manifold pressure.
- Vector with gravity and atmosferic conditions: $yatm = [\rho\ ps\ T\ \mu\ g]$

4.2.4 ACTRIM

The ACTRIM routine allows you to calculate trim conditions. First you will be asked to load the necessary data from the file, then you can choose the type of stationary flight to be analyzed (in this discussion, option 1 "Steady wings-level flight" has always been used). Finally, the flight conditions will be requested: H, V, heading, deltaf, MAP or Γ. The user has the possibility to choose whether to keep a certain manifold pressure value fixed or whether to impose the value of the flight path angle Γ. In the second case an initial estimate of the MAP value will be requested, but the real MAP value at sea level will be output anyway. ACTRIM actually converts the calculation of the trim conditions into a maximum and minimum problem, in which the function of cost to minimize is: $J = 10(\dot{u}^2 + \dot{v}^2 + \dot{w}^2) + 100(\dot{p}^2 + \dot{q}^2 + \dot{r}^2)$. For this process it is possible to modify appropriately the tolerance, and the maximum number of iterations and function evaluations. The cost function has high relevance because, in order to correctly calculate the trim conditions, it is necessary to ensure that the minimum found is global and not local. Once the process converges, and the conditions are correctly calculated, it is possible to save them to a file with the ".TRI" extension. During this project, additional routines were used which, starting from the ACTRIM code lines, made it possible to automate the calculation of trim conditions for different flight conditions, or for different initial MAP values. These routines will be better explained in the section about calculating trim conditions.

4.2.5 Cost function

As mentioned, the cost function for minimization is

$$J = 10(\dot{u}^2 + \dot{v}^2 + \dot{w}^2) + 100(\dot{p}^2 + \dot{q}^2 + \dot{r}^2) \tag{4.2.3}$$

It is different from that used for the Beaver model. To use this specific function it was necessary to modify ACTRIM by inserting the lines of code to calculate the new terms used. The speed components in body axes are:

$$\begin{aligned} u &= V \cos\alpha \cos\beta \\ v &= V \sin\beta \\ w &= V \sin\alpha \cos\beta \end{aligned} \tag{4.2.4}$$

The respective derivatives, introduced in ACTRIM for the evaluation of J are:

$$\begin{aligned} \dot{u} &= \dot{V} \cos\beta \cos\alpha + V(-\sin\beta \cos\alpha \cdot \dot{\beta} - \cos\beta \sin\alpha \cdot \dot{\alpha}) \\ \dot{v} &= \dot{V} \sin\beta + V \cos\beta \cdot \dot{\beta} \\ \dot{w} &= \dot{V} \cos\beta \sin\alpha + V(-\sin\beta \sin\alpha \cdot \dot{\beta} - \cos\beta \cos\alpha \cdot \dot{\alpha}) \end{aligned} \tag{4.2.5}$$

The tollernace used are: 'TolX' $1 \cdot 10^{-30}$,'MaxFunEvals' $5 \cdot 10^5$,'MaxIter' $5 \cdot 10^5$

Chapter 5

Aerodynamic model for Bs Prime

5.1 Introduction

To build the aerodynamic model for the BS Prime, as already mentioned, the results obtained with DATCOM are exploited, which can be easily imported into MATLAB with the "datcomimport" command. These results do not allow to obtain all the derivatives used in the Beaver model, so an integration with those present on Napolitano [7] was convenient. It should be noted that in the following discussion also for the BS prime it was decided to exploit the same interpolations of the coefficients used for the Beaver.

```
CX0    = -0.01804;    CZ0     = -0.1198;     Cm0  = -0.2036;
CXa    =  0.1118;    CZa     = -5.731;     Cma  = -5.831;
CXa2   =  5.03296;     CZa3    =  9.719;      Cma2 =  1.288;
CXa3   =  0;          CZq     =  16.94;      Cmq  = -31.35;
CXq    =  0;          CZde    = -0.2634;     Cmde = -0.9516;
CXdr   =  0;          CZdeb2 =  0.0;         Cmb2 =  0.0;
CXdf   = -0.01334;    CZdf    = -0.7792;     Cmr  = -0.0;
CXadf  =  0.4095;     CZadf   = -0.4301;     Cmdf = -0.8073;

CY0    = -0.0;        Cl0     =  0.0;        Cn0  = -0.0;
CYb    = -0.4271;     Clb     = -0.03905;    Cnb  =  0.1316;
CYp    = -0.0739;     Clp     = -0.41547;    Cnp  = -0.055921;
CYr    =  0.3076;     Clr     =  0.09279;    Cnr  = -0.37982;
CYda   = -0.0;        Clda    = -0.1467;     Cnda =  0.00161;
CYdr   =  0.0534;     Cldr    =  0.0033;     Cndr = -0.0349;
CYdra  =  0.0;        Cldaa   = -0.0;        Cnq  =  0.0;
CYbdot=  0.0;                                Cnb3 =  0.0;
```

Figure 5.1.1: Modbuild aerodynamic stability and control derivatives matrix

5.2 Elaboration of DATCOM derivatives

- C_{x_0}, C_{x_α}, $C_{x_{\alpha^2}}$, $C_{x_{\alpha^3}}$

These values were obtained by interpolating the axial force values CA (changed in sign to adapt them to the Datum reference system) of the complete aircraft as a function of alpha obtained by DATCOM, as shown in the figure.

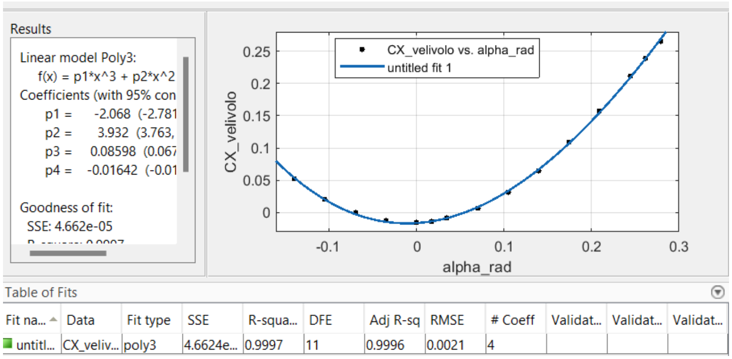

Figure 5.2.1: C_x interpolation

- $C_{x_{\delta_f}}$

It was obtained by interpolating the induced drag increments provided by DATCOM for the various deflections of the flaps (10° e 30°). Since DATCOM also gives the values as a function of alpha (thus building a matrix), the increments corresponding to $\alpha = 0$ have been considered. It takes into account the variation of C_x only due to the deflection of the flaps, so it is independent of the incidence. Therefore, at zero incidence, the coefficient of axial force coincides with the induced drag. The figure shows the graph relating to the interpolation.

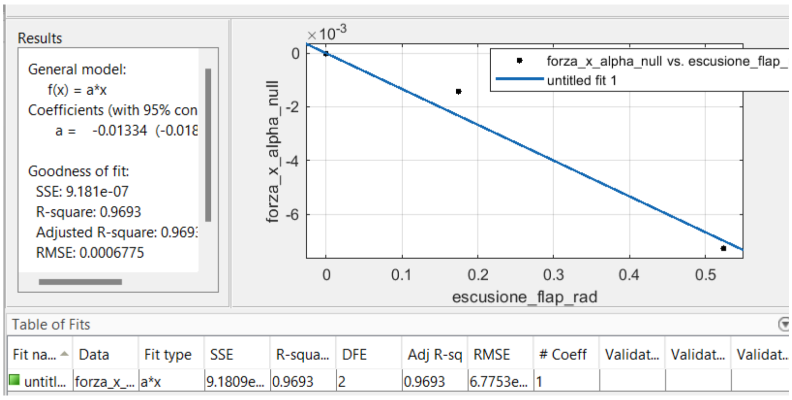

Figure 5.2.2: $C_{x_{\delta_f}}$ interpolation

- $C_{x_{\alpha\delta_f}}$

The drag variation due to the deflection of the flaps also depends on the alpha incidence, so this coefficient serves precisely to take this into account. In fact, note that when the incidence is not zero, it is necessary to take into account both the increase in lift and drag, and calculate the component of both forces along the X axis:

$$C_x = -C_L \cdot \sin(\alpha) + C_d \cdot \cos(\alpha) \tag{5.2.1}$$

Note that the 5.2.1 exploits the signs and conventions of DATCOM, so in order to conform to the datum reference system, the sign must be changed. An arbitrary number of pairs was taken to perform the interpolation (α, δ_f). For each pair of values, their product and the relative force increments along X was calculated. A graph was then created with the values of the product $\alpha \cdot \delta_f$ on the horizontal axis, and the corresponding increments of C_x on the vertical axis, and the values were linearly interpolated.

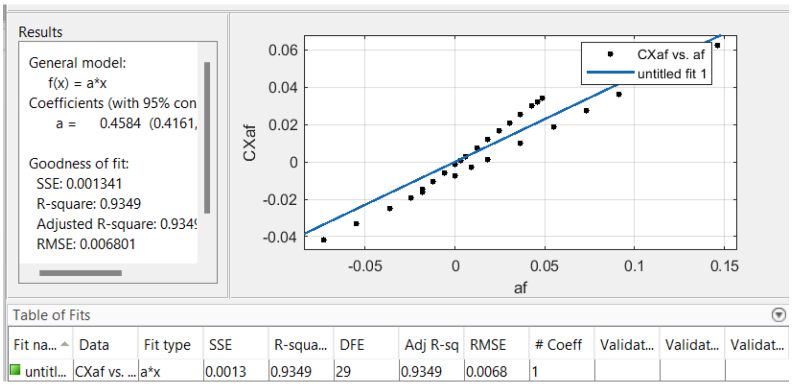

Figure 5.2.3: $C_{x_{\alpha\delta_f}}$ interpolation

- C_{z_0}, C_{z_α}, $C_{z_{\alpha 3}}$

These values were calculated with the same method as the analogous coefficients along X, interpolating the CN values (changed in sign to adapt them to the Datum reference system).

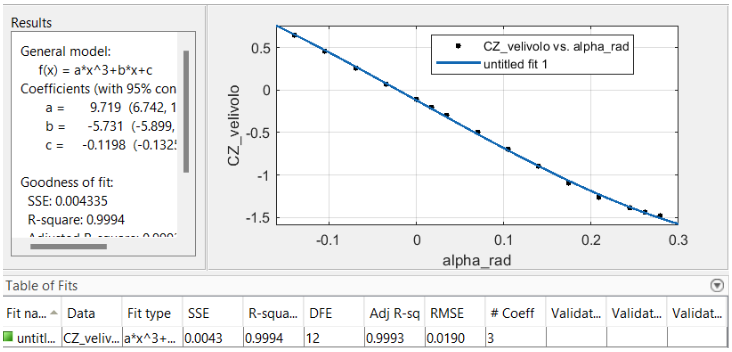

Figure 5.2.4: C_z interpolation

- $C_{z_{\delta_e}}$

This value was calculated by linearly interpolating the lift variation values as a function of the deflection of the elevator provided by DATCOM. In this case too, the increments corresponding to zero incidence were considered.

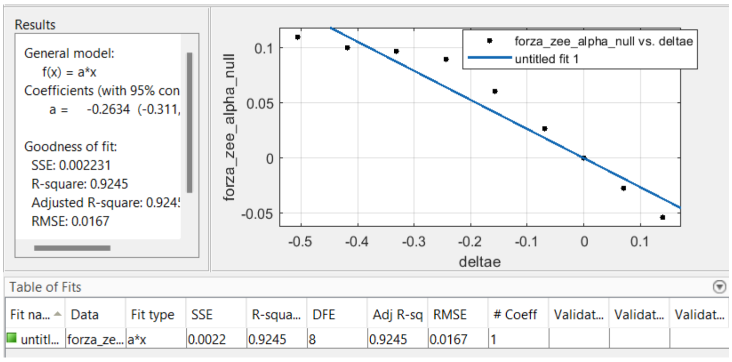

Figure 5.2.5: C_{z_q} interpolation

- C_{z_q}

It is taken as a first approximation as the value of C_{L_q} provided by DATCOM. In fact, it should be noted that $C_{z_q} = C_{L_q}$ only in case of zero incidence; therefore, for $\alpha \neq 0$ an approximation is in fact performed.

- $C_{z_{\delta_f}}, \; C_{z_{\alpha\delta_f}}$

These values were obtained analogously to the corresponding values along X. With the exception that equation 5.2.1 becomes:

$$C_x = C_L \cdot \cos(\alpha) + C_d \cdot \sin(\alpha) \tag{5.2.2}$$

This too must be changed in sign.

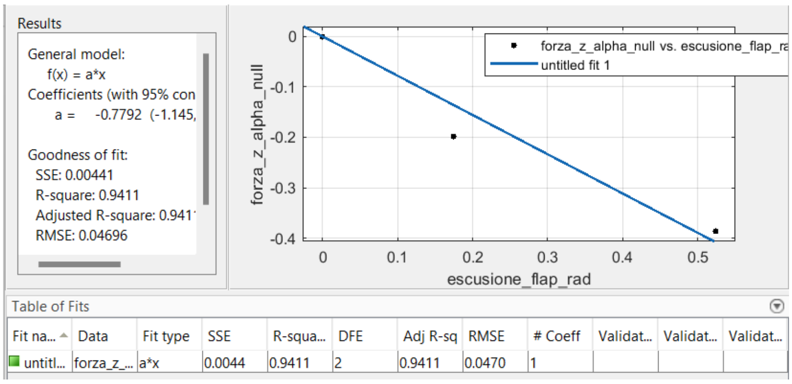

Figure 5.2.6: $C_{z_{\delta_f}}$ interpolation

It can be easily seen from the Figure 5.2.7 that the interpolation for $C_{z_{\alpha\delta_f}}$ is very marked, i.e. affected by the highest interpolation error.

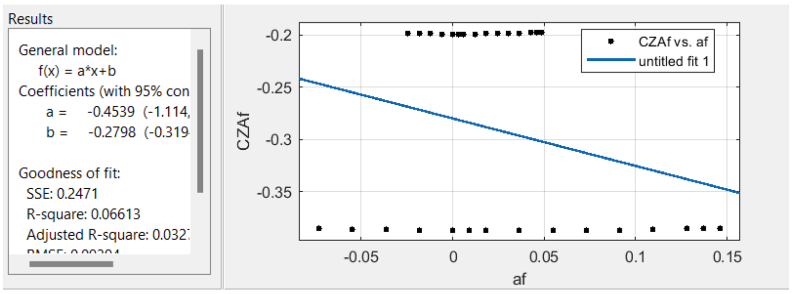

Figure 5.2.7: $C_{z_{\alpha\delta_f}}$ interpolation

- C_{m_q}

This value is directly provided by DATCOM.

- $C_{m_0}, \; C_{m_\alpha}, \; C_{m_{\alpha^2}}$

These values were calculated with the same method as the analogous coefficients along X axis, i.e. by interpolating the values of C_m provided by DATCOM.

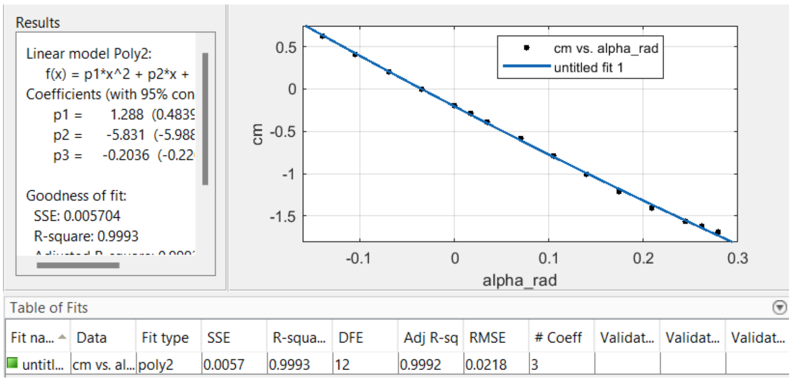

Figure 5.2.8: C_m interpolation

- $C_{m_{\delta_e}}$

This value was calculated by interpolating the variations provided by DATCOM of the C_m, due to the elevator deflections, as a function of the deflections themselves.

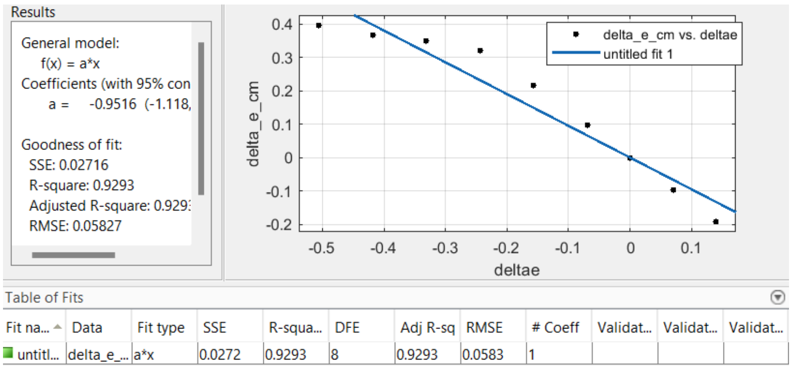

Figure 5.2.9: $C_{m_{\delta_e}}$ interpolation

- $C_{m_{\delta_f}}$

This value was calculated by interpolating the values provided by DATCOM as a function of the deflection of the flaps.

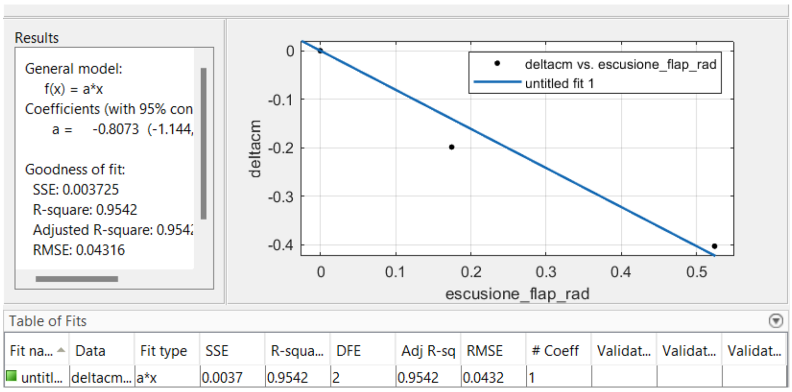

Figure 5.2.10: $C_{m_{\delta_f}}$ interpolation

- C_{y_β} e C_{n_β}

This values are directly provided by DATCOM.

- C_{l_β}, C_{l_p}, C_{n_p}, C_{l_r}, C_{n_r}

All these values are provided directly by DATCOM as a function of alpha. So an average value was simply taken from all those provided.

- $C_{l_{\delta_a}}$

This value was calculated by interpolating the variations provided by DATCOM of the C_l, due to the deflections of the aileron, as a function of the deflections of the aileron itself.

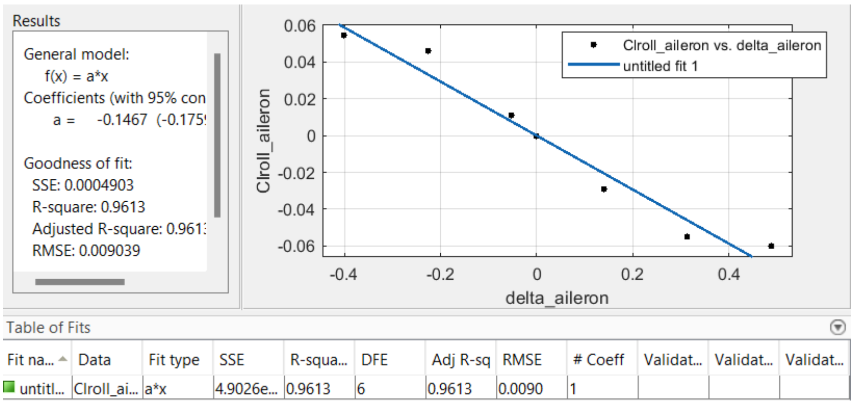

Figure 5.2.11: $C_{l_{\delta_a}}$ interpolation

- $C_{n_{\delta_a}}$

This value is provided by DATCOM as the incidence varies and the aileron deflection varies. To take into account only the contribution due to the aileron, the data for zero angle of attack was linearly interpolated.

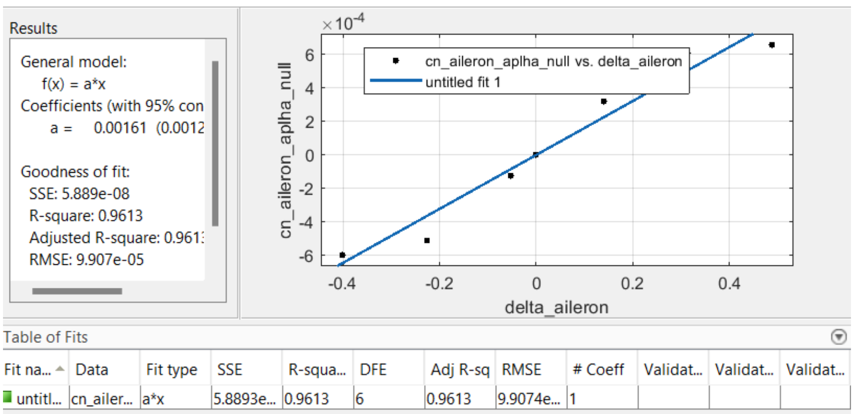

Figure 5.2.12: $C_{n_{\delta_a}}$ interpolation

5.3 Aerodynamics derivatives from Napolitano

As it can be seen in the previous section and as already mentioned, DATCOM does not allow to calculate many of the derivatives that were present in the Beaver model. This lack made it necessary to integrate the calculation methods just explained with further formulas, that made it possible to calculate at least some of the missing coefficients. These formulas are taken from Napolitano [7] and will be listed here. The following are the general formulas with the coefficients identified through the graphs. For a greater detail about the elements, explanations and calculations see the matlab sheet "derivate_ Napolitano.m".

- $C_{y_{\delta_r}}$

$$C_{y_{\delta_r}} = |(C_{L_{\alpha_v}})|\eta_v(S_v/S)\Delta KR \cdot \tau \tag{5.3.1}$$

Where the various terms are thus identified:

- $\Delta KR = 0.75$, graph in figure 5.3.2. $\eta_i = 0.18$ and $\eta_i = 1$ were used as abscissa, where the distance between the root and the tip of the rudder with respect to the X axis, to which b_v refers, was measured.
- $c_1 = 1.65$, graph in figure 5.3.3. $b_v/2 \cdot r1 = 1.475$ was used as abscissa, where $2 \cdot r1 = 0.8m$ was measured as shown in figure 5.3.3 and for b_v see section 2.5.
- $C_{L_{\alpha_v}}$ see the reference Napolitano [7, p. 65].
- $c_2 = 1$, graph in figure 5.3.5. $Z_H/b_v = 0.08$ was used as abscissa, where $Zh = |-0.1|\ m$ was measured as shown in 5.3.4. The quantity $x_{AC-H-V} = 0.3\ m$ derives from measurements made according to figure 5.3.5. The ratio $x_{AC-H-V}/b_v = 0.3$, as it is podssible to see, no curves are available for this value, probably because the measure of x_{AC-H-V} is affected by a small measurement error, or probably because the BS Prime has a very large vertical tail characterized by a much greater chord than the horizontal plane. Therefore, the closest curve has been chosen, i.e. the one to $x_{AC-H-V}/b_v = 0.5$.
- $Kh_v = 1.1$, graph in figure 5.3.1. $S_h/S_v = 1.65$ was used as the abscissa of the graph, for the values see section 2.5.
- $\tau = 0.27$, calculated as $\tau = \bar{c_r}/\bar{c_v}$. Where $\bar{c_r} = 0.320$ measured.
- $\eta_v = 0.9$, suggested on Napolitano as an average value [7]
- d = 1.31 m fuselage diameter at the wing, see figure 5.3.4.

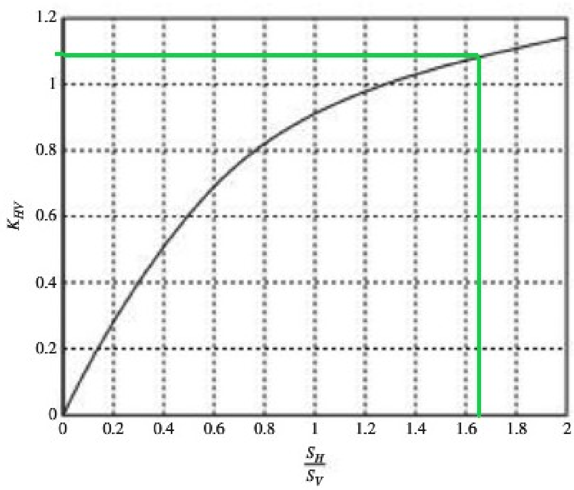

Figure 4.18 Factor for S_H/S_V Relative Size[5]

Figure 5.3.1: Coefficient: Kh_v

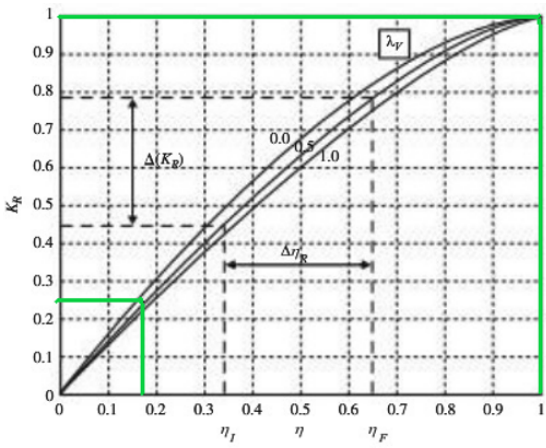

Figure 4.27 Span Factor between Rudder and Vertical Tail[5]

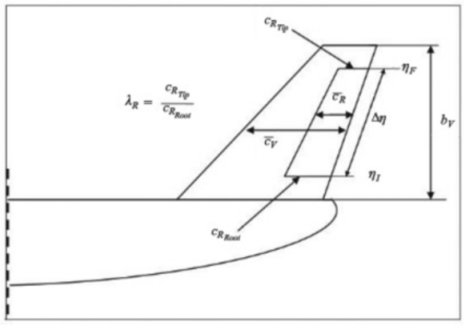

Figure 4.25 Geometric Parameters of the Rudder

Figure 5.3.2: Coefficient: ΔKR

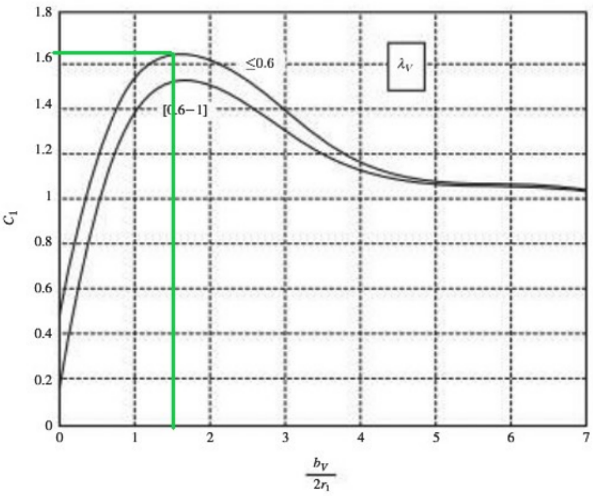

Figure 4.15 c_1 for the Evaluation of $AR_{V_{eff}}$[5]

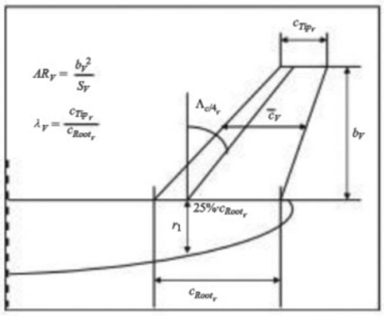

Figure 4.14 Geometric Parameters of the Vertical Tail

Figure 5.3.3: Coefficient: c1

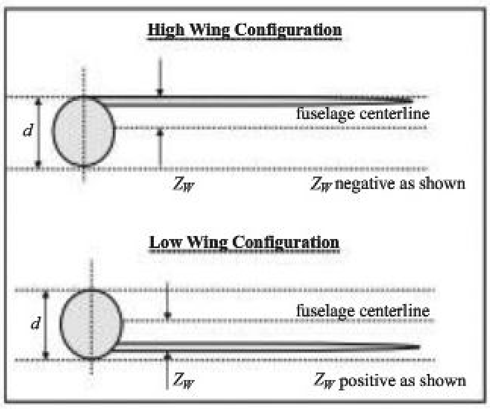

Figure 4.10 Geometric Parameters for Wing-Fuselage Integration (front view)

Figure 5.3.4: Coefficients: d and Z_w

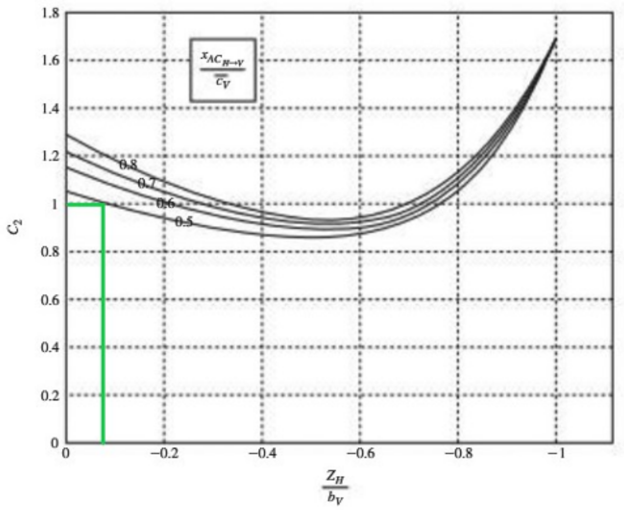

Figure 4.16 c_2 for the Evaluation of $AR_{V_{eff}}$[5]

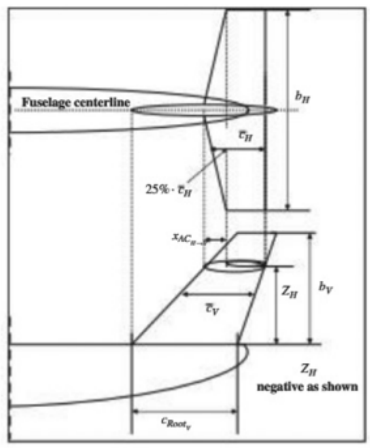

Figure 4.17 Geometric Parameters of the Interface between the Horizontal and Vertical Tail

Figure 5.3.5: Coefficient: c2

- $C_{l_{\delta_r}}$

$$C_{l_{\delta_r}} = C_{y_{\delta_r}} Z_r / b \tag{5.3.2}$$

With: $Zr = 0.485$ taken from datcom, corresponding to the Y of the vertical tail $\bar{c}$. The equation is thus defined, as it has already been taken into account that the angle between the stability and body axes is zero [7, p. 166].

- $C_{n_{\delta_r}}$

$$C_{n_{\delta_r}} = -C_{y_{\delta_r}} L_v / b \tag{5.3.3}$$

Where $L_v = 5.08$ m is the distance between datum and point of application of the force along Y on the rudder, due to the deflection of rudder. The equation is thus defined, as it has already been taken into account that the angle between the stability and body axes is zero [7, p. 176].

- $C_{y_{\dot{\beta}}}$, $C_{y_{\delta_a}}$

As reported in the Napolitano [7, pp. 147, 180] these derivatives for most aircraft are negligible, therefore they are considered null.

- C_{y_r}

$$C_{y_r} = -2 C_{y_{\beta_v}} X_v / b \tag{5.3.4}$$

Where $Xv = 4.6$ is distance between datum and vertical tail focus, $Zw = 0.35$ measured. The equation is thus defined, as it has already been taken into account that the angle between the stability and body axes is zero [7, p. 185]. Where $C_{y_{\beta_v}}$ is defined as:

$$C_{y_{\beta_v}} = -ky_v C_{L_{\alpha_v}} \eta_v (1 + \frac{d\sigma}{d\beta}) S_v / S;$$

- $\eta_v (1 + \frac{d\sigma}{d\beta})$ see Napolitano [7, p. 142].
- $C_{L_{\alpha_v}}$ see Napolitano [7, p. 65].
- ky_v= 0.78 see figure 5.3.6.
- S_v and S see section 2.5.

- $b_v/2 \cdot r1 = 1.475$ was used as abscissa, where $2 \cdot r1 = 0.8m$ was measured as shown in figure 5.3.3 and for b_v see section 2.5.

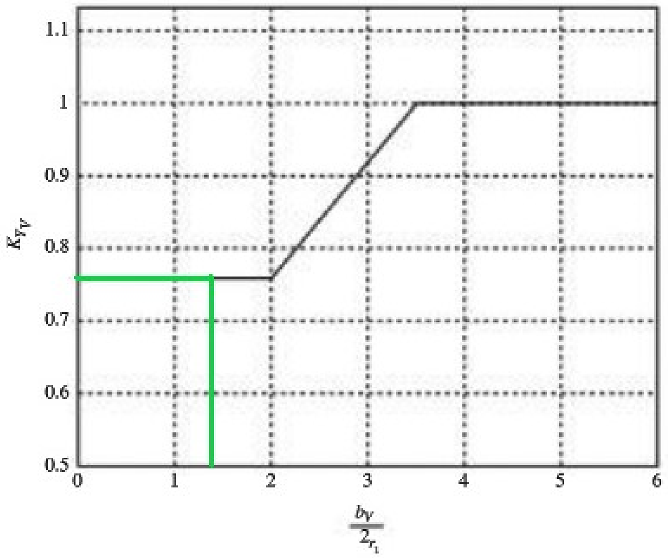

Figure 4.13 Empirical Factor for the Lateral Force at the Vertical Tail due to β

Figure 5.3.6: Coefficient: ky_v

The Napolitano derivatives are a function of Mach. As already been observed, the flight speed values led to Mach numbers that are less than 0.3. Therefore the assumption of incompressible fluid can be made, and the variations in the Mach number between 0 and 0.3 lead to small variations in the derivatives that can be negligible. In fact, at the maximum altitude only at V_{ne} (112.86 m/s) the Much is 0.34. Already at V_{no} (80.6 m/s) the Mach number is 0.25, since conventional flight speeds are already lower than V_{no} it is possible to suppose the incompressible fluid regime. The change in altitude between 0 m and the maximum altitude of 3049 m also produces negligible variations on the derivatives. It was not possible to calculate the totality of the coefficients used in the Beaver model, not having carried out tests in the wind tunnel. Therefore the derivatives, neither present in the literature nor provided by datcom, are:

$$C_{x_q},\ C_{x_{\delta_r\alpha}},\ C_{z\delta_e\beta^2},\ C_{m_{\beta^2}},\ C_{m_r},\ C_{y_0},\ C_{y_{\delta_r}},\ C_{l_0},\ C_{l_{\delta_a\alpha}},\ C_{n_0},\ C_{n_q},\ C_{n_{\beta^3}}$$

5.4 BS Prime aerodynamic model

Therefore the Aeromod block equations 4.2.5 and therefore the AM matrix, taking into account the above considerations on the aerodynamic derivatives, are adapted to the BS Prime in the following way:

$$
\begin{aligned}
C_x &= C_{x_0} + C_{x_\alpha}\alpha + C_{x_{\alpha^2}}\alpha^2 + C_{x_{\alpha^3}}\alpha^3 + +C_{x_{\delta_f}}\delta_f + C_{x_{\alpha\delta_f}}\alpha\delta_f \\
C_y &= C_{y_\beta} + C_{y_p}\frac{pb}{2V} + C_{y_r}\frac{rb}{2V} + C_{y_{\delta_a}}\delta_a + C_{y_{\delta_r}}\delta_r + C_{y_{\dot\beta}}\frac{\dot\beta b}{2V} \\
C_z &= C_{z_0} + C_{z_\alpha}\alpha + C_{z_{\alpha^3}}\alpha^3 + C_{z_q}\frac{\bar c q}{V} + C_{z_{\delta_e}}\delta_e + C_{z_{\delta_f}}\delta_f + C_{z_{\alpha\delta_f}}\alpha\delta_f \\
C_l &= C_{l_\beta}\beta + C_{l_p}\frac{bp}{2V} + C_{l_r}\frac{br}{2V} + C_{l_{\delta_a}}\delta_a + C_{l_{\delta_r}}\delta_r \\
C_m &= C_{m_0} + C_{m_\alpha}\alpha + C_{m_{\alpha^2}}\alpha^2 + C_{m_q}\frac{\bar c q}{V} + C_{m_{\delta_e}}\delta_e + C_{m_{\delta_f}}\delta_f \\
C_n &= C_{n_\beta}\beta + C_{n_p}\frac{bp}{2V} + C_{n_r}\frac{br}{2V} + C_{n_{\delta_a}}\delta_a + C_{n_{\delta_r}}\delta_r
\end{aligned}
\tag{5.4.1}
$$

Both the derivatives of DATCOM and those taken from Napolitano refer to the Datum reference system, so as to be able to contemplate the excursion of the center of gravity as the load conditions and fuel consumption vary. In order for the mathematical model to properly calculate the rotations with respect to the center of gravity, it was necessary to transport forces and moments with respect to the latter. Then the system of equations 6.3.9 in the Fmdims block becomes:

$$
\begin{aligned}
F_{x_a} &= \frac{1}{2}\rho V^2 S C_x \\
F_{y_a} &= \frac{1}{2}\rho V^2 S C_y \\
F_{z_a} &= \frac{1}{2}\rho V^2 S C_z \\
L_a &= \frac{1}{2}\rho V^2 S b C_l + (-z_g) F_{y_a} \\
M_a &= \frac{1}{2}\rho V^2 S \bar{c} C_m + (-x_g) F_{z_a} - (-z_g) F_{x_a} \\
N_a &= \frac{1}{2}\rho V^2 S b C_n - (-x_g) F_{y_a}
\end{aligned}
\tag{5.4.2}
$$

The new aerodynamic model of the Bs Prime on simulik is so characterized:

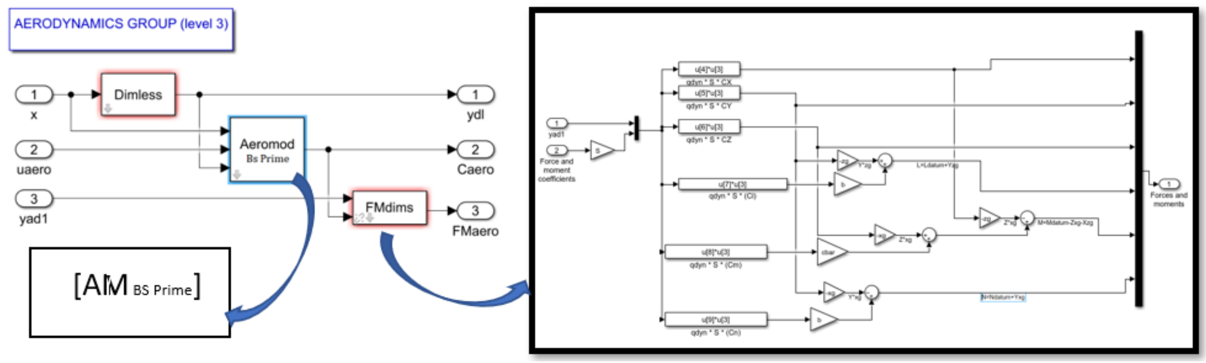

Figure 5.4.1: FDC BS Prime aerodynimics simulik grup

C_{x_0}	-0.0164	C_{z_0}	-0.1198	C_{m_0}	-0.2036
C_{x_α}	0.08598	C_{z_α}	-5.731	C_{m_α}	-5.831
$C_{x_{\alpha^2}}$	3.932	$C_{z_{\alpha^3}}$	9.719	$C_{m_{\alpha^2}}$	1.288
$C_{x_{\alpha^3}}$	-2.068	C_{z_q}	16.94	C_{m_q}	-31.35
$C_{x_{\delta_f}}$	-0.01334	$C_{z_{\delta_e}}$	-0.2634	$C_{m_{\delta_e}}$	-0.9516
$C_{x_{\alpha\delta_f}}$	0.4584	$C_{z_{\delta_f}}$	-0.7792	$C_{m_{\delta_f}}$	-0.8073
		$C_{z_{\alpha\delta_f}}$	-0.4539		
C_{y_β}	-0.4271	C_{l_β}	-0.03905	C_{n_β}	0.1316
C_{y_p}	-0.0739	C_{l_p}	-0.41547	C_{n_p}	-0.055921
C_{y_r}	0.3076	C_{l_r}	0.09279	C_{n_r}	-0.37982
$C_{y_{\delta_a}}$	0	$C_{l_{\delta_a}}$	-0.1467	$C_{n_{\delta_a}}$	0.00161
$C_{y_{\delta_r}}$	0.0534	$C_{l_{\delta_r}}$	0.0033	$C_{n_{\delta_r}}$	-0.0349
$C_{y_{\dot\beta}}$	0				

Table 5.4.1: Aerodynamic derivatives calculated for model 1

Chapter 6

Engine-Propeller model for Bs Prime

6.1 Introduction

As for the engine-propeller model, a new one had to be defined for the BS Prime. In fact, in the first place, the model used for the Beaver would not be usable for the BS Prime, due to the fact that it was modeled by resorting to the use of particular derivatives of the engine, that have been identified with appropriate tests [16]. Secondly, the Beaver engine power is defined by a mathematical function based on the interpolation of the Beaver engine data. In addition, the Blackshape Prime is equipped with a

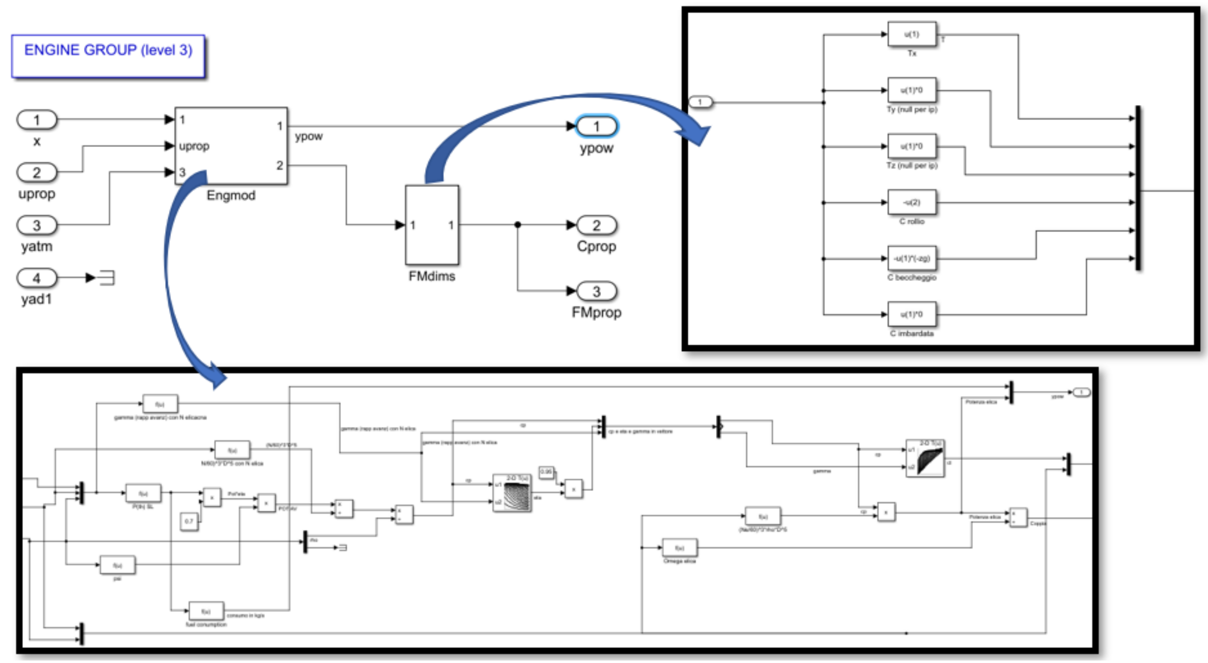

Figure 6.1.1: FDC BS Prime engine-propeller Simulink group

variable pitch propeller, which therefore allows the RPM be kept fixed. The pilot can act directly on the pitch of the propeller by varying the RPM, while acting on the throttle he changes the manifold pressure. All this led to the creation of a new model, Figure 6.1.1. This model receives as input data the manifold pressure (pz or MAP) and the number of revolutions (n or RPM), as well as all atmospheric data, and the state vector X. The model consists of two blocks: the first, Engmod, in which the engine and the propeller are modeled, giving the generated thrust and torque values (T and C) as output, and the ypow vector, which instead contains the fuel consumptio Fc and the power Π_e of the propeller; the second block, FMdims, in which the forces and moments due to thrust and torque are calculated.

6.2 Assumptions

1. The power is assumed to be a function of the manifold pressure and the RPM, i.e. $P = f(pz, n)$. The assumption is justified as on the BS Prime the pilot can act separately on the throttle and on the prop. These two elements therefore influence the power of the engine. Note that the model included in FDC, relating to the Beaver aircraft, also calculates the power in a similar way: $P \propto (a+pz)(b+n)$

Power setting	speed $[RPM]$	Power $[HP]$	MAP $["Hg]$	F.C. $[GAL/h]$
Take Off	5800	100	28	7
Max. Continuous	5500	90	27	6.5
75 %	5000	68	26	5
65 %	4800	60	26	4.5
55 %	4300	50	24	3.5

Table 6.2.1: Engine data sl. rounded

To derive the function that links the power to the MAP and to RPM, reference is made to the data in the engine Operators Manual [4], shown in Table 6.2.1. These values are used in matlab's "Curve fitting" toolbox for interpolation. Using a trial and error approach and taking a cue from

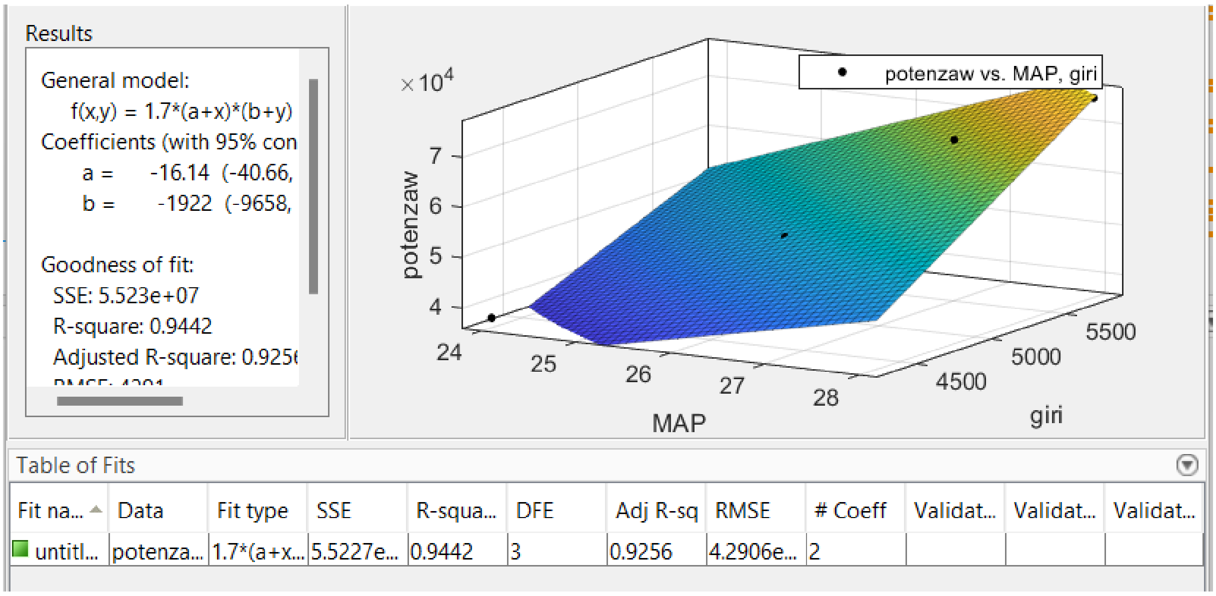

Figure 6.2.1: Power interpolation as a function of MAP and RPM

the general formula used by FDC, we have reached the following function for calculating the power at sea level, as the data refer to this condition.

$$\Pi_{sl} = 1.7(pz - 16.14) \cdot (n - 1922) \tag{6.2.1}$$

Where the power Π_{sl} is expressed in $Watt$, while the mainfold pressure pz is in $''Hg$ and the number of revolutions n in RPM. The equation 6.2.1 allows to obtain a realistic variation of the power: the power increases as the RPM increases at a fixed MAP; at a fixed number of RPM the power increases as the MAP increases.

2. The initial value of the propeller efficiency is 0.7;

3. The efficiency of the speed reducer is 0.95;

4. The torque can be calculated simply as Π/ω_e. (The manufacturer has not provided any curve or table for the torque coefficient);

5. The thrust pitch is zero, ie no component along Y_{datum} *and* Z_{datum}. The torque on the propeller is only around the X_{datum} axis.

6. Note that the percentage values shown in Figure 6.2.1 in the Power Setting column refer to the condition of Maximum Countinuous, that is, they are percentages with respect to this condition. So the maximum throttle occurs in take off condition, i.e. maximum power, and the others as follows:

Throttle [%]	100	90	68	60	50
Power $[HP]$	100	90	68	60	50
Power $[kW]$	75.4	67.9	51.3	45.3	37.7

Table 6.2.2: Percentage values of the throttle in relation to power

7. Inside the model the throttle value will be obtained as a function of the power. Note that both in the flight manual and in the engine data sheet, there are no data relating MAP, RPM and power for values below 50 HP. Therefore it is not possible to define these values for the idle conditions, i.e. zero throttle. In this condition, only RPM is obtained from the manual: 2500 RPM. If one proceeded to a simple interpolation of the power and throttle data with the data of the Table 6.2.2, an unrealistic condition would be obtained, whereby at zero throttle there would be zero engine power. To avoid this condition, it was assumed that in idle the engine generates a power of 2 kW. For the interpolation of the throttle as a function of the power, therefore, the Table 6.2.3 is used, which is equivalent to the Table 6.2.2 to which the idle condition has been added.

Throttle [%]	100	90	68	60	50	0
Power $[kW]$	75.4	67.9	51.3	45.3	37.7	2

Table 6.2.3: Percentage values of the throttle in relation to power

So the throttle percentage (ξ) is given by the following equation:

$$\xi = 0.00136 \cdot \Pi_{d_{sl}} - 2.033 \tag{6.2.2}$$

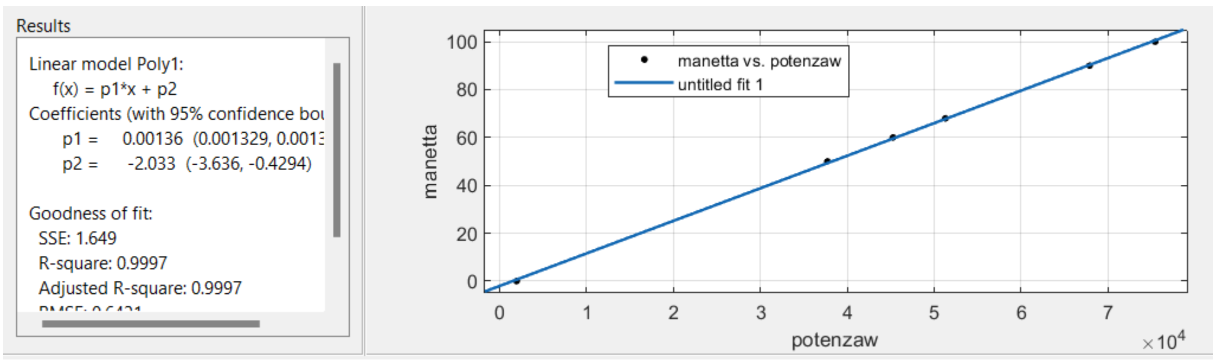

Figure 6.2.2: Interpolation of the throttle percentage as a function of the power in kW

8. As far as fuel consumption is concerned, it is a function of the engine power, so it is expressed by the equation that derives from the interpolation of consumption as a function of power. Also for the fuel consumption, a similar reasoning is followed for the throttle and the power: to avoid the unrealistic condition for which in idle the consumption is zero, a value taken from the engine manual is entered. [4, p. 59]. This data, however, is to be taken as a hypothesis since, although it refers to the number of RPM in idle, it does not refer to the MAP value that would occur in this condition. So taking the data of the table in Figure 6.2.1, adding the condition of idle and converting the GAL/h into L/s, we obtain the Table 6.2.4.

Power $[kW]$	75.4	67.9	51.3	45.3	37.7	2
Fuel c. $[L/s]$	0.0074	0.0068	0.0053	0.0047	0.0037	0.0011

Table 6.2.4: Percentage of throttle in relation to fuel consumption

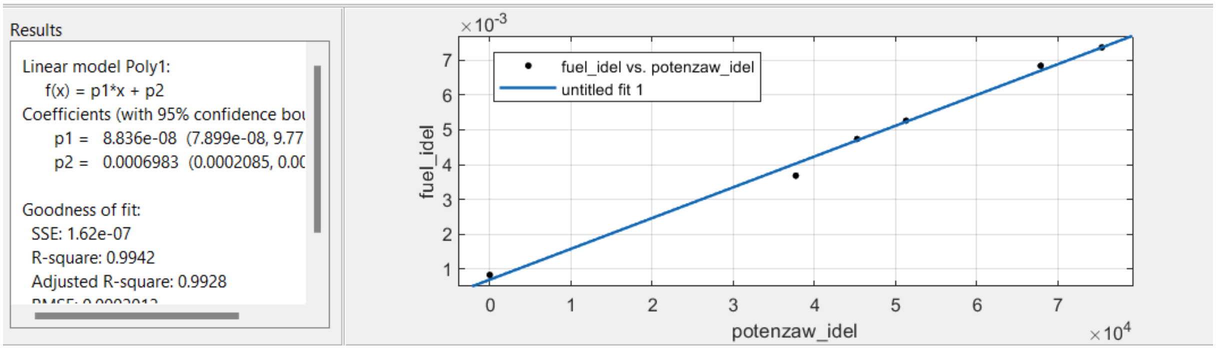

Figure 6.2.3: Fuel consumption interpolation as a function of power

By interpolating the data in the table, the following equation is obtained for fuel consumption L/s as a function of power in kW

$$fc = 8.836 \cdot 10^{-8} \Pi_{d_{sl}} + 6.983 \cdot 10^{-4} \tag{6.2.3}$$

9. Note that the constructor provides different C_t, C_p, η tables for different propeller RPM values: 1800, 2000, 2200 and 2386 RPM. In addition, for each number of RPM there are several different flight levels: flight level 50, 100, 150 and sea level. Some tests were carried out only on the simulink engine-propeller model, in order to evaluate how much the variation of these parameters affected the thrust and torque values. At a fixed number of engine RPM and throttle for each test, the tables for C_t, C_p, η were used, corresponding to different RPMs and different flight levels, as shown in the table 6.2.5.

PRM Propeller	Flight level	Thrust $[N]$	Torque $[Nm]$
2000 RPM	Sea Level	455.7	168.1
2000 RPM	150	452.3	167.5
2000 RPM	100	453.5	167.7
1800 RPM	100	452.6	167.6
2386 RPM	100	454.7	168.0

Table 6.2.5: Variation of thrust and torque with the variation of RPM propeller and flight level

As can be seen from the table, there is more data relating to 2000 RPM as it is an intermediate value between the maximum and minimum RPM of the propeller. From the tests it is therefore understood that it is possible to implement in the Simulink model only the tables relating to an intermediate value of both RPM and flight level, as the thrust and torque values vary minimally in the 6.2.5 table. The idea of identifying a single table for C_t, η is aimed at minimizing the computational load of the model and making it faster, as it will have to be inserted in FDC. In particular the maximum relative error:

$$er = \frac{thrust_{max} - thrust_{min}}{thrust_{max}} = 0.48\% \ < 1\%$$

Then the data for 2000 RPM and fl 50 were chosen, which is an intermediate value for the Bs Prime.

6.3 Model description

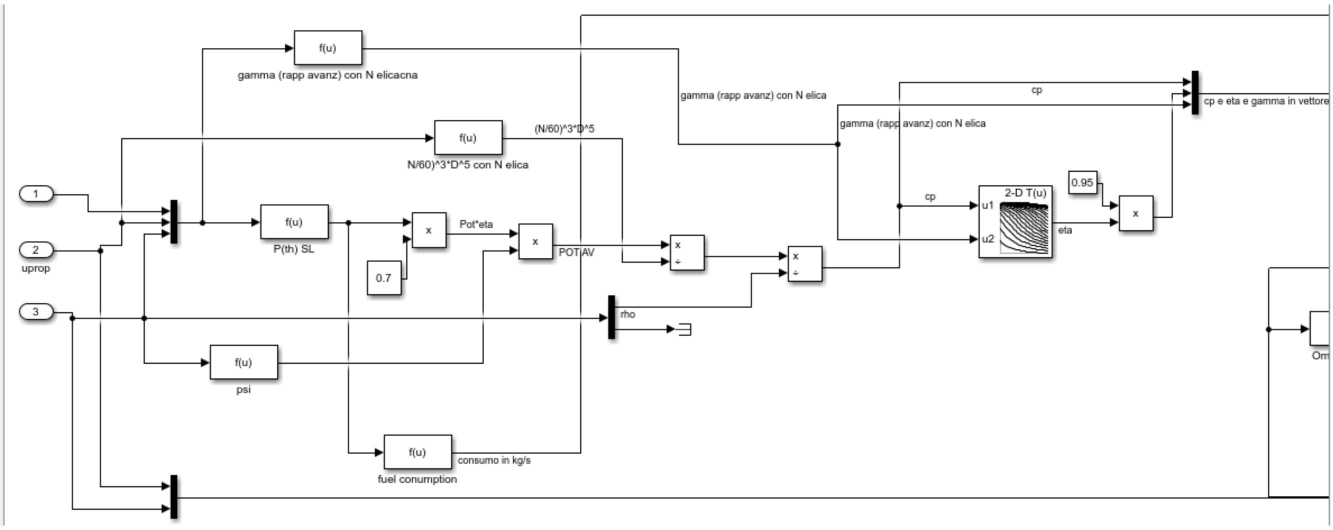

Figure 6.3.1: Engmod block model: power and cp computation

The model is contained in the block Engmod. The steps of the model are shown in Figure 6.3.1. The starting point are the input vectors, with which it is possible to implement the equation 6.2.1, thus obtaining the power at sea level. This value is then multiplied by the propeller efficiency, initially assumed to be 0.7 as per hypothesis. Hence the available power at sl. is:

$$\Pi_{d_{sl}} = \Pi_{sl} \cdot \eta \tag{6.3.1}$$

At the same time, the value Ψ is calculated, which takes into account the change in altitude:

$$\Psi(z) = \rho/\rho_{sl}\sqrt{T_{sl}/T} \tag{6.3.2}$$

Multiplying 6.3.2 by 6.3.1 it is possible to obtain the power available at altitude:

$$\Pi(z) = \Pi_{d_{sl}} \cdot \Psi(z) \tag{6.3.3}$$

Now it is possible to derive the relative power coefficient as:

$$C_p = \frac{\Pi}{(N_e/60)^3 D^5 \rho} \tag{6.3.4}$$

To calculate this coefficient, the formula proposed by the propeller manufacturer is used. [1]

[1] See appendix H. The formula shown is

$$C_p = \frac{\Pi \ 1000}{(N_e/60)^3 D^5 \rho}$$

In parallel it is possible to calculate the advancement ratio as:

$$\gamma = \frac{(V \cdot 3.6 \cdot 16.66)}{(DN_e)} \tag{6.3.5}$$

This equation is the one reported by the propeller manufacturer[2]. In particular, V is TAS in km/h, D is the diameter of the propeller in m and N_e is the number of revolutions of the propeller in RPM. Through these first steps the values of γ and Cp have been determined, necessary to enter the tables provided by the manufacturer to identify a new and more precise efficiency value. With the latter it is possible to recalculate the power and the relative Cp. Parallel to this process, the fuel consumption (fc) in L/s is calculated starting from the power available at sea level, using the interpolation equation 6.2.3. Having the fuel density [3], whose value can be set in the file "primo_run_fdc.m", it is possible to convert the fc from L/s to kg/s and obtain Fc:

$$Fc = fc \cdot \rho_{fuel} \cdot 10^{-3} \tag{6.3.6}$$

With the new power coefficient and the advancement ratio γ calculated previously, it is possible to use the table relating to the Thrust Coefficient C_t.

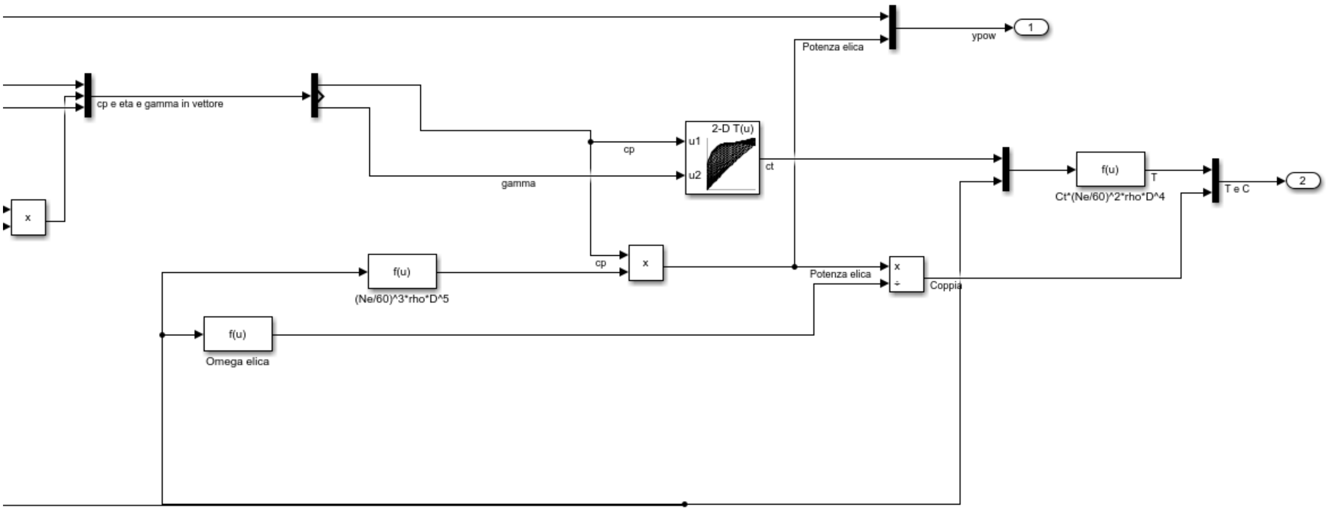

Figure 6.3.2: Engmod block model: C_t, T and C computation

Also in this case there are several tables for different values of flight level and RPM, but as has already been illustrated in Table 6.2.5 it is not necessary to implement all of them. Once the Ct is obtained, the thrust is obtained:

$$T = C_t(N_e/60)^2 \rho D^4 \tag{6.3.7}$$

This equation is obtained by inverting the one proposed by the propeller manufacturer.[4] As for the torque, since the manufacturer does not provide any table or graph for the torque coefficient C_q, the only way to calculate it is as power divided by the angular velocity of the propeller:

$$C = \Pi/\omega_e \tag{6.3.8}$$

as defined in the assumptions. Once the values of T and C, that is thrust and torque, have been obtained, we proceed to break them down along the Body axes. These operations are contained in the block FMdims. For assumption 5 it follows that $T = T_x$, and that the torque on the propeller will only give the rolling

observe that the manufacturer uses the power in kW which is why the value 1000 appears in the numerator, instead the power used by us is expressed only in W

[2]See appendix H

[3]See section 3.3.8

[4]See appendix H

moment. The pitching moment component due to thrust was then inserted as: $T \cdot z_g$ with z_g value of the coordinate of the center of gravity with respect to the datum.

The equations of forces and moments used for the FMdims block due to propeller are defined as follows:

$$\begin{aligned} F_{x_p} &= T \\ F_{y_p} &= 0 \\ F_{z_p} &= 0 \\ L_p &= C \\ M_p &= z_g \cdot T \\ N_p &= 0 \end{aligned} \tag{6.3.9}$$

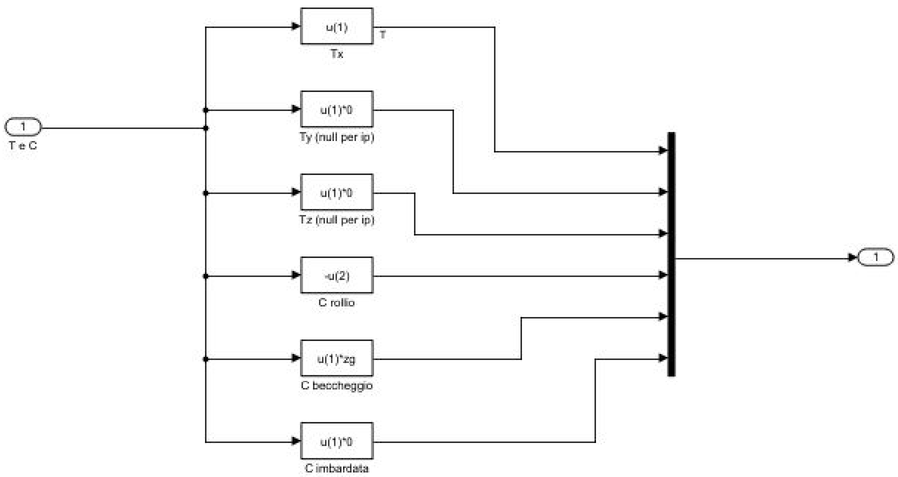

Figure 6.3.3: FMdims block: T e and C decomposition on body axis

Chapter 7

Conclusion and future Development

As mentioned, the discussion just concluded constitutes only the first part of the project, which will be continued in the thesis "Validation of a mathematical model for the BS Prime". Since this thesis will deal with the part on validation, which is therefore absent in this discussion, it is difficult to draw conclusions on the validity of the newly created model. However, due to the inevitable approximations committed during the development of the mathematical model, it is reasonable to expect that this model will at least partially deviate from the real results. It will therefore be necessary to make corrections in an iterative way to get closer to the expected results.

Appendix A

GUIDE for DATCOM

Below is a practical guide for the correct use of DATCOM for Windows. It is possible to download DATCOM for free at the following link:

`http://www.holycows.net/datcom/buy.html`

Be sure to save the "Datcom" folder in the following file path: This PC-> WINDOWS (C:) -> Users -> "username". Inside you will find the "bin", "doc", and "examples" folders. After installation it is necessary to perform the steps specified on the page (in the "FREE VERSIONS" section, just below the download links). If you want to create an input file it is advisable to use the Notepad ++ program. To run a file follow this sequence of operations:
- save it in the "bin" folder with the extension ".dcm";
- open the command prompt (from the windows search bar) by clicking on "run as administrator";
- inside the prompt, change the directory by typing "cd 'copypaste of the file path up to bin'", for example: begin verbatim cd C: Users ..... Datcom bin ") end verbatim - type "datcom.bat" followed by the name you gave your input file, including the .dcm extension
For others informations see: De Marco and Coiro [3] and Williams and Vukelich [19]

Appendix B

Datcom input file

```
*   File : BSPRIME.dat
DIM FT
DERIV RAD
DAMP
PART
TRIM
*DUMP ALL
WT=1322.77,

$FLTCON WT = 1322.77, LOOP=2.0, RNNUB = 1000000.0,
NMACH=1.0, MACH(1)=0.206,
NALT=1.0, ALT(1)=4000.0,
NALPHA=15.0,
ALSCHD(1)= -8.0, -6.0, -4.0, -2.0,
0.0, 1.0, 2.0, 4.0, 6.0, 8.0, 10.0, 12.0, 14.0,15.0,16.0,
STMACH=0.6, TSMACH=1.4, TR=1.0$

$OPTINS SREF=102.36, CBARR=4.11, BLREF=25.4531, ROUGFC=0.16E-3$

$SYNTHS XCG=4.7458, ZCG=-0.0,
XW=5.9531,  ZW=-1.148,    ALIW=0.0,
XH=19.0507, ZH=0.1916,    ALIH=-2.0,
XV=17.313,  ZV=0.0,
XVF=17.313, ZVF=-0.656,
SCALE=1.0, VERTUP=.TRUE.$

$BODY NX=9.0,
X(1)=0.0,1.1349,2.4131,4.7458,6.1415,9.5479,12.3490,16.5023,23.2829,
R(1)=0.0,0.5735,1.1541,1.2304,1.3343,1.3174,1.0404,0.7102,0.3001,
ZU(1)=0.0,0.5119,0.7985,0.9417,1.2491,2.2922,2.0025,1.3217,0.4829,
ZL(1)=0.0,-0.5119,-1.1365,-1.6050,-1.7852,-2.0153,-1.6838,-0.8854,0.0,
BNOSE=1.0, BLN=4.7458,
BTAIL=1.0, BLA=0.0,
ITYPE=1.0, METHOD=1.0$

$WGPLNF CHRDR=6.2283,   CHRDTP=2.1093, CHRDBP=4.5,
SSPN=12.956,   SSPNE=11.4824, SSPNOP=9.0891,
SAVSI=10.0, SAVSO=3.0,
CHSTAT=0.25, TWISTA=-1.0,
DHDADI=4.0,
TYPE=1.0$

NACA-W-4-2408
SAVE
```

```
$SYMFLP FTYPE=1.0,    NDELTA=3.0,
DELTA(1)=0.0,10.0,30.0,
PHETE=0.052, PHETEP=0.0391,
CHRDFI=1.222,   CHRDFO=0.76,
SPANFI=1.66,  SPANFO=7.7,
NTYPE=1.0$
CASEID FLAPS: BS PRIME Aircraft
NEXT CASE

DELTAL(1)=-28.0,-18.0,-8.0,0.0,3.0,13.0,23.0,
DELTAR(1)=28.0,18.0,8.0,0.0,-3.0,-13.0,-23.0,
SPANFI=7.7, SPANFO=12.217,
PHETE=0.05228,
CHRDFI=0.94, CHRDFO=0.57$
CASEID AILERONS: BS PRIME Aircraft
SAVE
NEXT CASE

NACA-H-5-63-010
$HTPLNF CHRDR=3.178, CHRDTP=1.411,
SSPN=4.95, SSPNE=4.552,
SAVSI=10.0,
CHSTAT=0.25, TWISTA=-1.0,
DHDADI=-3.0,
TYPE=1.0$

$VTPLNF CHRDTP=1.672, SSPNE=3.21,  SSPN=3.871, CHRDR=5.42,
SAVSI=55.0,  CHSTAT=0.25, TYPE=1.0$

$VFPLNF CHRDR=4.075, CHRDTP=4.075, CHSTAT=0.5, DHDADO=0.0,
SAVSI=-26.0, SSPN=0.73472,  SSPNE=0.36736, TYPE=1.0$

$SYMFLP FTYPE=1.0,
NDELTA=9.0, DELTA(1)=-29.0,-24.0,-19.0,-14.0,-9.0,-4.0,0.0,4.0,8.0,
PHETE=0.0522, PHETEP=0.0523,
CHRDFI=1.21,  CHRDFO=0.69,
SPANFI=1.09,   SPANFO=9.0,
CB=0.40,      TC=0.061,         NTYPE=1.0$

NACA-V-5-63-010

CASEID TOTAL: nuovo1 II Model BSPRIME Aircraft
```

Appendix C

Msss distribution: Raymer and BS155

```
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%                Suddivisione masse Raymer
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%suddivisone masse con coefficienti presi dal raymer a pag 599. [kg]
%NB peso e considerazizone all'elica
wtotale=600; %[kg]  PESO DA MANUALE
wala= 0.16*wtotale;
wtail_o=0.043*wtotale;
wtail_v=0.033*wtotale;
motore=60;
propeller=5;
carrello=0.055*wtotale;

pilota1=90;
pilota2=90;
pilota=pilota1+pilota2;
bagalio= 27;
fuel=43; %nb i tank contengono al max 31l ciascuno


body=wtotale-(wala+wtail_o+ wtail_v+ motore+ carrello ...
+propeller+ pilota+ bagalio+ fuel);

W_vuoto=wtotale-(pilota+ bagalio+ fuel);

%%
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%                 suddivisione masse con BS115
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%

%percentuali rispetto basic empty w.
%ala 12.4 percento, tail verticale 3.37 perc, tail H 3.37 perc, body 39,88
%perc, motore 27.97 perc, propeller 4.63 perc, nouse landig gera 2.4 perc,
%min landig gera 6.06 perc

wempty = 390; %[kg] PESO A VUOTO PRESO DA MANUALE
wala1 = 55;%0.124*wempty
wtail_o1 = 15;%0.0337*wempty
wtail_v1 = 13;%0.0337*wempty
%motore1=0.2798*wempty
motore1 = 67.7;
propeller1 = 18;%0.0463*wempty
carrello_main1 = 24;%0.0606*wempty
carrello_nose1 = 10;%0.0240*wempty
```

```
body1 = 31.7680+155.5320;%0.3988*wempty

somma = wala1+wtail_o1+wtail_v1+motore1+propeller1+ ...
carrello_main1+carrello_nose1+body1;
```

Appendix D

"primo_run.fdc.m"

MATLAB file "primo_run.m". Not all the eta and ct coefficients have been entered, for all the values see either the manufacturer's tables or the matlab file. A few data were omitted since they don't belong to us and we were not allowed to publish them.

```
%%%%%%%%%%%%%  PIANO DI CARICO DATI MODIFICABILI
pilota1 = 85;
pilota2 = 85;
pilota=pilota1+pilota2;
bagaglio = 0;
fuel = 40;

densita_fuel = 737.4;% [kg/m3]valor medio ma da cambiare in base al tipo
volume_massimo_serbatoi = 62;%[litri]
massa_massima_fuel = densita_fuel*volume_massimo_serbatoi/1000;
%nb i tank contengono al max 31 litri ciascuno
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%

%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%                      DATI NON MODIFICABILI
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%


%%%%%%%%%%%%%%%%%%%%%% Velivolo%%%%%%%%%%%%%%%%%%%
wempty = 390; %[kg] PESO A VUOTO PRESO DA MANUALE
wala1 = 55;
wtail_o1 = 15;%
wtail_v1 = 13;
motore1 = 67.7;
propeller1 = 18;
carrello_main1 = 24;
carrello_nose1 = 10;
body1 = 31.7680+155.5320;
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%

%%%%%%%%%% PROPELLERE_ENGIN%%%%%%%%%%
riduzione=2.43; % rapporto di riduzione del riduttore
eta = 0.7; %rendimento eleica insiale
D = 1.75; %diametro dell'elica in metri

% eta e ct per 2000 gir/min a f150(1524 m) valore intermendio

%eta
A_2000_f150_eta=[ 0 0.10000 0.20000 0.30000 0.40000 0.50000 0.60000 0.70000
    0.80000 0.90000 ....];
%corretta questa sotto senza prima riga gamma e prima colonna di cp
```

```
A2000_fl100_eta=A_2000_fl50_eta([2:end],[2:end]);

%ct
A_2000_fl50_ct=[0 0.10000 0.20000 0.30000 0.40000 0.50000 0.60000 0.70000
    0.80000 0.90000 ....];
%corretta queat sotto senza prima riga gamma e prima colonna di cp
A2000_fl100_ct=A_2000_fl50_ct([2:end],[2:end]);

%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%                        Calcolo baricento
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%



%%%%%%%%%%%%%%%%%%%%%%% Massa elementi %%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
wala = wala1;
wtail_o = wtail_o1;
wtail_v = wtail_v1;
motore = motore1;
propeller = propeller1;
carrello_main = carrello_main1;
carrello_nose = carrello_nose1;
body = body1;

%massimo peso decollo
MTOW = wempty+pilota1+pilota2+bagaglio+fuel;


%%%%%%%%%%%%%%%%%%%%%%%%%  Posizioni lungo x y z %%%%%%%%%%%%%%%%%%%%%%%%


%posizioni lungo x a partire dalla punta del muso [m]
x_propeller = 0;
x_motore = 0.91;
x_tailo = 6.35;
x_carrello_nose = 0.950;%da manuale di volo
x_carrello_main = 2.688;%da manuale di volo
x_fuel = 2.2; %da manuale di volo
x_tailv = 6.35;

%x baricentro fusoliera approssimata come due coni
x_troncomuso = 2.9116*3/4;
x_troncocoda = ((6.7936-2.9116)/4)+2.9116;
x_body_troncomuso = (x_troncomuso*body*2/3+x_troncocoda*body*1/3)/(body);

%baricentro fusoliera usando similitudine BS115
x_bodi115=(75.5*1003+3.2*(-1001)+1.3*(-909)+33.3*2122+5*(-1300)+ ...
4.2*(-1350)+12.7*(-260)+7.5*(-450)+1.3*(150)+13.5*(534)+ ...
10.7*(-1046)+4.7*(365)+18.5*(273)+11*(45)+2.5*(507)+ ...
2.5*(1282))/207+800;

x_bodyBSP=(x_bodi115*7437/7178)/1000;%rispetto al datum

%posizioni lungo x a partire dal datum [m]
x_datum = 1.45;
x_propeller1 = x_datum;
x_motore1 = -(0.91-x_datum);
x_ala1 = -(0.68+0.35*1.252); %pos del terzo della m.a.c. rispetto al datum
x_tailo1 = -(6.35-x_datum);
x_tailv1 = -(6.25-x_datum);%misurate
x_pilota11 = -1; % manuale
```

```
x_pilota21 = -1.8;% manuale
x_carrello_main1 = -(x_carrello_main-x_datum);
x_carrello_nose1 = -(x_carrello_nose-x_datum);
x_fuel1 = -(2.2-x_datum); %manuale
x_body1 = -x_bodyBSP;
x_bagaglio1 = -2.25;%manuale di volo



%posizione lungo y [m]

%wing
hw = 7.76/2;
br = 1.9;% corda al root
bt = 0.67;%corda al tip
bp = 1.3720;% corda al break point
he = 2.7029;% distanza outboard panel

y_ala1e = -he/3*(bp+2*bt)/(bp+bt)-(hw-he);% semiala sx negativa outboard
    panel
y_ala1i = -(hw-he)/3*(br+2*bp)/(br+bp);% semiala sx negativa inboard panel

Si = (br+bp)*(hw-he)/2;%superficie interna trapezio ala
Se = (bt+bp)*(he)/2;%superficie esterna trapezio ala

y_semiala1 = (y_ala1e*Se+y_ala1i*Si)/(Si+Se); %semiala sisnistra negativo
y_semiala2 = -y_semiala1;%semiala destra positivo

%tank fuel
y_fuel1 = -0.732;% tank  sinistro negativa
y_fuel2 = 0.732;% tank  destro positiva

%tail
h = 1.51;
B = 0.97;
b = 0.43;
y_tailo1 = -h/3*(B+2*b)/(B+b);%sinistra negativa
y_tailo2 = h/3*(B+2*b)/(B+b);%destra positiva
y_bagaglio = 0;
y_motore = 0;
y_pilota1 = 0;
y_pilota2 = 0;
y_carrello = 0;
y_body = 0;
y_tailv = 0;
y_propeller = 0;

%posizione lungo z tutto in metri concorde datum verso l'alto
z_ala = -(-0.35); %posizione della radice dell'ala, misurata dal disegno
z_semiala = -(-0.35+y_semiala2*tand(4)); %tenendo conto di angolo diedro
z_fuel = -(-0.30);
z_motore = -(-0.10);
z_pilota1 = 0;
z_pilota2 = -0.1;
z_tailo = -(0.113-y_tailo2*tand(3.5));
z_tailv = -0.7;
z_carrello_nose = -(-0.30);
z_carrello_main = -(-0.28);
z_body = -(-0.1);
z_propeller = 0;
z_bagaglio = -(-0.1);
```

```
%%%%%%%%%%%%%%%%%%%%%%%%% CALCOLO BARICENTRO %%%%%%%%%%%%%%%%%%%%%%%%%%%

% calcolo baricentro sfruttando manuale di volo
fuel_momentum = fuel*(-);%fuel in kg
pilota1_momentum = pilota1*-;
pilota2_momentum = pilota2*(-);
aircraft_momemntum = wempty*(-);
bagaglio_momentum = bagaglio*(-);

somma_momentum=(fuel_momentum+pilota1_momentum +pilota2_momentum+...
    aircraft_momemntum+bagaglio_momentum);

% BARICENTRO LUnGO X calcolo usando la formula del manuale
x_g_percentuale_mac=(((pilota1_momentum+pilota2_momentum+ ...
fuel_momentum+bagaglio_momentum+ ...
aircraft_momemntum)/MTOW)-0.68)/1.252*100;

x_g_mac_datum= (x_g_percentuale_mac*1252/100+680)/1000;
%[m] posizione di xg a partire da x_g_percentuale_mac a cui si aggiunge
%la distanza dal datum della corda media a.

xg = -(x_g_mac_datum); %il segno meno dovuto al sistema di riferimento,
%in questo caso centrato nel datum e con asse x rivolto in avanti



% Baricentro Zg rispetto al datum
zg = (z_motore*motore+z_semiala*wala+z_tailo*wtail_o+ ...
z_tailv*wtail_v +z_pilota1*pilota1+z_pilota2*pilota2+ ...
z_carrello_main*carrello_main+z_carrello_nose*carrello_nose+ ...
z_fuel*fuel+z_body*body+z_propeller*propeller+ ...
z_bagaglio*bagaglio)/MTOW;

%posizioni rispetto al baricentro

x_motoreg = -(xg-x_motore1);
x_alag = -(xg-x_ala1) ;
x_tailog = -(xg-x_tailo1);
x_tailvg = -(xg-x_tailv1) ;
x_pilota1g = -(xg-x_pilota11);
x_pilota2g = -(xg-x_pilota21);
x_carrello_maing = -(xg-x_carrello_main1);
x_carrello_noseg = -(xg-x_carrello_nose1);
x_fuelg = -(xg-x_fuel1);
x_bodyg = -(xg-x_body1);
x_propellerg = -(xg-x_propeller1);
x_bagagliog = -(xg-x_bagaglio1);

z_semialag = -(zg-z_semiala);
z_fuelg = -(zg-z_fuel) ;
z_motoreg = -(zg-z_motore);
z_pilota1g = -(zg-z_pilota1);
z_pilota2g = -(zg-z_pilota2) ;
z_tailog = -(zg-z_tailo);
z_tailvg = -(zg-z_tailv);
z_carrello_maing = -(zg-z_carrello_main) ;
z_carrello_noseg = -(zg-z_carrello_nose);
z_bodyg = -(zg-z_body);
z_propellerg = -(zg-z_propeller);
z_bagagliog = -(zg-z_bagaglio);
```

```
%%%%%%%%%%%%%%%%%%%%%%%%% Momenti di inerzia %%%%%%%%%%%%%%%%%%%%%%%%%%%%%%


%Momenti di inerzia BARICENTRICI

Ixx = (motore*(z_motoreg^2+y_motore^2) + ...
wala*(z_semialag^2+y_semiala2^2) + ...
fuel*(z_fuelg^2+y_fuel2^2) + ...
pilota1*(z_pilota1g^2+y_pilota1^2) + ...
pilota2*(z_pilota2g^2+y_pilota2^2) + ...
wtail_o*(y_tailo2^2+z_tailog^2) +...
wtail_v*(z_tailvg^2+y_tailv^2) + ...
carrello_nose*(y_carrello^2+z_carrello_noseg^2)   +   ...
carrello_main*(y_carrello^2+z_carrello_maing^2)   +  ...
body*(z_bodyg^2+y_body^2) + ...
propeller*(y_propeller^2+z_propellerg^2) + ...
bagaglio*(z_bagagliog^2+y_bagaglio^2));

Iyy = (motore*(z_motoreg^2+x_motoreg^2) + ...
wala*(z_semialag^2+x_alag^2)     + ...
fuel*(z_fuelg^2+x_fuelg^2) + ...
pilota1*(z_pilota1g^2+x_pilota1g^2) + ...
pilota2*(z_pilota2g^2+x_pilota2g^2) + ...
wtail_o*(x_tailog^2+z_tailog^2 )+ ...
wtail_v*(x_tailvg^2+z_tailvg^2) + ...
carrello_nose*(x_carrello_noseg^2+z_carrello_noseg^2) + ...
carrello_main*(x_carrello_maing^2+z_carrello_maing^2) + ...
body*(z_bodyg^2+x_bodyg^2) + ...
propeller*(x_propellerg^2+z_propellerg^2) + ...
bagaglio*(z_bagagliog^2+x_bagagliog^2));

Izz = (motore*(y_motore^2+x_motoreg^2) + ...
wala*(y_semiala1^2+x_alag^2)     + ...
fuel*(y_fuel1^2+x_fuelg^2) + ...
pilota1*(y_pilota1^2+x_pilota1g^2)  + ...
pilota2*(y_pilota2^2+x_pilota2g^2) + ...
wtail_o*(x_tailog^2+y_tailo1^2) + ...
wtail_v*(x_tailvg^2+y_tailv^2) + ...
carrello_nose*(x_carrello_noseg^2+y_carrello^2)   +    ...
carrello_main*(x_carrello_maing^2+y_carrello^2)   +  ...
body*(y_body^2+x_bodyg^2) + ...
propeller*(x_propellerg^2+y_propeller^2) + ...
bagaglio*(x_bagagliog^2+y_bagaglio^2));

Jxz = -(motore*z_motoreg*x_motoreg     + ...
wala*z_semialag*x_alag      +    ...
fuel*z_fuelg*x_fuelg    +   ...
pilota1*z_pilota1g*x_pilota1g     +  ...
pilota2*z_pilota2g*x_pilota2g    +  ...
wtail_o*x_tailog*z_tailog   +   ...
wtail_v*x_tailvg*z_tailvg    +  ...
carrello_nose*x_carrello_noseg*z_carrello_noseg   +  ...
carrello_main*x_carrello_maing*z_carrello_maing   +  ...
body*z_bodyg*x_bodyg    + ...
propeller*x_propellerg*z_propellerg    + ...
bagaglio*x_bagagliog*z_bagagliog);

%inerzie aggiuntive rispetto ai singoli baricentri dove ali e tail visti
%come aste e cilindo lungo y, body come cilindro, motore cilindro.
```

```
%ali
Ixx_wing=(1/12)*wala*(hw)^2;% vista come asta
Izz_wing=Ixx_wing;
Iyy_wing=(1/2)*wala*(0.626)^2; %visto come cilindro

%tail orizzontale
Ixx_tailo=(1/12)*wtail_o*(1.5091)^2;
Izz_tailo=Ixx_tailo;
Iyy_tailo=(1/2)*wtail_o*(0.35)^2; % media tra c tip e c root

%tail verticale
Ixx_tailv=(1/12)*wtail_o*(0.9787)^2; %visto come asta
Iyy_tailv=Ixx_tailv;
Izz_tailv=(1/2)*wtail_o*(0.55)^2;% visto come cilindro

%body visto come due coni
Ixx_body=(3/10)*body*2/3*(0.65)^2+(3/10)*body*1/3*(0.65)^2;
Iyy_body=(1/12)*body*(7.178)^2;
Izz_body=Iyy_body;

Ixx_motore=(1/2)*motore*0.3506^2;
Iyy_motore=(1/12)*motore*(0.7112)^2;
Izz_motore=Iyy_motore;

Ixx_tot=Ixx+Ixx_wing+Ixx_tailv+Ixx_tailo+Ixx_body+Ixx_motore;
Iyy_tot=Iyy+Iyy_wing+Iyy_tailv+Iyy_tailo+Iyy_body+Iyy_motore;
Izz_tot=Izz+Izz_wing+Izz_tailv+Izz_tailo+Izz_body+Izz_motore;

%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
disp('=======================');
disp(' Controls');
disp('=======================');
disp(' ');
% controlli
curiosity=0;
lander=0;
cassini=0;
% controllo sul MTOW
if(MTOW>600) fprintf ('-> Stop MTOW>600 kg, MTOW=%f \n',MTOW)
cassini=1;
end

%controllo sul fuel
if(massa_massima_fuel<fuel)
curiosity=1;
fprintf ('stop to much fuel, fuell> max fuel\n ')
end
if(fuel==0)
cassini=1;
fprintf ('               Stop, zero fuel  \n ')
end
if cassini==1
disp('    STOP, NOT READY for trim condition, see errors ');
disp(' ')
end
% se troppo leggero compare un warning
if(MTOW<441)
curiosity=3;
fprintf ('               Warning maybe to light\n ')
disp('          Check again mass distribution ');
disp(' ')
```

```
end

%controllo su masse minore di 460 kg (vedere xg envelope)
if(MTOW<=460 && massa_massima_fuel>fuel)
fprintf ('-> TOW = %.1f Kg ok,  FUEL =%.1f Kg ok\n',MTOW,fuel)
fprintf('\n-> Gravity center coordinates given from datum:\n')
fprintf('\n          Xg=%.3f m  Yg=0 m   Zg=%.3f m\n',xg, zg)
fprintf('\n-> Xg given as a percentage of main aerodynamic corde: %.2f % \n',
    x_g_percentuale_mac)

% controllo sull'xg che deve stare nel xg envelope
if(x_g_percentuale_mac>= 16 && x_g_percentuale_mac<=36)
fprintf('\n         16< xg  MAC <36 satisfies limits\n ')
end
if(x_g_percentuale_mac< 16)
curiosity=1;
fprintf('\n         xg  MAC <16 does NOt satisfy limits \n')
disp('      Not ready for trim condition ');
end
if(x_g_percentuale_mac> 36)
curiosity=1;
fprintf('\n         xg MAC >36 does NOT satisfy limits \n')
disp('      Not ready for trim condition ');
end
%controllo su massa e momneto totale che devono stare nel masse e moment
    envelope
momento1= -; % curva di sinitra del frafico al di sotto dei 460kg
momento2=-; % curva di detsra
if (somma_momentum>momento1 && somma_momentum<momento2)
fprintf('\n-> Weight and moment( %.2f Kgm) are in the range of  Weight and
    Moment Envelope',somma_momentum)
fprintf('\n          Momentt satisfis the condition    %.2f <Moment< %.2f \n'
    , momento1,momento2)
else
lander=1;
fprintf('\n -> Weight and moment(%.2f Kgm) are NOT in the range of  Weight
    and Moment Envelope\n',somma_momentum)
fprintf('\n    Moment does not satisfy the condition %.2f <moment< %.2f \n',
    momento1,momento2)
disp('       Not ready for trim condition ');
end
%print dei momenti di inerza
fprintf('\n-> Moment of inertia:\n')
fprintf('\n   Ixx=%.2f  Iyy=%.2f Izz=%.2f Jxz =%.2f  in Kgm^2 \n',Ixx_tot,
    Iyy_tot, Izz_tot, Jxz)
fprintf('\n')

if curiosity==0 && lander==0
disp('             Ready for trim condition ');
else
disp('             NOT READY for trim condition ');
end
end

%controllo su masse > di 460 kg(controllare cg envelope)
if(MTOW>460 && MTOW<=600 && massa_massima_fuel>fuel)
fprintf ('-> TOW = %.1f Kg ok,  FUEL =%.1f Kg ok\n',MTOW,fuel)
fprintf('\n-> Gravity center coordinates given from datum:\n')
fprintf('\n          Xg=%.3f m  Yg=0 m   Zg=%.3f m\n',xg, zg)
fprintf('\n-> Xg given as a percentage of main aerodynamic corde: %.2f % \n',
    x_g_percentuale_mac)
```

```
%calcolo del xg del xg envlope nel tratto lineare
xg_confronto=(MTOW-460)/25.454+16;

% controlli sul xg che sia nel xg envelope

% controllo se xg ricade nel tratto lineare del xg envlope
if (x_g_percentuale_mac>= 16 && x_g_percentuale_mac<21.5)
if (x_g_percentuale_mac>=xg_confronto)
fprintf('\n         %.2f< xg  MAC <36 satisfies limits\n ',xg_confronto)
else
curiosity=1;
fprintf('\n         %.2f> xg  MAC does NOT satisfy limits\n ',xg_confronto)
disp('       Not ready for trim condition ');
end
end

%controllo se xg ricade non nel tratto lineare del xg envelope
if(x_g_percentuale_mac>= 21.5 && x_g_percentuale_mac<=36)
fprintf('\n         %.2f< xg  MAC <36 satisfies limits\n ',xg_confronto)
end
if(x_g_percentuale_mac< xg_confronto)
curiosity=1;
fprintf('\n         xg  MAC <16 does NOT satisfy limits \n')
disp('      Not ready for trim condition ');
end
if(x_g_percentuale_mac> 36)
curiosity=1;
fprintf('\n       xg MAC >36 does NOT satisfy limits \n')
disp('       Not ready for trim condition ');
end

%controllo che  massa e momento sinao nel mass and momento evnelope
momento3=-; %curva di sinsitra del grafico al di sopra dei 460kg
momento2=-; %curva di destra
if (somma_momentum>momento3 && somma_momentum<momento2)
fprintf('\n-> Weight and moment( %.2f Kgm) are in the range of  Weight and
    Moment Envelope',somma_momentum)
fprintf('\n              Moment satisfies the condition    %.2f <Moment< %.2f
    \n', momento3,momento2)
else
lander=1;
fprintf('\n -> Weight and moment(%.2f Kgm) are NOT in the range of  Weight
    and Moment Envelope\n',somma_momentum)
fprintf('\n    Moment does not satisfy the condition %.2f <moment< %.2f \n',
    momento3,momento2)
disp('       Not ready for trim condition ');
end
%print momenti di inerzia
fprintf('\n-> Moment of inertia:\n')
fprintf('\n   Ixx=%.2f  Iyy=%.2f Izz=%.2f Jxz =%.2f  in Kgm^2 \n',Ixx_tot,
    Iyy_tot, Izz_tot, Jxz)
fprintf('\n')

% print se  possibile fare condizioni di trim o no
if curiosity==0 && lander==0
disp('           Ready for trim condition ');
else
disp('           NOT READY for trim condition, see errors ');
end
end
```

```
disp('=======================');


xg = -xg; %questo cambiamento di segno esiste solo per evitare di cambiare
%le equazioni per il trasporto dei momenti che abbiamo  inserito in fdc
zg = -zg; %idem come sopra
zg_solo_per_propeller = zg;
xg = 0;%DA USARE SOLO SE RIFERITO RISPETTO AL BARICENTRO
% zg = 0;%SOLO SE RIFERITO AL DATUM
```

Appendix E

Datcom.out processing for interpolation

```
%NBBBBB fare attenzione ai segni
format long
alldata_1=datcomimport('usato2.out')
alpha=alldata_1{1}.alpha;
alpha_rad=alpha.*pi./180;

%FLAP%%%%%
deltaflap= alldata_1{1}.delta;
deltacl= alldata_1{1}.dcl_sym;
deltacd=alldata_1{1}.dcdmin_sym;
cdindotta=alldata_1{1}.dcdi_sym;

%derivata Cmdf
deltacm=alldata_1{1}.dcm_sym;% incremnto cm dovuto flap
deltacm=[deltacm(1) deltacm(2) deltacm(3)];

%%%%%%%% VELIVOLO %%%%%%

clfinale=alldata_1{3}.cl;
cdfinale=alldata_1{3}.cd;
cm=alldata_1{3}.cm;

%componenti coeff. di forza in assi datum
cxpuro=alldata_1{3}.ca;
czpuro=alldata_1{3}.cn;

% coeff forza in assi bodi centrati in datum
CX_velivolo=-cxpuro;
CZ_velivolo=-czpuro;

% derivata Clr media, non unico valore
Clr=alldata_1{3}.clr;
Clr_fin=sum(Clr)/length(alpha)

% derivata Cnr media, non unico valore
Cnr=alldata_1{3}.cnr;
Cnr_fin=sum(Cnr)/length(alpha)

%derivata Clp media, non unico valore
Clp=alldata_1{3}.clp;
Clp_fin=sum(Clp)/length(alpha)

% derivata Cnp media, non unico valore
Cnp=alldata_1{3}.cnp;
```

```
Cnp_fin=sum(Cnp)/length(alpha)

% derivata Cyp media,  non unico valore
Cyp=alldata_1{3}.cyp;
Cyp_fin=sum(Cyp)/length(alpha)

%derivata Clb media , non unico valore
Clb=alldata_1{3}.clb;
Clb_fin=sum(Clb)/length(alpha);

%EQILIBRATORE%%%%%
deltae=(alldata_1{3}.delta).*pi./180;% deflessione equilibratore in rad
%derivata Cmde
delta_e_cm=alldata_1{3}.dcm_sym; %incremento cm dovuto equilibratore

%%%% aileron clroll delta a
Clroll_aileron=alldata_1{2}.clroll; %incrementi sul cl roll dovuti a
    escursioni aileron
delta_aileron=(alldata_1{2}.deltar).*pi./180;%escursioni aileron

%%%% cn delta a
cn_aileron=alldata_1{2}.cn_asy; % variazione di cn causato al variare di
    aileron e incidenza

for i=1:1:length(alpha)
if(alpha(i)==0)
t=i;
cn_aileron_aplha_null=cn_aileron(t,:);
end
end

%%%% COMPONENTI DI FORZA CALCOLATI%%%%
%aumento cl dovuto flap e variazione alpha
cl_flap0=deltacl(1);
cl_flap1=deltacl(2);
cl_flap2=deltacl(3);

%aumento cd dovuto flap e variazione alpha e cd indotta
cd_flap0=cdindotta(:,1);
cd_flap1=cdindotta(:,2);
cd_flap2=cdindotta(:,3);

%cx e xz dovuti flapp 0
cl= cl_flap0;
cd= cd_flap0;
i=1;
for k=1:1:length(alpha)
alpha1=alpha(k);
cx0(i)=-cl*sind(alpha1)+cd(i)*cosd(alpha1);
cz0(i)=cl*cosd(alpha1)+cd(i)*sind(alpha1);
i=i+1;
end

%cx e xz dovuti flapp 1
cl= cl_flap1;
cd= cd_flap1;
i=1;
for k=1:1:length(alpha)
alpha1=alpha(k);
cx1(i)=-cl*sind(alpha1)+cd(i)*cosd(alpha1);
cz1(i)=cl*cosd(alpha1)+cd(i)*sind(alpha1);
```

```
i=i+1;
end

%cx e xz dovuti flapp 2
cl= cl_flap2;
cd= cd_flap2;
i=1;
for k=1:1:length(alpha)
alpha1=alpha(k);
cx2(i)=-cl*sind(alpha1)+cd(i)*cosd(alpha1);
cz2(i)=cl*cosd(alpha1)+cd(i)*sind(alpha1);
i=i+1;
end

% cambaindo di segno ai coeff di forza si ha il segno che rispecchia xyz
% body

cx0=(-1)*cx0;
cx1=(-1)*cx1;
cx2=(-1)*cx2;
cz0=(-1)*cz0;
cz1=(-1)*cz1;
cz2=(-1)*cz2;

Xf=[-cxpuro cx0' cx1'  cx2' alpha']
Zf=[-czpuro cz0' cz1'  cz2' alpha']

%calcolo Cxdf e Xzdf
%NNNNBBB CX dleta flap Cz delta flap contempla il caso non contemplato Cx e
%Cz alpha*delta flap, ossia cosidero Cx e Cz valori a alpha=0

escusione_flap=[0 deltaflap(2)  deltaflap(3)];
escusione_flap_rad=escusione_flap.*pi./180;

escusione_flap_rad20= deltaflap.*pi./180;

%trovo valore per incidenza nulla
for i=1:1:length(alpha)
if(alpha(i)==0)
forza_x_alpha_null=[cx0(i) cx1(i)  cx2(i)];
forza_z_alpha_null=  [ cz0(i) cz1(i)   cz2(i)];
end
end

%calcolo Cxadf e Xzadf

adf0=0.*alpha_rad;% moltiplicazione delta flap0 per incidenze
adf10=escusione_flap_rad(2).*alpha_rad;
adf30=escusione_flap_rad(3).*alpha_rad;

af=[ adf10 adf30 ];
CXaf=[ cx1 cx2];
CZAf=[ cz1 cz2 ];

%%%FORZE EQUILIBRATORE %%%
%Incremento dovuto deflessioen eq
delta_e_cl= alldata_1{3}.dcl_sym;
delta_e_cd=alldata_1{3}.dcdmin_sym;
cd_idotta_eq=alldata_1{3}.dcdi_sym

% considero solo 2 valori di escursione eq escluso zero e copreso max
```

```
% escursioni positivi e negative

for j=1:1:length(deltae)
cl= delta_e_cl(j)
cd=cd_idotta_eq(:,j);
i=1;
for k=1:1:length(alpha)
alpha1=alpha(k);
cze(j,i)=cl*cosd(alpha1)+cd(i)*sind(alpha1);
i=i+1;
end
end

%concorde con assi bodi
cze=-cze

%NNNNBBB  Cz delta eq contempla il caso per cui volgio variazione di z
%dovuto solo a eq

for i=1:1:length(alpha)
if(alpha(i)==0)
forza_zee_alpha_null= [cze(:,i)];
end
end
```

Appendix F

Matlab code for derivatives from Napolitano

```
        %% CYdr, Cndr, Cldr
Zw = 0.35;% preso secondo figura 4.10 pagina 141, positivo
d = 1.31 ; %diametro della fusoliera come indicato in figura 4.10 pagina 141,
%preso il valore della sezione maggiore su datcom
AR = 6.611;
b = 12.956*2/3.28;
Sv = 1.28; %da datcom
S = 9.51;
M = 0.106;
beta = (1-M^2)^0.5;
tau = 0.27; %calcolato come corda rudder fratto corda tv presa da datcom
    (0.320/1.18)
etav = 0.9; %napolitano docet
lambdalev = 60*pi/180; %misurato
lambdaquartiv = 55*pi/180; %da datcom
lambdamezziv = 50*pi/180; %misurato
ARv = 1.09;
Khv = 1.1; %da figura 4.18 pag 145 libro usando in ascissa Sh/Sv = 1.65
%calcolato con valori da datcom
c1 = 1.65; % calcolato da grafico 4.15 pag 144 libro usando in ascissa bv/2*
    r1=1.475
% in cui 2*r1 = 0.8 e bv = 1.18
c2 = 1; %calcolato da figura 4.16 pag 144 del libro usando in ascissa Zh/bv
    in cui
% bv = 1.18 e Zh = -0.1 (da pagina 145 libro figura 4.17, distanza tra base
    tailv e tailo)
%e usando come parametro x_AC_H-V/macv = 0.30/1.18 misurato
%(con dati da datcom)
ARveff = c1*ARv*(1+Khv*(c2-1)); % p 203 libro
k = 1+((8.2-2.3*lambdalev)-ARveff*(0.22-0.153*lambdalev))/100;% p 203 libro
CLalphav = 2*pi*ARveff/(2+sqrt(((ARveff^2*beta^2/k^2)*(1+tan(lambdamezziv)/
    beta^2)) +4));%p 203 libro
deltaKR = 0.75; %calcolato da 149 figura 4.27 usando etai = 0.1707 e etaf = 1
CYdr = abs(CLalphav)*etav*(Sv/S)*deltaKR*tau %da pagina 148 libro

Zr = 0.485; %da datcom (Ymac del tail verticale)
Cldr = CYdr*Zr/b

Lv = 5.08-1.0280; %distanza tra datum e punto di applicazione della forza
% lungo y dovuta alla variazione di rudder, misurata dal disegno
Cndr = -CYdr*Lv/b
```

```
%% CYr

kyv = 0.77; %da figura 4.13 pagina 143, con in ascissa bv/2r = 1.475
Xv = 4.6-1.0280; %distanza tra datum e fuoco tail verticale misurata
etav_unopiudesigmasudebeta = 0.724+3.06*(Sv/S)/(1+cos(lambdaquartiv)) + 0.4*
    Zw/d + 0.009*AR;
CYbetav = -kyv*CLalphav*etav_unopiudesigmasudebeta*Sv/S;
CYr = -2*CYbetav*(Xv/b);
```

Appendix G

Matlab code for engine-propeller

Below is the matlab code for the operation of the simulink model of the engine and propeller only. For the data on eta and ct use the file "engine_propeller.m", for reasons of space it has been omitted in the following.

```
%vettore di stato
V = 60; %m/s
alpha = 3;
beta = 0;
p = 0;
q = 0;
r = 0;
psi = 0;
theta = 0;
phi = 0;
xe = 0;
ye = 0;
H = 1000;

x = [V alpha beta p q r psi theta phi xe ye H]';

% mootre + eliche NB esiste legame n e manetta inizlae vedere sezione 5
% manuale di volo
n = 4800;
pz = 24; %percentuale di manetta
uprop = [n pz];
riduzione=2.43; % rapporto di riduzione del riduttore
eta = 0.7; %rendimento eleica insiale
D = 1.75; %diametro dell'elica in metri


%condizioni alla quota di volo
rho = 1.1117;
ps = 898.76;
T = 281.65;
mu = 1.76e-5;
g = 9.81;
yatm = [rho ps T mu g];

a = 336.44;
M = V/a;
qdyn = 0.5*rho*V^2;
yad1 = [a M qdyn];

% condizioni standard s.l.
```

```
rhoSL = 1.225;
TSL = 288.15;
psi = (yatm(1)/rhoSL)*sqrt(TSL/yatm(3));

densita_fuel = 737.4;% [kg/m3]valor medio ma da cambiare in base al tipo
% eta e ct per 2000 gir/min a fl50(1524 m) valore intermendio

%eta
A_2000_fl50_eta=[ 0 0.10000 0.20000 0.30000 0.40000 0.50000 0.60000 0.70000
    0.80000.....];
%corretta questa sotto senza prima riga gamma e prima colonna di cp
A2000_fl100_eta=A_2000_fl50_eta([2:end],[2:end]);


%ct
A_2000_fl50_ct=[0 0.10000 0.20000 0.30000 0.40000 0.50000 0.60000 0.70000
    0.80000....];
%corretta queat sotto senza prima riga gamma e prima colonna di cp
A2000_fl100_ct=A_2000_fl50_ct([2:end],[2:end]);
```

Appendix H

Poropeller manufacturer equations

mt-propeller
ENTWICKLUNG GMBH

Airport Straubing-Wallmühle
94348 Atting / Germany
Telefon 49-(0)9429-9409-0
Telefax 49-(0)9429-8432
E-mail: sales@mt-propller.com

Propeller Performance (SI-Units)

$$C_P = \frac{P*1000}{\rho*(N/60)^3*D^5} \qquad \text{Power Coefficient [-]}$$

$$J = \frac{V*16{,}66}{N*D} \qquad \text{Advanced Ratio [-]}$$

V=0:

$$T = C_T*\rho*(N/60)^2*D^4 \qquad \text{Thrust [N]}$$

$$C_T = \frac{T}{\rho*(N/60)^2*D^4}$$

V>0:

$$T = \frac{3600*\eta*P}{V} \qquad \text{Thrust [N]}$$

C_T=	Thrust Coefficient [-]
P=	Engine Power [kW]
T=	Thrust [N]
N=	Propeller Speed [1/min = RPM]
D=	Prop. Diameter ∅ [m]
ρ=	Density of the air = 1,225 kg/m³ @ SL, ISA
η=	Propeller efficiency [-]
V=	Aircraft True Airspeed [km/h]

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