

CRANFIELD UNIVERSITY

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**Aerodynamics of Propulsion System
Integration
for
Modern Aero-Engines**

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”Uczony jest w swojej pracowni nie tylko technikiem,
Lecz również dzieckiem wpatrzonym w zjawiska przyrody,
Wzruszające jak czarodziejska baśń.”

”A scientist in their lab is not only a technician,
but also a child gazing at nature’s phenomena,
which are moving like a magic fairy tale”

M. Skłodowska-Curie

Abstract

Over the past years, the fan diameter of a turbofan engine for commercial aircraft tended to increase and based on the requirement to improve the propulsive efficiency it is expected that the engine diameter will continue to increase. In this context, the issue of the integration of a larger powerplant with the airframe is a major concern. It is anticipated that the increase of engine diameter will eventually lead to an increase of the drag due to the engine installation, which could potentially outweigh the benefits of very high by-pass ratio cycles. This research aims to quantify the aerodynamic sensitivity to the key parameters of engine installation such as engine position, size and power-setting for prospective large turbofans.

A numerical study with the use of Computational Fluid Dynamics was completed to determine the effect of engine position, size and power setting on the performance of a typical 300 seater aircraft with podded engines at cruise condition. Two engines with very-high by-pass ratio and with different fan diameters and specific thrusts were simulated in isolation and in range of positions under the wing. In addition, the detailed breakdown of the interference force was investigated to determine the contribution of aerodynamic forces based on the components such as wing, nacelle and fuselage. The assessment of the flowfields was done to determine the aerodynamic origins of beneficial or penalising forces and to explain the observed trends in the aerodynamic forces.

The work concluded with the evaluation of the fuel burn reduction for the larger engine configuration. Over the long-haul cruise phase, the cycle benefits for the large engine with low specific thrust deteriorated from -5.8% reduction in fuel burn to -4.8% as compared with the baseline engine due to the effect of engine weight and throttle dependent aerodynamic forces. It was also found that the large engine was less sensitive to the engine position. Over the same range of positions, the sensitivity of 1.7% of reference thrust for the baseline engine reduced to 1.3% sensitivity for the 1.23 times larger engine. Overall, to aid the preliminary design of prospective large turbofans, a model for the quantification of the engine installation effect was proposed based on the available dataset. The model aims to outline the general trend in the determination of engine installation at the early stage of the preliminary design. A basic verification of the model was completed with an agreement of 1.25 aircraft drag count in the prediction of the overall installation effect based on the available dataset.

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Contents

Acknowledgements	vii
Abstract	vii
List of Contents	xiii
List of Figures	xxiii
List of Tables	xxv
Nomenclature	xxix
1 Introduction	1
1.1 Motivation	1
1.2 Aims and objectives	2
1.2.1 Research aim	2
1.2.2 Research objectives	3
2 Background	5
2.1 Experimental studies of isolated engines	5
2.1.1 Isolated Nacelles	5
2.1.2 Nozzles and Turbine Powered Simulators	7
2.2 Experimental studies of propulsion system integration	12
2.2.1 Through flow nacelles	12
2.2.1.1 DLR-F6 model	15
2.2.1.2 A Common Research Model	16
2.2.2 Turbine powered simulators	18
2.2.2.1 ALVAST	20
2.3 Numerical studies of propulsion system integration	22
2.3.1 Lower fidelity Computational Fluid Dynamics	22
2.3.2 Navier-Stokes CFD	23
2.3.3 Guidelines for numerical drag prediction	29
2.3.3.1 Meshing guidelines	30
2.3.3.2 Turbulence modelling	31
2.4 Drag extraction theory	33
2.4.1 Near-field drag extraction theory	33
2.4.2 Far-field drag extraction theory	34
2.5 The summary of literature review	35
3 Scope of the project	37
3.1 The gap in knowledge	37
3.2 The project roadmap	38
3.3 The test matrix	39

4	Methodology	43
4.1	Aircraft performance	43
4.1.1	Base of Aircraft Data (BADA)	44
4.1.2	Estimation of weight for the propulsion system	45
4.1.3	Cruise calculation	46
4.2	Engine design and performance methods	47
4.2.1	Engine performance modelling	47
4.2.2	Engine geometry generation	49
4.3	Computational Fluid Dynamics Methodology	49
4.3.1	Computational Domain	50
4.3.2	Meshing strategies	50
4.3.3	Solver settings and boundary conditions	54
4.3.4	Turbulence model	54
4.3.5	Iterative convergence	56
4.4	Current thrust and drag book-keeping system	56
4.4.1	Thrust and drag book-keeping system for the engine	57
4.4.2	Thrust and drag book-keeping system for the aircraft	59
4.5	Summary	62
5	Validation of numerical methods	63
5.1	Validation cases for isolated engines	63
5.1.1	ARA Cowls	63
5.1.2	DLR TPS	66
5.1.3	Dual Separate Flow Reference Nozzle	68
5.2	Clean-wing CRM	70
5.2.1	Grid independence study	70
5.2.2	Turbulence model study	73
5.2.3	Reynolds number study	76
5.3	CRM with through-flow nacelle	77
5.3.1	The effect of TFN installation	77
5.3.2	Pressure distribution	82
5.4	Summary and conclusions	84
6	Engine installation studies	85
6.1	Drag decomposition for configuration with a through flow nacelle	85
6.2	Engine characteristics	90
6.3	The effect of aircraft incidence on the engine installation	93
6.4	The influence of a engine pylon on the engine installation	102
6.5	The effect of engine position on the installation	106
6.5.1	The effect in the drag domain	106
6.5.2	The effect in the thrust domain	111
6.5.3	The overall installation effect	113
6.6	The effect of engine size	118
6.6.1	The effect of size with constant thrust	118
6.6.2	The comparison of results with previous studies	127
6.7	The effect of flight Mach number	128
6.8	The effect of engine power setting	135

6.9	Cruise fuel burn analysis	142
6.10	Conclusions	145
7	Discussion and synthesis	147
7.1	Model for engine size	147
7.2	Model for engine power setting	152
7.3	Model for engine position	156
7.4	Complete model	159
7.5	Verification of the low order models	159
7.6	Conclusions	163
8	Summary and conclusions	165
8.1	Main accomplishments	165
8.2	Main conclusions	167
8.3	Novelty	168
8.4	Publications	168
9	Future work	171
	References	178
A	Drag extraction methods	179
A.1	The Wake Integral Method	179
A.2	The drag extraction in ALVAST TPS studies	181
B	Engine design frameworks	185
B.1	Cycle optimisation routine - Automatch	185
B.2	GEMINI	186
B.2.1	Engine cycle input	187
B.2.2	Geometry generation	188
B.2.3	CFD process	189
C	Domain and grid studies	191
C.1	Domain studies	191
C.1.1	ARA Cowls	191
C.1.2	DSFRN	192
C.1.3	Isolated engine	192
C.2	Grid convergence	193
C.2.1	DLR-TPS	194
C.2.2	DSFRN	194
C.2.3	Isolated engine	195
C.2.4	Common Research Model	195
C.3	Iterative convergence	199
D	Detailed boundary conditions	201
D.1	Boundary conditions for the flight Mach number study	201
E	Additional results	203
E.1	Validation of drag methods for isolated nacelles	203
E.1.1	ARA	203

E.2	DSFRN - validation activity	205
E.3	Additional results on the engine installation	205
E.3.1	The effect of size with constant efflux	205
E.3.2	The effect of specific thrust	212

List of Figures

1.1	Detrimental installation effects outweighing the benefits of high by-pass ratio	2
2.1	Comparison of NACA-1 cowl with an example of ARA Cowl 1	6
2.2	Jet interference mechanism for the conical afterbody	8
2.3	Pressure distribution on the fan-cowl, core-cowl and plug of JT9D derived from Navier-Stokes calculation compared with the measurements	8
2.4	Experimental setup for the RB211 separate exhaust	9
2.5	Overall thrust force (F_M) for the RB211 model referenced to the idealised gross thrust of the exhaust (F_{ID}) as a function of fan exhaust pressure ratio	10
2.6	The experimental pressure distribution for the RB211 and CF6 core-cowls compared with the modified method of characteristics for RB211 ("theory")	10
2.7	Isentropic Mach number distributions on the nacelle, core-cowl and plug of the DLR-TPS model	10
2.8	The geometry of the Dual Separate Flow Reference Nozzle	11
2.9	An example of aft-mounted nacelle	13
2.10	Impact of chordwise and spanwise locations	13
2.11	NLF nacelle installation effects; $M=0.85$ subscale high-wing transport aircraft model	14
2.12	A DLR-F6 model mounted in wind tunnel of NASA NTF with the through-flow nacelle of CFM-56 with the long cowl	15
2.13	The installation drag ($C_{D\ inst}$) for the three configurations with CFM-56 long cowl. The experimental results at $M_\infty = 0.75$ compared with computations	16
2.14	A comparison between Airbus A330 and the CRM; based on Vassberg et al.	17
2.15	a) Through-flow nacelle compared with b) cold-jets simulator	19
2.16	The comparison of ALAVST airframe with the turbofan simulator (TF), very high by-pass ratio simulator (VHBR) and ultra-high by-pass ratio simulator (UHBR).	20
2.17	Installation drag ($C_{D,I}$) and jet interference ($C_{D,J}$) in relation to by-pass ratio; $M_\infty = 0.75$; subscale	21
2.18	Pressure distribution (C_p) at the wing cross-section inboard of the nacelle at DLR-F6 configuration; $M_\infty = 0.75$; $C_L \approx 0.5$; $Re = 3 \times 10^6$	23
2.19	The geometries of a) CFM-56 long-cowl b) CFM-56 short-cowl c) VHBR nacelles; The lift-drag polars for the numerical and experimental results for the DLR-F6 configuration for d) CFM long-cowl nacelles and e) short-cowl nacelle compared with very-high by-pass ratio (VHBR) nacelles; . . .	24

2.20	The installation drag ($\Delta C_{D \text{ inst}}$) for numerical and experimental results for the DLR-F6 configuration for a) range of long-cowl nacelles b) short-cowl nacelle compared with very-high by-pass ratio (VHBR) nacelle . . .	25
2.21	The pressure distribution (C_p) for numerical and experimental results for the DLR-F6 configuration with the CFM-56 long-cowl nacelle at inboard cross-section; $M_\infty = 0.75$; $C_L = 0.5$	26
2.22	The lift-drag polar for a) experimental and b) numerical results for the ALVAST configuration with the TF and VHBR turbine powered simulators; $M_\infty = 0.75$	27
2.23	The pressure distribution (C_p) for experimental and numerical results of ALVAST airframe with a) TF engine and b) VHBR engine on the inboard side of the installation; $M_\infty = 0.75$; $C_L = 0.5$	28
2.24	a) Considered engine positions for the VHBR engine on the ALVAST airframe and the pressure distribution (C_p) for b) range of vertical engine positions and c) range of horizontal engine positions; $M_\infty = 0.75$; $C_L = 0.5$	29
2.25	Jet-engine streamtubes and streamtube forces	33
2.26	Jet-engine aircraft domain decomposition	34
3.1	Project Roadmap; Main four branches of project are grouped vertically: Engine Studies, Aircraft Studies, Installation Studies and Drag Methods; The thesis chapters are annotated on the work blocks.	38
3.2	Sketch of key installation parameters	40
3.3	Map of investigated engine positions for the two engines in the current study in the context of existing engine installations based on Oliveira et al.	41
4.1	Flow chart for the integrated fuel burn calculation during the cruise phase of flight; the CFD results discussed in Chapter 6	47
4.2	Sketch of the three-spool engine architecture with key performance parameters	48
4.3	An example of the engine geometry created by GEMINI	49
4.4	The domain size for a) the isolated ARA fan-cowl b) the isolated DSFRN exhaust system c) the isolated engine and d) the aircraft configurations .	51
4.5	An example of the “fine” mesh for the aircraft configuration with the engine and the drag-domain pylon	53
4.6	Decomposition of modified Near Field Method forces acting on the entry streamtube and nacelle	57
4.7	The sketch of coordinate systems for the aerodynamic forces, aircraft and engine	59
4.8	Definition of installation drag at constant angle of attack ($\Delta C_{D \text{ inst} AoA}$) and at constant overall vertical force ($\Delta C_{D \text{ inst}}$); definition of the loss of lift due to the installation ($\Delta C_{L \text{ inst} AoA}$)	62
4.9	Definition of interference drag at constant angle of attack ($\Delta C_{D \text{ interf} AoA}$) and at constant overall vertical force ($\Delta C_{D \text{ interf}}$); definition of the interference loss of lift due to the installation at constant angle of attack ($\Delta C_{L \text{ interf} AoA}$)	62

5.1	a) ($C_{D'}$) as a function of Mach number for ARA cowls 1 and 3 at $MFCR = 0.7$; $AoA_{eng} = 0^\circ$; b) ($C_{D'}$) as a function of MFCR for ARA cowls 1 and 3 at $Re = 2.2 \times 10^6$; $M_\infty = 0.85$; $AoA_{eng} = 0^\circ$; CFD compared with measurements	65
5.2	Pressure distribution (C_p) on the ARA Cowls 1 and 3 at $M_\infty = 0.85$; $MFCR = 0.47$; $AoA_{eng} = 0^\circ$	66
5.3	The DLR TPS test case at $M_\infty = 0.17$ at FPR=1.2 for (a) Mass flow in the by-pass duct (W_{Bypass}) as fraction of reference massflow (W_{ref}) compared with experimental results; (b) Overall gross thrust of the TPS (F_{Gross}) as fraction of reference gross thrust from the experiment ($F_{Gross REF}$) compared with experimental results	68
5.4	(a) Contours of Mach number for the DLR TPS test case at $M_\infty = 0.17$ at FPR=1.6 ($FNPR \approx 1.64$) and (b) Isentropic Mach number in the by-pass duct compared with measurements	68
5.5	Contours of Mach numbers for DSFRN; $M_\infty = 0.01$, $FNPR = 2.6$	69
5.6	Variation of (a) C_d^{BPD} , (b) C_d^{Core} and (c) C_V with FNPR and CNPR for the DSFRN. Current numerical results (solid black line and diamonds) compared with the experimental results (blue line) and the range of PAW2 results (dashed lines)	70
5.7	Computational drag polar from for clean wing CRM (WBT0) for ‘coarse’, ‘medium and ‘fine mesh’ compared with computational data for rigid geometry by Rivers et al and measurements using $k-\omega$ SST and S-A models; band of structured-mesh solutions from 4th Drag Prediction Workshop marked in grey; all data at $Re = 5 \times 10^6$ and $M_\infty = 0.85$; all current CFD with $k-\omega$ SST	71
5.8	Pressure distribution C_p for clean wing CRM (WBT0) for two mesh resolutions ‘medium’ and ‘fine’ at $AoA_{A/C} = 2.5^\circ$; $C_{L A/F} = 0.51$; $y/(b/2) = 0.283$; both with structured grid and $k-\omega$ SST turbulence model; numerical results compared with experiments by Rivers et al. at $AoA = 2.90^\circ$; $C_{L A/F} = 0.494$; $y/(b/2) = 0.283$; All data at $Re = 5 \times 10^6$ and $M_\infty = 0.85$ 73	
5.9	Computational drag polar from for rigid clean wing CRM (WBT0) for ‘medium mesh’ for different turbulence models: Spalart-Allmaras (S-A) and $k-\omega$ SST and transitional $k-\omega$ SST; all data compared with measurements; all data at $Re = 5 \times 10^6$ and $M_\infty = 0.85$	74
5.10	Pressure distribution C_p for clean wing CRM (WBT0) for “medium” mesh resolution at $AoA_{A/C} = 2.5^\circ$; $C_{L A/F} = 0.51$; $y/(b/2) = 0.283$; $k-\omega$ SST compared with $k-\omega$ SST $\gamma-\theta$ turbulence model; numerical results compared with experiments by Rivers et al. at $AoA = 2.91^\circ$; $C_{L A/F} = 0.52$; $y/(b/2) = 0.283$; All data at $Re = 5 \times 10^6$ and $M_\infty = 0.85$; based on del Vas	74
5.11	Contours of skin friction coefficient (C_f) for clean wing CRM (WBT0) for “medium” mesh for different turbulence models: $k-\omega$ SST and transitional $k-\omega$ SST $\gamma-\theta$; all data at $Re = 5 \times 10^6$ and $M_\infty = 0.85$; based on del Vas	75
5.12	Computational drag polar for rigid clean wing CRM (WBT0) for ‘medium mesh’ for three Reynolds numbers 5×10^6 , 19.8×10^6 and 305×10^6 ; all data compared with measurements; all data computed with $k-\omega$ SST at $M_\infty = 0.85$	76

5.13	Lift-drag polar for CRM with (WBNP) and without a TFN (WB) for “coarse”, “medium” and “fine” meshes; computed with $k-\omega$ SST; numerical results compared with experimental by Rivers et al. at $Re = 5 \times 10^6$ and $M_\infty = 0.83$	78
5.14	Installation drag coefficient ($\Delta C_{D\ inst}$) for CRM WB and WBNP subscale as function of lift coefficient ($C_{L\ tot}$); comparison between CFD coarse, medium, fine meshes and experimental data by Rivers et al.; fully turbulent with $k - \omega$ SST; $M_\infty = 0.83$; $Re = 5 \times 10^6$	79
5.15	Installation drag coefficient ($\Delta C_{D\ inst}$) for CRM WB and WBNP subscale as function of lift coefficient ($C_{L\ tot}$); comparison between CFD coarse, medium and fine meshes and experimental data by Rivers et al.; modelled transition with $k - \omega$ SST $\gamma - \theta$; $M_\infty = 0.83$; $Re = 5 \times 10^6$	80
5.16	Installation drag coefficient ($\Delta C_{D\ inst}$) for CRM WB and WBNP subscale as function of lift coefficient ($C_{L\ tot}$); comparison between CFD medium mesh and experimental data by Rivers et al.; $k - \omega$ SST; $Re = 5 \times 10^6$	80
5.17	Installation lift coefficient ($\Delta C_{L\ inst}$) for CRM WB and WBNP subscale as function of lift coefficient; comparison between CFD medium mesh and experimental data for Mach number of 0.85	81
5.18	Comparison of pressure distribution C_p for CRM with (WBNP) and without the TFN (WB) with the experimental data measured at (a) chosen spanwise positions of the wing; b) $y/(b/2) = 0.201$ c) $y/(b/2) = 0.283$ d) $y/(b/2) = 0.397$; Computation with $k-\omega$ SST fully turbulent, “medium” mesh; clean wing (WB) $C_{L\ tot} = 0.5$ for both experimental and numerical; experimental comparison between WB and WBNP at constant angle of attack $AoA_{EXP} = 2.87^\circ$; numerical comparison between WB and WBNP at constant angle of attack $AoA_{EXP} = 2.5^\circ$; $Re = 5 \times 10^6$; $M_\infty = 0.83$;	83
6.1	(a) pressure and (b) viscous contributions to total drag coefficient for CRM WBT0NP subscale as function of aircraft angle of attack; results for CFD fine mesh data for Mach number of 0.83	86
6.2	Drag breakdown for the configurations with (WBT0NP) and without (WBT0N) a pylon. The drag contribution for the nacelle (Nac.) and pylon (Pyl.) in chosen aircraft configurations compared against drag coefficient for isolated nacelles	88
6.3	a) Comparison of drag coefficient as a function of aircraft $AoA_{A/C}$ for airframe (WBT0) and nacelle on their own and as component of WBT0N configuration; b) Aerodynamic interference ($C_{D\ interf AoA}$) as a function of aircraft $AoA_{A/C}$ for airframe, nacelle and total aerodynamic interference; $Re = 5 \times 10^6$; $M_\infty = 0.83$	89
6.4	An example of the engine geometry created by GEMINI	91
6.5	Contours of Mach number for a) E1 engine and b) E2 engine at $Re = 45 \times 10^6$, $M_\infty = 0.82$, $AoA_{eng} = 4.25^\circ$	92
6.6	Contours of Mach number (M) at a) engine axis for a) CRM clean-wing configuration, b) isolated E1 engine and c) CRM with E1 engine in position C3; $Re = 45 \times 10^6$; $M_\infty = 0.82$; $AoA_{A/C} = 2.5^\circ$	94

6.7	a) The total drag ($C_{D\ tot}$) and overall vertical force ($C_{Z'}$) polar for the CRM without (Clean) and with (E1 in pos C3) the E1 engine in position C3; b) the installation drag coefficient ($\Delta C_{D\ inst}$; solid line) compared with the modified nacelle drag ($C_{D^* \ nac}$; dashed line) of isolated and wing installed engines; $Re = 45 \times 10^6$; $M = 0.82$;	95
6.8	a) The loss of the overall lift coefficient ($\Delta C_{Z'}$) as a function of aircraft incidence ($AoA_{A/C}$), based on the values of overall vertical force coefficient ($C_{Z'}$) for the clean-wing and E1 engine configurations b) the spanwise distribution of non-dimensional lift coefficient ($C_{L\ loc} \cdot \frac{c_{loc}}{c_{ref}}$) for CRM with (installed) and without (clean) engine at $AoA_{A/C} = 0.0^\circ$ and $AoA_{A/C} = 2.5^\circ$; $Re = 45 \times 10^6$; $M = 0.82$;	96
6.9	Pressure coefficient distribution at topline of the fan-cowl for installed engine at given aircraft angles of attack ($AoA_{A/C}$), compared with isolated engines at its angle of attack (AoA_{eng}); $Re = 45 \times 10^6$; $M = 0.82$	97
6.10	The overall gross propulsive force (GPF_ξ) for the Baseline engine in isolation and under the wing (installed) compared with the modified gross propulsive force (GPF_ξ^*); non-dimensionalised by GPF_ξ of an isolated engine at $AoA_{A/C} = 0^\circ$ ($GPF_{ref} = 160462N$);	99
6.11	The discharge coefficient (C_d) for core nozzle and by-pass nozzle of the Baseline engine in isolation compared with the Baseline engine under the wing (installed); $Re = 45 \times 10^6$; $M = 0.82$;	99
6.12	Overall installation force (F_{inst}) diagram for the CRM with E1 engine in position C3 at a range of overall vertical force coefficient ($C_{Z'}$); $Re = 45 \times 10^6$; $M_\infty = 0.82$	100
6.13	The breakdown of the interference force (F_{interf}) on to interference on thrust (ΔF_N^*), interference on the nacelle (ΔD_{nac}^*), and interference on the airframe ($\Delta D_{A/F}$) for the CRM with E1 engine in position C3; $Re = 45 \times 10^6$; $M_\infty = 0.82$	101
6.14	Contours of Mach number for the CRM with E1 engine in position D3 a) without the pylon and b) with the pylon; $Re = 45 \times 10^6$; $M = 0.82$; $AoA_{A/C} = 2.5^\circ$	102
6.15	The overall vertical force ($C_{Z'}$) and aircraft total drag ($C_{D\ tot}$) polar for the CRM without an engine, with the E1 engine in D3 position and with the E1 engine in D3 position with the pylon; $Re = 45 \times 10^6$; $M_\infty = 0.82$	103
6.16	The a) airframe drag ($C_{D\ A/F}$) for the configuration without engine, with E1 engine in D3 position and with E1 engine in D3 position with the pylon b) the modified nacelle drag ($C_{D^* \ nac}$) and pylon drag ($C_{D\ pyl}$) for the isolated E1 engine, E1 engine under the wing in D3 position and the E1 engine under the wing in D3 position with the pylon; $Re = 45 \times 10^6$; $M = 0.82$; $AoA_{A/C} = 2.5^\circ$	104
6.17	Overall installation force (F_{inst}) diagram for the CRM with E1 engine in position D3 at a overall vertical force coefficient ($C_{Z'} = 0.5$); $Re = 45 \times 10^6$; $M_\infty = 0.82$	105
6.18	Contours of Mach Number at span-wise cross-section through the engine axis for the clean-wing isolated engine and a range of considered positions (A1, A2, A3, B1, B2, B3, C1, C2, C3); $Re = 45 \times 10^6$; $M = 0.82$; $AoA_{A/C} = 2.5^\circ$	107

- 6.19 Overall vertical force ($C_{Z'}$) and total drag ($C_{D\ tot}$) polar for CRM without (Clean) and with engine in a) three horizontal positions A1, B1 and C1; b) three vertical positions C1, C2, C3; c) four horizontal positions A3, B3, C3, D3; $Re = 45 \times 10^6$; $M = 0.82$ 108
- 6.20 a) Pressure drag coefficient and b) viscous drag coefficient as a function of aircraft angle of attack ($AoA_{A/C}$) for clean-wing aircraft, isolated nacelle and three engine-aircraft configurations (A1/B1/C1); $Re = 45 \times 10^6$; $M = 0.82$; $Re = 45 \times 10^6$; $M = 0.82$; 109
- 6.21 a) Pressure distribution at nacelle top-line ($\Theta = 0^\circ$) for engine E1 in three engine positions (A1, B1 and C1) compared with an isolated engine for idealised engine incidence $AoA_{eng,ideal} = 1.75^\circ$ ($AoA_{A/C} = 0.0^\circ$) and b) $AoA_{eng,ideal} = 4.25^\circ$ ($AoA_{A/C} = 2.5^\circ$); $Re = 45 \times 10^6$; $M = 0.82$; 110
- 6.22 a) Pressure distribution at wing cross-section at engine installation plane ($y/((b/2)) = 0.283$) at $AoA_{A/C} = 0^\circ$ for engine E1 in three engine positions (A1, B1 and C1) compared with clean-wing configuration at $AoA_{A/C} = 0.0^\circ$ and b) compared at $AoA_{A/C} = 2.5^\circ$; $Re = 45 \times 10^6$; $M = 0.82$; 110
- 6.23 Contours of pressure distribution C_p on a) the wing in presence of the engine in position C1 viewed from the underside. Contours of pressure distribution on unwrapped surfaces of fan-cowl, core-cowl and plug for engines in position: b) C1, c) B1 and d) A1; $AoA_{A/C} = 2.5^\circ$; $Re = 45 \times 10^6$; $M = 0.82$; nacelle inboard for $\Theta < 0^\circ$ 111
- 6.24 Specific gross propulsive force (GPF_ξ) and modified gross propulsive force (GPF_ξ^*) for isolated nacelle compared with a) three engine horizontal positions (C1, B1 and A1) and b) three engine vertical positions (C1, C2, C3); $Re = 45 \times 10^6$; $M = 0.82$ 112
- 6.25 Pressure distribution at top-line ($\Theta = 0^\circ$) of engine core-cowl of an isolated engine compared with a) three wing installed engines in horizontal positions (C1, B1 and A1) b) three wing-installed engines in vertical positions (C1, C2, C3) at a respective engine incidence ($AoA_{eng,ideal}$); Engine installation angle $AoA_{inst} = 1.75^\circ$; $Re = 45 \times 10^6$; $M = 0.82$ 113
- 6.26 Overall installation force diagram for engine E1 in different engine positions; standard net thrust of isolated engine at cruise ($F_{N\ iso|Cz'=0.5}$) plus the effect of exhaust forces ($+\theta_{exhaust}$), nacelle drag ($+D_{nac}^*$), and detailed interference effects ($\Delta F_N^* / \Delta D_{nac}^* / \Delta D_{A/F}$); $Re = 45 \times 10^6$; $M = 0.82$; $C_{Z'} = 0.5$ 115
- 6.27 Map of the overall interference (F_{interf}) for baseline engine (E1), where the axial (dx) and vertical (dz) displacements were referenced to respective nacelle lengths (L_{nac}); $Re = 45 \times 10^6$; $M = 0.82$; $C_{Z'} = 0.5$ 115
- 6.28 The overall installation force diagram for engine E2 in different engine positions; standard net thrust of isolated engine at cruise ($F_{N\ iso|Cz'=0.5}$) plus the effect of exhaust forces ($+\theta_{exhaust}$), nacelle drag ($+D_{nac}^*$), and detailed interference effects ($\Delta F_N^* / \Delta D_{nac}^* / \Delta D_{A/F}$); $Re = 45 \times 10^6$; $M = 0.82$; $C_{Z'} = 0.5$ 116
- 6.29 Map of the overall interference (F_{interf}) for the larger engine (E2), where the axial (dx) and vertical (dz) displacements were referenced to respective nacelle lengths (L_{nac}); $Re = 45 \times 10^6$; $M = 0.82$; $C_{Z'} = 0.5$ 117

6.30	Contours of Mach number for engine-aircraft configurations a) with E1 engine and b) with E2 engine in engine position C3 at $AoA_{A/C} = 2.5^\circ$; $Re = 45 \times 10^6$; $M_\infty = 0.82$	118
6.31	a) Overall vertical force ($C_{Z'}$) and total drag ($C_{D\ tot}$) polar for CRM aircraft without (Clean) and with engines (E1 and E2), compared with drag polar of the clean-wing; b) total lift ($C_{L\ tot}$) as function of aircraft incidence ($AoA_{A/C}$) $Re = 45 \times 10^6$; $M_\infty = 0.82$	119
6.32	Pressure distribution for aircraft without (Clean) and with engines (E1 and E2); $Re = 45 \times 10^6$; $M_\infty = 0.82$; $AoA_{A/C} = 2.5^\circ$; $y/(b/2)$ from 0.2 to 0.5	120
6.33	Distribution of difference drag force per area (ΔD_{tot}) between a) configuration E1 and clean-wing configuration; between b) configuration E2 and clean-wing configuration; $Re = 45 \times 10^6$; $M_\infty = 0.82$; $AoA_{A/C} = 2.5^\circ$	122
6.34	Distribution of difference drag and thrust force per area ($\Delta \phi_{tot}$) between a) configuration E1 and isolated engine; between b) configuration E2 and isolated engine; $M_\infty = 0.82$; $AoA_{A/C} = 2.5^\circ$	122
6.35	Installation force diagram for two engine sizes E1 and E2 in a) baseline position C3 b) close coupled position C1 and c) furthest upstream position A1; standard net thrust of isolated engine at cruise ($F_{N\ iso Cz'=0.5}$) plus the effect of exhaust forces ($+\theta_{exhaust}$), nacelle drag (D_{nac}^*), and detailed interference effects ($\Delta F_N^* / \Delta D_{nac}^* / \Delta D_{A/F}$); $Re = 45 \times 10^6$; $M = 0.82$; $C_{Z'} = 0.5$	124
6.36	Map of the difference in overall installation effect ($F_{inst E2} - F_{inst E1}$) between E2 and E1 engines, where the axial (dx) and vertical (dz) displacements were referenced to nacelle length of E1 engine ($L_{nac E1}$); $Re = 45 \times 10^6$; $M = 0.82$; $C_{Z'} = 0.5$	125
6.37	Installation force diagram (X_{inst}) for two engine sizes E1 and E2 in a) baseline position C3 b) close coupled position C1 and c) furthest upstream position A1; standard net thrust of isolated engine at cruise ($F_{N\ iso Cz'=0.5}$) plus the effect of exhaust forces ($+\theta_{exhaust}$), nacelle drag (D_{nac}^*), and detailed interference effects ($\Delta F_N^* / \Delta D_{nac}^* / \Delta D_{A/F}$); $Re = 45 \times 10^6$; $M = 0.82$; $C_{Z'} = 0.5$	126
6.38	The installation force in "ALVAST definition" (X_{inst}) at $Re = 45 \times 10^6$; $M = 0.82$; $C_{Z'} = 0.5$; compared with the experimental results from ALVAST studies	128
6.39	a) Aircraft total drag ($C_{D\ tot}$) ; b) drag gradient ($\frac{dC_{D\ tot}}{dM}$) as a function of flight Mach number (M_∞) for the clean-wing CRM and the CRM with E2 engine in position C3; $AoA_{A/C} = 2.5^\circ$	129
6.40	Airframe drag ($C_{D\ A/F}$) for the clean-wing and the airframe in presence of E2 engine as a function of flight Mach number (M_∞); $AoA_{A/C} = 2.5^\circ$	130
6.41	Modified nacelle drag ($C_{D^*_{nac}}$) for the isolated E2 engine and E2 under the wing, both compared with the installation drag ($\Delta C_{D\ inst}$) for the presence of E2 engine under the wing of CRM in position C3 as a function of flight Mach number (M_∞); $AoA_{A/C} = 2.5^\circ$	130
6.42	Aircraft total lift ($C_{L\ tot}$) for the clean-wing CRM and the CRM with E2 engine in position C3; Loss of lift ($\Delta C_{L\ tot}$) due to the presence of E2 engine under the wing of CRM as a function of flight Mach number (M_∞); $AoA_{A/C} = 2.5^\circ$	131

- 6.43 Pressure distribution (C_p) at the wing cross-section inboard of the engine ($y/(b/2)=0.3$) for the clean-wing and the CRM with E2 engine in C3 position and C_p at the topline of the nacelle for the isolated engine and E2 engine in C3 position; M_∞ from 0.8 to 0.86; $AoA_{A/C} = 2.5^\circ$ 132
- 6.44 Thrust coefficient (C_V) for E2 engine in isolation and in configuration C3 under the wing of CRM as a function of flight Mach number (M_∞); $AoA_{A/C} = 2.5^\circ$ 133
- 6.45 Lift-drag polar for E1 engine in position C3 in four engine power settings from PS1 to PS4; $Re = 45 \times 10^6$; $M_\infty = 0.82$ 136
- 6.46 Modified engine drag ($C_{D^* nac}$) as function of engine ideal incidence ($AoA_{eng,ideal}$) for the E1 engine in isolation and under the wing in baseline position (C3) for a range of power settings from PS1 to PS4; $Re = 45 \times 10^6$; $M_\infty = 0.82$ 136
- 6.47 a) Pressure distribution (C_p) on the top line of the fan-cowl for E1 engine in baseline position (C3) in four engine power settings PS1 to PS4 at $AoA_{eng,ideal} = 4.25^\circ$ ($AoA_{A/C} = 2.5^\circ$), compared with the isolated E1 engine at $AoA_{eng,ideal} = 4.25^\circ$; b) Pressure distribution (C_p) on the wing at $y/(b/2)=0.3$ in presence of E1 engine in C3 position for PS1 to PS4 compared with the CRM clean-wing; $Re = 45 \times 10^6$; $M_\infty = 0.82$; 138
- 6.48 a) Pressure distribution (C_p) on the top line of the fan-cowl for E2 engine in baseline position (C3) in four engine power settings PS1 to PS4 at $AoA_{eng,ideal} = 4.25^\circ$ ($AoA_{A/C} = 2.5^\circ$), compared with isolated E2 engine at $AoA_{eng,ideal} = 4.25^\circ$; b) Pressure distribution (C_p) on the wing at $y/(b/2)=0.3$ in presence of E2 engine in C3 position for PS1 to PS4 compared with the CRM clean-wing; $Re = 45 \times 10^6$; $M_\infty = 0.82$; 138
- 6.49 Pressure distribution (C_p) on the top line of the core-cowl for E1 engine in position C3 for engine power settings from PS1 to PS4 at $AoA_{eng,ideal} = 4.25^\circ$ ($AoA_{eng} = 2.5^\circ$), compared with isolated engine at $AoA_{eng,ideal} = 4.25^\circ$; $Re = 45 \times 10^6$; $M_\infty = 0.82$; 139
- 6.50 Non-dimensionalised modified gross propulsive force (GPF^*/GPF_{ref}^*) for E1 engine in position C3 (C3) in four engine power settings compared with isolated engine; $Re = 45 \times 10^6$; $M_\infty = 0.82$ 139
- 6.51 The overall installation force diagram for engine E1 in four power settings from PS1 to PS4; standard net thrust of isolated engine at cruise ($F_{N iso|C_{Z'}=0.5}$) plus the effect of exhaust forces ($+\theta_{exhaust}$), nacelle drag ($+D_{nac}^*$), and detailed interference effects ($\Delta F_N^* / \Delta D_{nac}^* / \Delta D_{A/F}$); $Re = 45 \times 10^6$; $M = 0.82$; $C_{Z'} = 0.5$ 140
- 6.52 The overall installation force diagram for engine E2 in four power settings from PS1 to PS4; standard net thrust of isolated engine at cruise ($F_{N iso|C_{Z'}=0.5}$) plus the effect of exhaust forces ($+\theta_{exhaust}$), nacelle drag ($+D_{nac}^*$), and detailed interference effects ($\Delta F_N^* / \Delta D_{nac}^* / \Delta D_{A/F}$); $Re = 45 \times 10^6$; $M = 0.82$; $C_{Z'} = 0.5$ 141
- 6.53 Spatial sensitivity map of the fuel consumption ($\dot{m}_{fuel}$) for the baseline engine E1 as compared with the baseline position C3; $Re = 45 \times 10^6$; $M_\infty = 0.82$; ($C_{Z'} = 0.47$ for engine cruise power setting (MFCR=0.75, FNPR=2.71) 142
- 6.54 Spatial sensitivity map of the fuel consumption ($\dot{m}_{fuel}$) for the baseline engine E2 as compared with the baseline engine E1 in respective position; $Re = 45 \times 10^6$; $M_\infty = 0.82$; both engine-aircraft configuration at 'SoC' power setting and with balanced aircraft 143

6.55	Overall installation effect (F_{inst}) as percentage of reference thrust ($F_{N ref}$) for a) baseline engine E1 and b) E2 engine over the range of vertical force coefficients coefficients ($C_{Z'}$) and range of power settings represented by MFCR; $Re = 45 \times 10^6$; $M_\infty = 0.82$;	144
7.1	Proposed correlation for $\kappa_{\sigma 1}$ based on the E1, 1.23xE1 and 1.45xE1 CFD results for the D_{nac}^* ; $M_\infty = 0.82$; $Re = 45 \times 10^6$ $C_{Z'} = 0.5$	148
7.2	Proposed correlation for $\kappa_{\sigma 2}$ based on the E1, 1.23xE1 and 1.45xE1 CFD results for the ΔF_N^* ; $M_\infty = 0.82$; $Re = 45 \times 10^6$ $C_{Z'} = 0.5$	149
7.3	Proposed correlation for $\kappa_{\sigma 3}$ based on the E1, 1.23xE1 and 1.45xE1 CFD results for the ΔD_{nac}^* ; $M_\infty = 0.82$; $Re = 45 \times 10^6$ $C_{Z'} = 0.5$	150
7.4	Proposed correlation for $\kappa_{\sigma 4}$ based on the E1, 1.23xE1 and 1.45xE1 CFD results for the $\Delta D_{A/F}$; $M_\infty = 0.82$; $Re = 45 \times 10^6$ $C_{Z'} = 0.5$	150
7.5	Proposed correlation for $\kappa_{\psi 1}$ based on the E1 and E2 CFD results for the D_{nac}^* ; $M_\infty = 0.82$; $Re = 45 \times 10^6$ $C_{Z'} = 0.5$	153
7.6	Proposed correlation for $\kappa_{\psi 2}$ based on the E1 and E2 CFD results for the ΔF_N^* ; $M_\infty = 0.82$; $Re = 45 \times 10^6$ $C_{Z'} = 0.5$	154
7.7	Proposed correlation for $\kappa_{\psi 3}$ based on the E1 and E2 CFD results for the ΔD_{nac}^* ; $M_\infty = 0.82$; $Re = 45 \times 10^6$ $C_{Z'} = 0.5$	155
7.8	Proposed correlation for $\kappa_{\psi 4}$ based on the E1 and E2 CFD results for the $\Delta D_{A/F}$; $M_\infty = 0.82$; $Re = 45 \times 10^6$ $C_{Z'} = 0.5$	156
7.9	The sensitivity map for the ΔF_N^* relative to E1 in position C3; $M_\infty = 0.82$; $Re = 45 \times 10^6$ $C_{Z'} = 0.5$	157
7.10	The sensitivity map for the ΔD_{nac}^* relative to E1 in position C3; $M_\infty = 0.82$; $Re = 45 \times 10^6$ $C_{Z'} = 0.5$	158
7.11	The sensitivity map for the $\Delta D_{A/F}$ relative to E1 in position C3; $M_\infty = 0.82$; $Re = 45 \times 10^6$ $C_{Z'} = 0.5$	158
A.1	ARA experimental pitot-static rake	180
A.2	The split between the thrust and drag domains	181
A.3	Thrust accounting for a Turbine Powered Simulator (TPS) on ALVAST	182
A.4	The experimental procedure for the thrust and drag assessment	182
B.1	A typical convergence study from genetic algorithm; Plot of best and worst individual within each generation compared with an average result	186
B.2	The layout of engine design platform GEMINI based on Goulos et al.	187
B.3	The geometric representation of Bernstein polynomial	189
C.1	A sample iterative convergence for engine-aircraft configuration	199
C.2	A sample iterative convergence of the rearward oriented force (C_{RWD}) for engine-aircraft configuration	200
E.1	Drag coefficient ($C_{D'}$) as a function of freestream Mach number for ARA Cowl 1 for CFD analysis and experimental data at a) MFCR=0.6; b) MFCR=0.7	204
E.2	a) Comparison between three size engines and contours of Mach number (M) for b) E1 engine c) 1.23xE1 engine and d) 1.45xE1 engine; $M_\infty = 0.82$; $AoA_{A/C} = 2.5^\circ$	206
E.3	Lift-drag polar for three engine sizes E1, 1.23xE1 and 1.45xE1 in position C3; $Re = 45 \times 10^6$; $M_\infty = 0.82$	207

E.4	The engine installation drag at constant lift ($C_{D\ inst}$) as function of aircraft incidence ($AoA_{A/C}$) for three engine sizes E1, 1.23xE1 and 1.45xE1; all in position C3; $Re = 45 \times 10^6$; $M_\infty = 0.82$	207
E.5	Loss of lift ($\Delta C_{L\ tot}$) for three engine sizes E1, 1.23xE1 and 1.45xE1 in position C3; $Re = 45 \times 10^6$; $M_\infty = 0.82$	208
E.6	The spanwise lift distribution over the wing of the clean-wing CRM compared with the distribution for the wing in presence of engine in one of three sizes E1, 1.23xE1 and 1.45xE1; all in position C3; $Re = 45 \times 10^6$; $M_\infty = 0.82$; $AoA_{A/C} = 2.5^\circ$	208
E.7	Pressure distribution (C_p) at the wing cross-sections of the wing with $y/(b/2)$ of 0.2, 0.3, 0.4 and 0.5 for the clean-wing and the CRM with three sizes of the E1 engine: E1, 1.23xE1 and 1.45xE1; $M_\infty = 0.82$; $AoA_{A/C} = 2.5^\circ$	209
E.8	Installation force diagram for three engine sizes E1, 1.23xE1 and 1.45xE1 in baseline position C3; standard net thrust of isolated engine at cruise ($F_{N\ iso Cz'=0.5}$) plus the effect of exhaust forces ($+\theta_{exhaust}$), nacelle drag (D_{nac}^*), and detailed interference effects ($\Delta F_N^* / \Delta D_{nac}^* / \Delta D_{A/F}$); $Re = 45 \times 10^6$; $M = 0.82$; $C_{Z'} = 0.5$	210
E.9	Contours of Mach number for engine-aircraft configurations a) with E2 engine b) with 1.23xE1 engine; both in position C3 at $AoA_{A/C} = 2.5^\circ$; $Re = 45 \times 10^6$; $M_\infty = 0.82$	212
E.10	Installation force diagram for E2 engine and 1.23xE1 engine in the position C3; standard net thrust of isolated engine at cruise ($F_{N\ iso Cz'=0.5}$) plus the effect of exhaust forces ($+\theta_{exhaust}$), nacelle drag (D_{nac}^*), and detailed interference effects ($\Delta F_N^* / \Delta D_{nac}^* / \Delta D_{A/F}$); $Re = 45 \times 10^6$; $M = 0.82$; $C_{Z'} = 0.5$	213

List of Tables

2.1	Summary of nozzle operating conditions for DSFRN computations	11
3.1	Range of investigated engine positions for engine E1 and E2 over the selected range of engine power settings ($F_{N \text{ cruise}}$) and aircraft incidence ($AoA_{A/C}$)	42
4.1	Comparison of the WATE++ estimations for the E1 and E2 engines with the open domain data for RR Trent 1000 and GE NX	46
4.2	Domain specification for ARA Cowl, DLR-TPS, DSFRN, isolated en- gines and CRM configurations	50
4.3	Mesh topology for ARA Cowl, DLR-TPS, DSFRN, isolated engines and CRM configurations	52
4.4	The mesh size in million elements for ARA Cowl, DLR-TPS, DSFRN, isolated engines and CRM configurations	52
5.1	Aircraft total drag ($C_{D \text{ tot}}$) at chosen total lift coefficients ($C_{L \text{ tot}}$) for a range of mesh densities and compared with the experimental results; $Re = 5 \times 10^6$; $M_\infty = 0.85$; CFD with $k - \omega \text{ SST}$	72
5.2	Drag coefficient ($C_{D \text{ tot}}$) for chosen values of lift coefficient ($C_{L \text{ tot}}$) for the experimental and numerical results for the “medium” mesh cases at $M_\infty = 0.85$ and three Reynolds numbers 5×10^6 , 19.8×10^6 and 30×10^6	77
6.1	Key performance parameters for E1 and E2 engines	90
6.2	Engine power settings for the baseline engine (E1) and the large engine (E2)	91
6.3	Aircraft total drag ($C_{D \text{ tot}}$ [dc]) at a range of overall vertical forces ($C_{Z'}$) from 0.2 to 0.5 for CRM clean-wing, CRM with E1 engine in D3 position and CRM with E1 engine in D3 position and a pylon	103
7.1	The verification of the size model for the X_{inst}	151
7.2	Verification of the size model (κ_σ) for the D_{nac}^* , ΔF_N^* , ΔD_{nac}^* and $\Delta D_{A/F}$	151
7.3	Verification of the curve-fit for the effect of modified nacelle drag (D_{nac}^*)	160
7.4	Verification of the curve-fit for the interference effect on the modified thrust (ΔF_N^*)	160
7.5	Verification of the curve-fit for the interference effect on the modified nacelle drag (ΔD_{nac}^*)	161
7.6	Verification of the curve-fit for the interference effect on the modified nacelle drag ($\Delta D_{A/F}$)	162
7.7	Verification of the model for the installation effect ($X_{inst C3}$)	162

C.1	The results of domain study for the ARA Cowl 1 at $M=0.4$ and $MFCR=0.7125191$	191
C.2	The domain study for the DSFRN at $FNPR=2.2$	192
C.3	Sensitivity of discharge coefficient of the by-pass duct (C_d^{BPD}), core duct (C_d^{Core}), thrust coefficient (C_V^*), modified nacelle drag ($C_{D^* nac}$) of an isolated engine with respect to the domain size (R_{dom})	193
C.4	Grid Convergence Index (GCI) for the thrust coefficient (C_V^*) and discharge coefficients for the by-pass (C_d^{BPD}) and core (C_d^{Core}) nozzles of the DSFRN at $FNPR = 2.2$, $CNPR = 1.8$	195
C.5	Grid Convergence Index (GCI) for the thrust coefficient (C_V^*), modified nacelle drag coefficient ($C_{D^* nac}$) and discharge coefficients for the by-pass (C_d^{BPD}) and core (C_d^{Core}) nozzles of the isolated E1 engine at $M_\infty = 0.82$, $AoA_{eng} = 4.25^\circ$, $FNPR = 2.71$, $CNPR = 1.37$	195
C.6	The mesh size in million elements for chosen CRM configurations	196
C.7	Grid Convergence Index (GCI) for the drag coefficient ($C_{D tot}$) for the clean-wing with a tail (WBT0) at a range of aircraft angle of attack ($AoA_{A/C}$); $M_\infty = 0.85$, $Re = 5 \times 10^6$	196
C.8	Grid Convergence Index (GCI) for the lift coefficient ($C_{L tot}$) for the clean-wing with a tail (WBT0) at a range of aircraft angle of attack ($AoA_{A/C}$); $M_\infty = 0.85$, $Re = 5 \times 10^6$	196
C.9	Grid Convergence Index (GCI) for the lift and drag coefficients ($C_{L tot}$; $C_{D tot}$) for the clean-wing (WB) configuration of CRM at a range of aircraft angle of attack ($AoA_{A/C}$); $M_\infty = 0.82$, $Re = 5 \times 10^6$	197
C.10	Grid Convergence Index (GCI) for the lift and drag coefficients ($C_{L tot}$; $C_{D tot}$) for the CRM configuration with a through flow nacelle (WBNP) at a range of aircraft angle of attack ($AoA_{A/C}$); $M_\infty = 0.82$, $Re = 5 \times 10^6$	197
D.1	Engine boundary conditions for the E2 engine for a range of flight Mach numbers (M_∞)	202
E.1	Isolated nacelle configurations in validation study	203
E.2	Effect of sting corrections on the evalation of the nacelle drag	204
E.3	Summary of results from DSFRN computations and comparison to experimental nozzle performance	205
E.4	Total aircraft drag ($C_{D tot}$) at chosen total lift coefficients ($C_{L tot}$) for three engine sizes E1, 1.23xE1 and 1.45xE1	207

Nomenclature

Abbreviations

A/C	Aircraft - airframe and engine installation
A/F	Airframe - wing/body/tail
AoA	Angle of Attack
$AoA_{A/C}$	Angle of Attack for the aircraft
AoA_{eng}	Angle of Attack for the isolated engine
AoA_{inst}	Engine installation angle
AoA_{upwash}	Upwash angle of attack due to wing pressure-field
ARA	Aircraft Research Association
$BADA$	Base of Aircraft Data
BPD	Bypass Duct
BPR	By-Pass Ratio
CFD	Computational Fluid Dynamics
$CNPR$	Core Nozzle Pressure Ratio
COT	Combustor Outlet Temperature
CRM	The NASA Common Research Model
CST	Class Shape Transformation function
dc	Aircraft drag counts
DLR	German Institute of Aviation
DPW	Drag Prediction Workshop
$DSFRN$	Dual Separate Flow Reference Nozzle

<i>EoC</i>	End of Cruise
<i>FNPR</i>	Fan Nozzle Pressure Ratio
<i>FPR</i>	Fan Pressure Ratio
<i>GCI</i>	Grid Convergence Index
<i>GEM</i>	Geometric Engine Modeller
<i>GEMINI</i>	Geometric Engine Modeller Including Nozzle Installation
<i>GPF</i>	Gross Propulsive Force
<i>HPR</i>	High-speed shaft Pressure Ratio
<i>HPT</i>	High Pressure Turbine
<i>IPR</i>	Intermediate-speed shaft Pressure Ratio
<i>IPT</i>	Intermediate Pressure Turbine
<i>LPT</i>	Low Pressure Turbine
<i>MFCR</i>	Mass Flow Capture Ratio
<i>NPF</i>	Net Propulsive Force
<i>OPR</i>	Overall Pressure Ratio
<i>SFC</i>	Specific Fuel Consumption
<i>SM</i>	Superposition Model
<i>SoC</i>	Start of Cruise
<i>SoDe</i>	Start of Descent
<i>TEM</i>	Total Energy Model
<i>TET</i>	Turbine Entry Temperature
<i>TFN</i>	Through-flow nacelle
<i>TPS</i>	Turbine Powered Simulator
<i>WB</i>	Wing Body
<i>WBNP</i>	Wing Body Nacelle Pylon
<i>WBT0</i>	Wing Body Tail at 0°
<i>WBT0E</i>	Wing Body Tail at 0° Engine

$WBT0EP$	Wing Body Tail at 0° Engine Pylon
$WBT0N$	Wing Body Tail at 0° Nacelle
$WBT0NP$	Wing Body Tail at 0° Nacelle Pylon

Greek Symbols

κ_ψ	Installation coefficient for the effect of engine power setting
κ_σ	Installation coefficient for the effect of engine size
$\kappa_{position}$	Installation coefficient for the effect of engine position
λ	Independent variable in the proposed model
ϕ	Force in the drag domain
ρ	Density
θ	Force in the thrust domain
ϱ, Θ, ζ	The engine coordinate system - cylindrical
ξ, η, ζ	The engine coordinate system - cartesian

Roman Symbols

A_{ref}	Aircraft reference area
H	Altitude
$\Delta C_{D \text{ } inst AoA}$	Installation drag coefficient at constant angle of attack
$\Delta C_{D \text{ } inst}$	Installation drag coefficient at constant overall lift
$\Delta C_{L \text{ } inst AoA}$	Loss of lift coefficient at constant angle of attack
$C_{D \text{ } interf AoA}$	Interference drag coefficient at constant angle of attack
$C_{D \text{ } interf}$	Interference drag coefficient at constant overall lift
C_D	Drag coefficient
C_d^{BPD}	Bypass discharge coefficient
C_d^{Core}	Core discharge coefficient
c_{loc}	Local wing chord
C_L	Lift coefficient
C_p	Pressure coefficient

c_{ref}	Wing reference chord
C_V	Thrust coefficient
D	Drag
$D_{A/F}$	Airframe drag
D_{FF}	Engine fan-face diameter
D_{max}	Nacelle maximum diameter
D_{nac}	Nacelle drag
D_{pyl}	Pylon drag
D_{tot}	Total aircraft drag
dx, dy, dz	Engine displacement from the wing leading edge at an installation plane
D_{nac}^*	Modified nacelle drag
F_G	Gauge stream force
F_{inst}	Overall installation force
F_{interf}	Overall interference force
F_N	Standard net thrust
F_{ov}	Overall net thrust
F_N^*	Modified net thrust
g	Gravitational acceleration
L	Lift
L_{core}	External core-cowl length
L_{nac}	Nacelle length
$\dot{m}$	Massflow rate
M	Mach number
m	Mass
M_∞	Freestream Mach number
M_{isen}	Isentropic Mach number
Re	Reynolds number

T	Temperature
t	Time
V	Velocity
x', y', z'	The aerodynamic coordinate system - cartesian
x, y, z	The airframe coordinate system - cartesian

CHAPTER 1

Introduction

“If one begins all deeds well,
It is likely that they will end well too.”

Sophocles

1.1 Motivation

Based on a constant economic demand to reduce aircraft operating costs, it is required to improve aircraft overall performance. Combined with the legal requirements to reduce the engine emissions^{1,2} such as NO_x and CO_2 , the demand for higher aircraft efficiency will require further advancements in the design of airframe and propulsion systems³. In response to the requirement for the high values of engine propulsive efficiency, the by-pass ratio (BPR) of the turbofan engine tends to increase^{3,4}. Future turbofans are expected to operate with high BPR, low Fan Pressure Ratios (FPR) and as a result a reduction of the specific fuel consumption (SFC) is anticipated⁵. It is also expected that the BPR will increase from approximately 10 for the current Trent 1000 and GENx⁴ class engines to values of BPR between 14 and 21 for future turbofans⁶. The anticipated increase of BPR will cause an increase in fan diameter⁶ and engine size to accommodate the larger engine mass flow. However, with larger fan diameters there is an associated increase in nacelle drag, and therefore the installation drag penalty which can potentially outweigh the improvements in cycle performance⁷(Figure 1.1). Moreover, it is a challenge to separate the aerodynamic effects on the wing from the effects on the engine. To achieve the expected improvement in overall aircraft efficiency, it is necessary to ensure that the integration of the engine with the airframe is understood. Based on the current knowledge, the overall effect of engine installation is estimated to be in the order of 30 to 50 aircraft drag counts (dc) penalty per two engines⁸ for a typical podded under-wing engine configuration of a wide-body twin-engine aircraft. This penalty is sensitive to the relative engine size and position on the wing⁸⁻¹⁰ and the effect of engine size becomes

significant for larger diameter engines¹¹. The presence of a relatively larger engine can result in a flow acceleration on the lower side of the wing and eventually in the loss of lift¹¹. Due to the development of numerical tools and an increase in computational power, a series of AIAA Drag Prediction Workshops (DPW) were carried out, where the second workshop was dedicated to propulsion installation effects^{12,13}. The outcomes of the 2nd DPW¹² triggered the design of a more modern aircraft geometry that represents a typical 300 seater aircraft. The publication of substantial experimental datasets, with and without through-flow nacelles^{14,15}, makes the Common Research Model (CRM) a benchmark validation activity to assess the effects of installation. Within this broad context, the NASA CRM can be modified to include an under-wing turbofan engine and can be used for a preliminary assessment of engine installation effects for future turbofans. As a result, the preliminary design of aircraft could be aided by well informed decisions on engine cycle, engine size and airframe integration. Therefore, the assessment of installation effects for a range of engine position, sizes and power settings is required. The assessment of the cruise fuel burn should be considered to take into account the effect of the engine power setting and weight and determine the integrated cruise fuel burn.

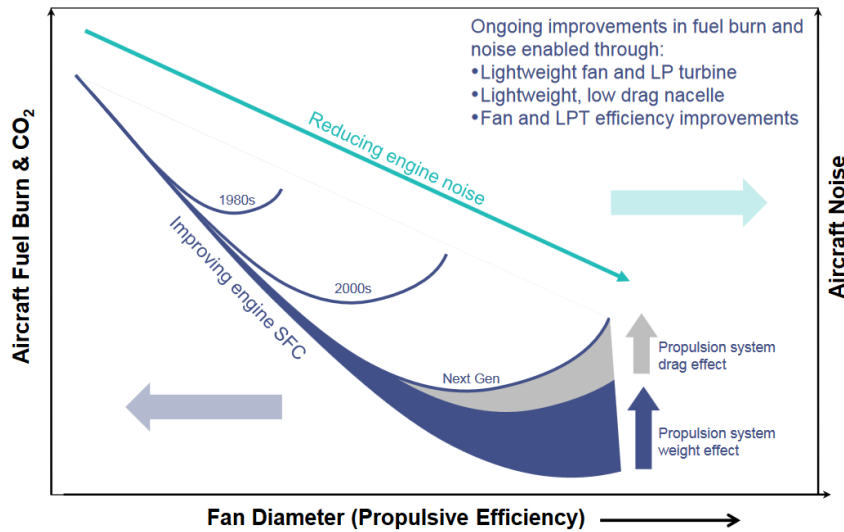


FIGURE 1.1: Detrimental installation effects outweighing the benefits of high by-pass ratio⁷

1.2 Aims and objectives

1.2.1 Research aim

The overall aim of this research was to develop methods for the assessment of aero-engine installation effects at cruise condition. The requirement was to quantify the sensitivity of aerodynamic effects to engine position, size and power setting. It was expected that an

evaluation of the integrated cruise fuel burn for prospective large by-pass ratio engines was completed for a commercial 300 seater aircraft. As a final aim, it was anticipated to propose a simple model for evaluation of the overall installation effect at a preliminary design stage for large turbofan engines.

1.2.2 Research objectives

A range of objectives have been identified to achieve the main research aim:

1. Develop an automated engine geometry generation tool for separate-exhaust configurations
2. Apply the thrust and drag accounting for separate-exhaust configurations
3. Develop a semi-automated framework for assessment of engine thrust and drag with means of CFD
4. Assess the impact of key installation parameters: engine size, engine position and engine power setting
5. Integrate the developed aerodynamic interference methods into a coherent drag assessment tool
6. Determine the integrated fuel burn over the cruise phase of flight of a typical commercial 300 seater for a range of engine configurations
7. Improve the thrust and drag assessment to determine the break-down of engine installation effect
8. Propose a simple prediction model for the effects of engine installation

CHAPTER 2

Background

“To know what you know
And what you do not know,
That is true knowledge.”

Confucius

The aim of this chapter is to provide the reader with a good understanding of the recent progress in the field of engine installation studies. The research was concentrated on a few aspects of the problem, namely: experimental and numerical methods of thrust and drag determination for isolated nacelles and engines (Section 2.1); the experimental (Section 2.2) and numerical (Section 2.3) methods for thrust and drag determination for the engine-aircraft configuration; drag and thrust book-keeping (Section 2.4).

2.1 Experimental studies of isolated engines

The understanding of the isolated engine aerodynamics is essential from the perspective of subsequent engine integration studies. Therefore, a range of previous experimental studies on the aerodynamic performance of isolated nacelles and nozzles were investigated to highlight the current state of knowledge in the field of nacelle design and testing.

2.1.1 Isolated Nacelles

A range of studies in the past were dedicated to the optimisation of the aerodynamic performance of nacelles to minimise drag, reduce spillage drag and maximise critical Mach number. Some of the initial wind tunnel experiments resulted in creation of the NACA A, B and C cowls which were designed to give the highest possible critical Mach number¹⁶. Furthermore, the NACA 1-series was developed as a result of these initial

tests¹⁷ to allow operation at higher freestream Mach numbers for a greater range of cowl dimensions. Based on the required cowl dimensions, the nacelle aeroline could be scaled by the required proportions such as maximum nacelle diameter, intake highlight diameter or the nacelle forebody length. However, further development of the engines required more complex designs to be developed for turbofan engines¹⁸. A greater interest was continuously placed on the off design characteristics of nacelles and reduction of spillage drag¹⁹. It was achieved by an increase in the lip radius (Figure 2.1) and decreased the curvature and acceleration over the nacelle forebody¹⁹. Furthermore, some more modern designs aimed to maintain the laminar flow regime over a significant portion of the nacelle forebody to result in a lower skin friction drag. As an example, the European Laminar Flow Investigation II (ELFIN II) and Laminar flow Research Action (LARA) projects developed laminar flow nacelles with transition positions at positions further downstream on the nacelle²⁰⁻²². An analysis of natural laminar flow (NLF) nacelles, with transition at 40% chord. A potential fuel burn reduction of 1.7% was demonstrated²³ by a test on a 3700km mission of a twin-engine aircraft with capacity of 110-130 passengers.

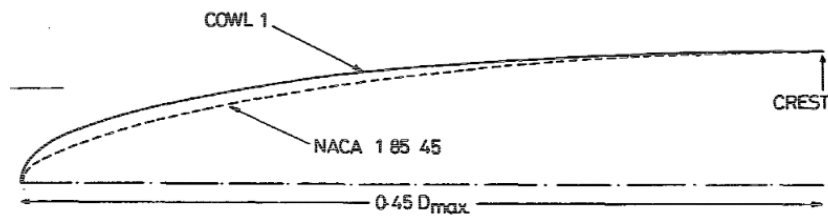


FIGURE 2.1: Comparison of NACA-1 cowl with an example of ARA¹⁹ Cowl 1

In the past, wind tunnel testing was the principal method to assess the aerodynamic performance of a nacelle in terms of drag. Experimental methods still remain of great pertinence to modern research due to the need for validation of numerical methods. A typical method to experimentally test nacelle geometries for gas turbine engines is the use of a sting mounted model. To enable the effects of mass flow capture ratio (MFCR), freestream Mach number, and incidence on the nacelle drag to be measured, the intake mass flow is controlled with use of the suction vent in the nacelle intake^{22,24,25}. The control of intake massflow affects the ingested streamtube and determines the aerodynamic condition on the nacelle. As a result, an assessment of a nacelle design in terms of an absolute drag value and trends in terms of transonic drag rise and spillage were investigated^{22,24,26}. Initially, the strain-gauge method was commonly used to measure force which acts on a wind tunnel model. However, due to the low order of magnitude of drag force, it was found to be hard to accurately extract drag with the use of a strain-gauge. Also, the order of magnitude of the drag of the nacelle is comparable with the drag of the sting on which the nacelle is mounted²⁷. As an alternative, the wake integral method²⁶ can be used. The wake integral method²⁶ directly quantifies the momentum loss caused by the model through pitot-static measurements taken with a rake positioned in the nacelle wake.

In this context, the work of Langley et al.¹⁹ was considered to be pertinent to current research. A range of nacelle geometries were designed to assess alternatives to the conventional forebody designs like NACA series 1 type¹⁹. It was an axisymmetric design with a cylindrical centrebody. The afterbody was defined by a simple conical section at a chosen boattail angle and the intermediate section defined by a circular arc profile to link the centrebody with the conical section of the afterbody. Even though this corresponds to a relatively old nacelle for an engine with a smaller bypass ratio, the work is vital due to the different forebody designs and the range of test conditions which were investigated. The cowls have been tested over a range of Mach numbers from $M_\infty=0.4$ to 0.9 and $MFCR$ from 0.4 to 0.8. During the tests and depending on the operating condition the Reynolds number based on the maximum diameter ranged from 1.5×10^6 to 2.4×10^6 . The geometries were tested at ARA on the ARA B5 rig in the 9'x8' transonic wind tunnel²⁴. As a result, a publicly available dataset suitable for the validation of numerical methods in assessment of nacelle drag was created. In summary of the experimental investigation for isolated nacelles, the ARA Cowls²⁴ are suitable for the development of a numerical strategy to determine the nacelle drag characteristics.

2.1.2 Nozzles and Turbine Powered Simulators

A review of the experimental tests on the performance of nozzles and turbine powered simulators was completed to determine a suitable validation test case for thrust determination of a separate-jet exhaust system.

In the field of simple conical nozzles, a range of afterbodies was tested experimentally²⁸ for a range of freestream Mach numbers from 0.6 to 1.2 to determine the effect of high velocity jet on the afterbody aerodynamics and the sensitivity of the interaction to the boat-tail angle of the afterbody. The boat-tail angle was increased from 0° to 20° . Moreover, a range of nozzle pressure ratios was tested and it was determined that “generally, boat-tail axial force continually decreased with increasing pressure ratio above jet exit pressure ratio of about 1.0”²⁸. To test the effect for different boat-tail angles the jet exit pressure ratio was kept constant and it was determined that “configurations with the larger boat-tail angles generally derived more benefit from the jet interference, compared with the jet-off conditions, than those with smaller boat-tail angles”²⁸. However, some further studies²⁹ that also tested single conical nozzles with boattail angles of 10° and 20° for Mach numbers from 0.6 to 1.3 which indicated that a further increase in nozzle pressure ratio may not be beneficial. For the increased nozzle pressure ratios and nozzle exit Mach numbers of 1 and 2, the highest afterbody drag was reported for high boattail angles and high transonic freestream Mach numbers. The mechanism of the detrimental interference (Figure 2.2) is driven by flow separation that can occur on the afterbody due to the shape of the jet plume.

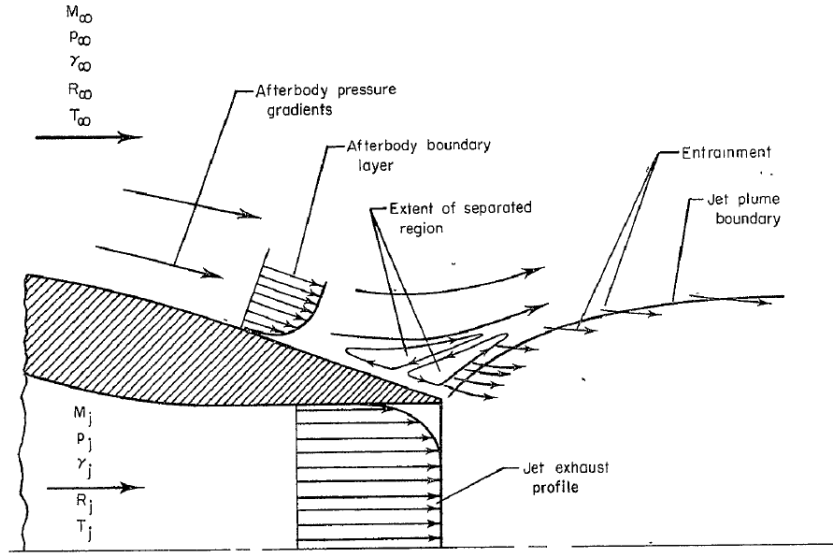


FIGURE 2.2: Jet interference mechanism for the conical afterbody²⁹

The tests of the separate exhaust turbine powered simulators (TPS) are the most relevant in the context of research on modern turbofan engines. The operation of a choked fan nozzle usually results in a supersonic flow with expansion and compression fans. An experimental campaign³⁰ for the separate exhaust engine (JT9D) at $M=0.9$ and with $FNPR=2.6$ resulted in determination of the pressure distribution on the fan-cowl, core-cowl and plug. Due to the fan of expansion shocks the pressure distribution indicates the areas of the core-cowl exposed to either forward or rearward oriented pressure force depending on the shocks location. Furthermore, a Navier-Stokes computation³⁰ showed a relatively good agreement with the experiment in terms of the C_p distribution (Figure 2.3).

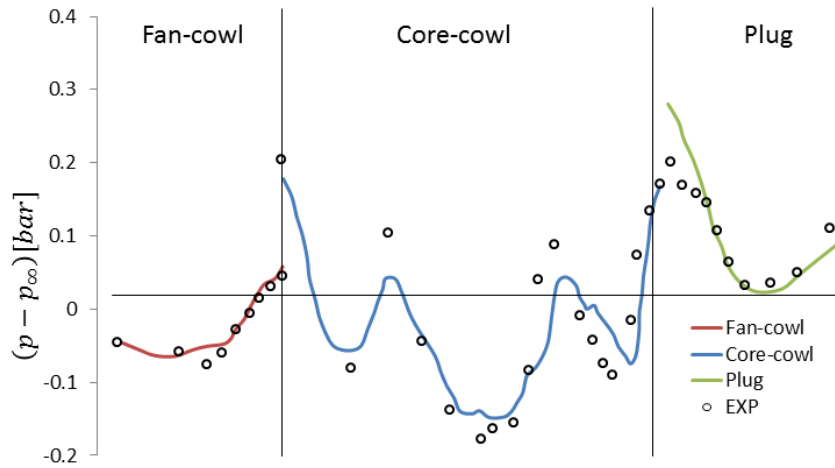


FIGURE 2.3: Pressure distribution on the fan-cowl, core-cowl and plug of JT9D derived from Navier-Stokes calculation compared with the measurements; based on Putnam et al.³⁰

A comparable experimental test of a separate exhaust engine was performed³¹ for the exhaust of RB211 in model scale of 1:22 (Figure 2.4). The exhaust was tested with a subscale model with the Reynolds number 6 times lower than the desired conditions with freestream Mach number of 0.85 at altitude of 10668 m. To determine the overall performance of the exhaust system the balance measurement for the thrust force was performed. As compared with the ideal thrust of the engine, the measured thrust of the exhaust system reduced (Figure 2.5) due to the increase of the freestream Mach number. However, the effect of thrust reduction due to freestream Mach number is not explained. The measurement of the pressure distribution of the core-cowl (Figure 2.6) were compared with the predictions based on the method of characteristics. The measurements indicated presence of both forward and rearward oriented pressure forces exerted on the core-cowl as a result of expansion and compression pattern in the jet. Even though the key features of the flow were captured, a notable difference between the prediction and the result was observed beyond 0.5 of the core-cowl length. Moreover, various types of core-cowls for short-fan-cowl engines were tested and some benefits are found by direction of the by-pass flow radially inwards towards the engine axis prior to the flow leaving the fan nozzle. Also the use of high angles for the conical design of the core-cowl was advised and a successful application of conical core-cowl with a boat-tail angle of 15° was implemented in commercial engines³¹.

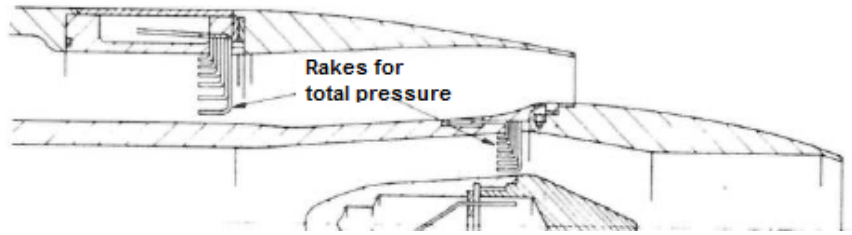


FIGURE 2.4: Experimental setup for the RB211 separate exhaust³¹

Some publicly available work in the field was identified that can be used for the validation of numerical methods in the field of separate exhaust nozzles. A turbine powered simulator with an axisymmetric nacelle based on the GE CF6-50C2 engine was tested experimentally^{32,33}. The highest value of the freestream Mach number was 0.17 over a range of fan pressure ratios (FPR) from 1.2 to 1.6, which corresponds to fan nozzle pressure ratios from 1.21 to 1.64. The pressure measurement from the tapping along the nacelle, core-cowl and plug surfaces were presented as isentropic Mach numbers (Figure 2.7), where freestream total pressure, fan nozzle total pressure and core nozzle total pressure were used as the respective reference values.

With an increasing need for validation of numerical methods, the Dual Separate Flow Reference Nozzle (DSFRN) was designed^{34,35} as a separate-jet nozzle representative of commercial turbofan engines with a BPR in the order of 10 to 12. It is the most relevant publicly available geometry with an open-domain experimental dataset. The model

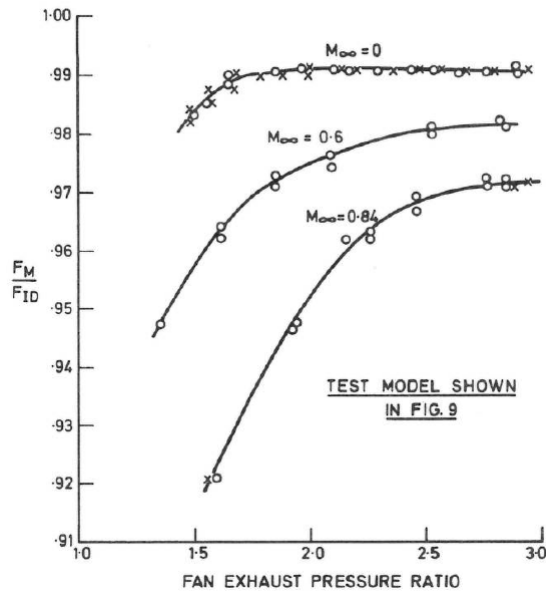


FIGURE 2.5: Overall thrust force (F_M) for the RB211 model referenced to the idealised gross thrust of the exhaust (F_{ID}) as a function of fan exhaust pressure ratio³¹

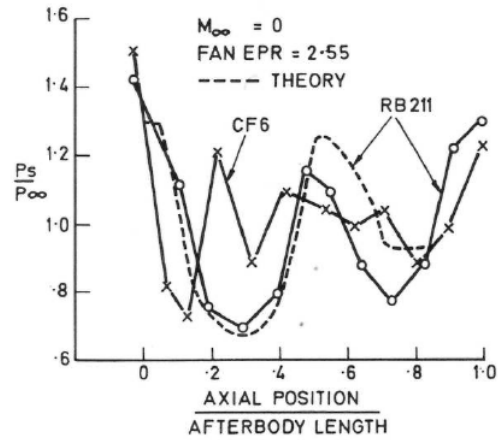


FIGURE 2.6: The experimental pressure distribution for the RB211 and CF6 core-cowls compared with the modified method of characteristics for RB211 ("theory")³¹

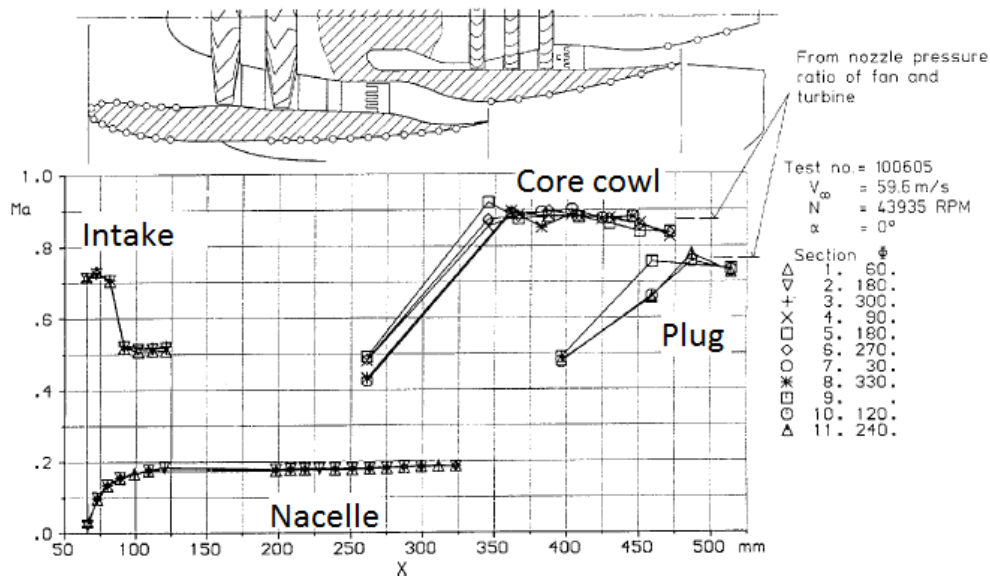


FIGURE 2.7: Isentropic Mach number distributions on the nacelle, core-cowl and plug of the DLR-TPS model³²

consisted of two flow streams (Figure 2.8), one to simulate the flow from a fan stream and one to simulate the flow from the core engine flow. The fan stream contained two bifurcation sections and the core duct contained a plug with support struts. In addition, both ducts have convergent nozzles at the duct exit. A pylon was present and attached to the upper surface of the nacelle. Only the fan stream flows over the pylon and the core stream discharges onto the heat shield underneath the pylon. The air flow upstream was

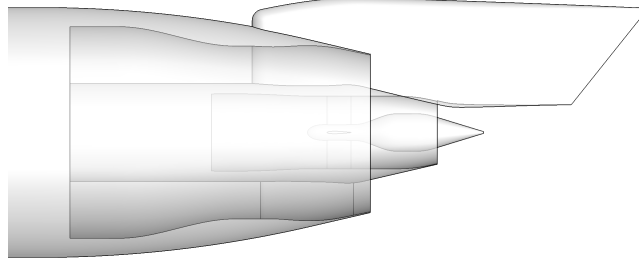


FIGURE 2.8: The geometry of the Dual Separate Flow Reference Nozzle; based on Mikkelsen et al.³⁴

throttled, metered and split into the core and fan streams with the mass flow through the core measured to determine the mass flow split. The thrust generated from the nozzle was measured via a three-component strain gauge force balance³⁴. To obtain an experimental nozzle performance characteristic for the DSFRN, the fan nozzle pressure ratio (FNPR) and core nozzle pressure ratio CNPR were varied. The DSFRN discharged into a quiescent atmosphere with a constant ambient pressure. The nozzle pressure ratio (NPR) for both streams was varied via the inlet total pressure of the respective stream. A total of eight FNPRs were investigated, and the ejection ratio (ER) between the core and fan total pressure kept at a constant value of 1.2 throughout. The total temperature at the inlet of both the core and fan streams was 294K, and remained constant across the range of the FNPRs³⁴ (Table 2.1). The increase of FNPR caused a linear increase of

<i>FNPR</i>	1.2	1.4	1.6	1.8	2	2.2	2.4	2.6	2.8
<i>CNPR</i>	1.2	1.17	1.33	1.5	1.67	1.83	2	2.17	2.33
<i>ER</i>	1.0	1.2	1.2	1.2	1.2	1.2	1.2	1.2	1.2
T_∞	294K								
M_∞	0								
p_∞	98181 Pa								

TABLE 2.1: Summary of nozzle operating conditions for DSFRN computations³⁴

discharge coefficient of the by-pass duct (C_d^{BPD}) from 0.962 for FNPR=1.2 to 0.987 for FNPR=2.0 and the further increase of FNPR resulted in choking the nozzle and the value of discharge coefficient approached asymptotically the value of 0.9895 for FNPR=2.8. Meanwhile, the increase of CNPR causes a approximately linear increase in the discharge coefficient of the core duct (C_d^{Core}) from 0.805 at CNPR=1.17 to 0.94 at CNPR=1.83. A

further increase in CNPR was mitigated by the suppression effect from the by-pass flow and a constant value of $C_d^{Core} = 0.94$ is reported for CNPR of 2.0 and 2.2. Eventually, the increase of CNPR to 2.4 resulted in an increase of C_d^{Core} to 0.965. Moreover, an additional case with FNPR of 1.2 and ER of 1.0 was tested and it was determined that the high ER has no effect on the discharge coefficient of the by-pass duct (C_d^{BPD}), but significantly affected the discharge coefficient of the core duct with $\Delta C_d^{Core} = -0.08$.

Overall, based on the review of experimental studies for nozzles and turbine powered simulators, it can be stated that the Dual Separate Flow Reference Nozzle³⁴ is the most notable validation case and should be adopted for validation in the current study. In addition to the DSFRN, the DLR-TPS case^{32,33} can be considered to ensure that the developed strategy is case independent.

2.2 Experimental studies of propulsion system integration

The installation of the engine on the airframe notably affects the aircraft performance in terms of lift and drag. One of the techniques to determine the impact of the engine on the airframe is to install through-flow nacelles and to measure the impact on the aircraft lift-drag polar (Section 2.2.1). However, this technique does not take into account the impact of the jet on the wing aerodynamics, as well as the impact of the wing on the engine thrust. To fully address the need for quantification of the jet-airframe interaction, the turbine powered simulator technique was developed (Section 2.2.2). Both the through-flow nacelle and the turbine powered simulator are in common use and provide a valuable source of information about installation effects.

2.2.1 Through flow nacelles

A through-flow nacelle is an external geometry of the fan-cowl, which is designed to represent the engine installation. A through-flow nacelle is unpowered and has sized exit to provide an intake massflow capture ratio representative of typical cruise condition. The installation of through-flow nacelles on the aircraft and evaluation of the overall vehicle performance is a common experimental technique to determine the suitable position of the powerplant for the aircraft. The use of a through-flow nacelle simplifies the analysis, as aircraft drag with a through-flow nacelle can be read directly from the aerodynamic balance. Therefore a direct comparison between airframe-nacelle and clean-wing configurations allows for the direct derivation of the installation drag term.

From a historical perspective, early studies concentrated on the search for an optimal position of the engine on the airframe. A great variety of nacelle locations were considered in the past. Some studies investigated the best position for the rear-mounted nacelles³⁶. Whereas, other studies^{37,38} concentrated on the podded configurations for high-wing

aircraft. As well as frontward podded under wing nacelles, some other concepts were investigated such as wing-rearward mounted (Figure 2.9). Overall, the D-shaped nacelles provided the best results for aft-mounted configuration³⁸ and were more efficient than forward-mounted nacelles considered.

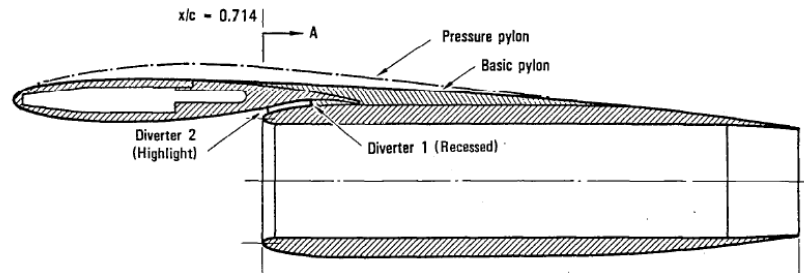


FIGURE 2.9: An example of aft-mounted nacelle³⁸

From the modern perspective, the frontward wing-mounted configurations are of greatest relevance. The magnitude of the overall installation effect for a frontward wing-mounted nacelle was found to be approximately 50 dc³⁹ for a typical high-wing transport aircraft at $M=0.80$. Furthermore, the effect notably depended on the chordwise and spanwise engine position of the nacelle³⁹. Up to 20dc difference in the interference drag was reported (Figure 2.10), for the different chordwise positions at cruise at Mach number 0.8 with a benefit for moving the nacelles axially in the upstream direction. Additionally, the spanwise positioning of the nacelle was reported to be the second main driver of interference drag and the overall magnitude for the in-board relocation was 10 dc (Figure 2.10). A broadly similar study⁴⁰ for a high-wing transport aircraft with through-flow nacelles was performed at Mach number of 0.8. It was also determined that the movement of the nacelles upstream from the wing offers interference improvements. The installation drag decreases by approximately 10dc⁴⁰.

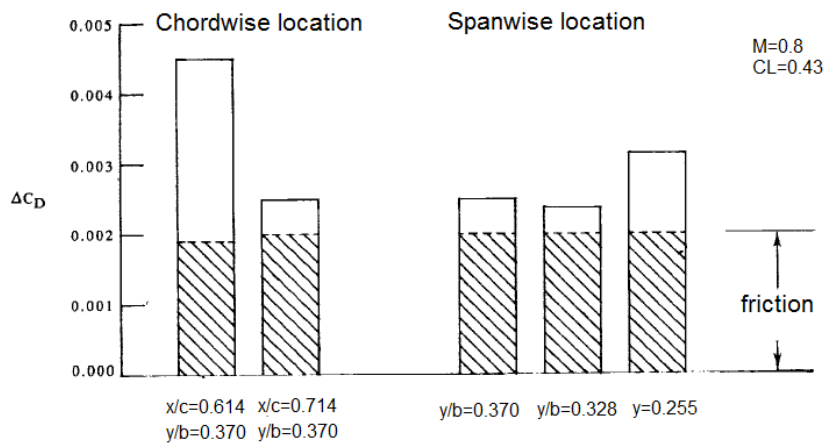


FIGURE 2.10: Impact of chordwise and spanwise locations; subscale model based on³⁹

In addition to the chordwise and spanwise engine position, the engine angular position is of interest. Different nacelle toe-in angles between 0° and 4° and cant angles between 0° and 3° were investigated³⁷ on a low-wing frontward podded configuration over a broad range of Mach numbers (0.3 to 1.0). A negligible effect of the engine angular position was reported³⁷. Nevertheless, other studies⁴¹ of toe-in angle might be beneficial. The increase of toe-in angle resulted in the rotation of the drag polar and a consecutive decrease of drag for greater lift coefficients. However, tailoring of the toe-in angle offers relatively small reductions of drag³⁹ by approximately 3 dc. The determination of the toe-in angle is considered to be an element of the final detailed design of the installation, due to the relatively modest sensitivity.

Furthermore, the mutual relation between natural laminar flow (NLF) nacelles and installation effects was considered⁴⁰. The study was conducted for a high-wing transport aircraft with through-flow nacelles at Mach number of 0.8. The difference between NLF cowl and fixed-transition cowl was assessed to be in order of magnitude of 10dc (Figure 2.11). Overall, no negative influence of the installation on the desired laminar flow was observed.

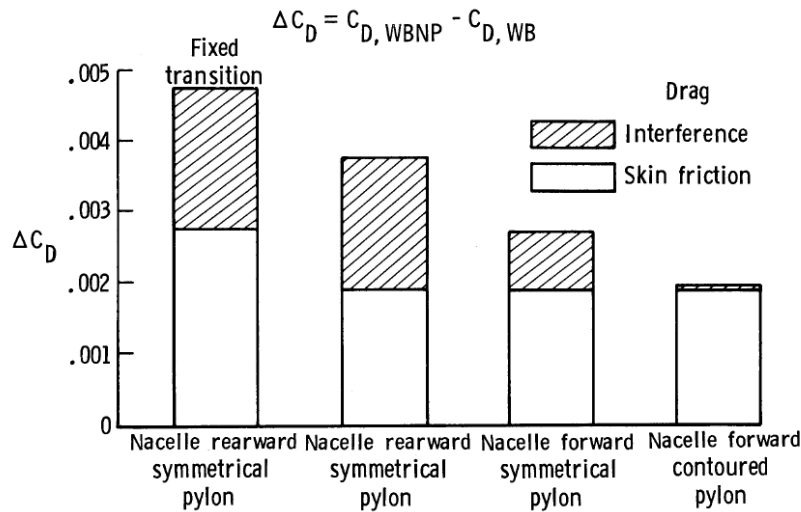


FIGURE 2.11: NLF nacelle installation effects; $M=0.85$ subscale high-wing transport aircraft model⁴⁰

The presence of the pylon can affect the magnitude of installation effect³⁷ on a low-wing airframe configuration. Different pylons were chosen for the nacelles investigated in the study³⁷ and it was concluded that inappropriate contouring of the pylon results in 5dc additional interference drag. Similarly, Lee et al.⁴² investigated the installation effect due to the pylon for a high-wing airframe configuration at flight Mach 0.8. The drag increase attributed to the presence of the pylon was 6dc. Moreover, it was concluded that a reduction of the curvature at the nose of a pylon helps alleviating the negative installation effects. Similarly, Lamb et al.⁴⁰ suggested that tailoring the pylon resulted

in 5dc reduction in aerodynamic interference terms. A study by Carlson et al.⁴³ concentrated on the flow physics of the nacelle pylon integration. Both swept and unswept pylons were considered⁴³ on the unswept prismatic wing in transonic flow. The study indicates that inappropriate shape of the pylon can result in a pylon induced flow separation in a transonic flow. Based on the reviewed literature, the presence of the pylon is anticipated to result in a penalty of 5 to 10dc within the present study.

2.2.1.1 DLR-F6 model

The DLR-F6 is a subscale model of a low-wing subsonic transport aircraft⁴⁴ (Figure 2.12). The design Mach number for the model is 0.75 and design lift coefficient is $C_L = 0.50$. DLR-F6 became a benchmark case for the validation of numerical methods, due to its public availability and the use in the 2nd Drag Prediction Workshop¹². The tests of the model with and without through-flow nacelles allowed for the determination of the engine installations effects such as increase in aircraft total drag and reduction of total lift. Two experimental campaigns were conducted namely one in the ONERA wind tunnels¹³, followed by the additional campaign at the NASA National Transonic Facility⁴⁴. In both cases the overall model lift, drag and pitching moment were obtained with the use of a strain gauge balance. The model has a set of 288 pressure orifices located at 8 span-wise sections. This allowed information about the pressure distribution on the wing. Similarly, 47 orifices are located in 3 azimuthal sections of the nacelle: 60°, 180°, 360°. Additionally, 9 locations have been selected for pressure measurement on the pylon structure. The DLR-F6 model is a relatively good source of data for the studies on the changes to lift and drag due to the installation of through-flow nacelles.

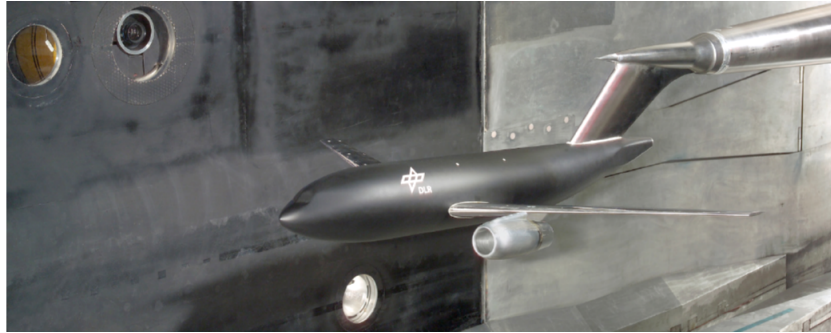


FIGURE 2.12: A DLR-F6 model mounted in wind tunnel of NASA NTF⁴⁴ with the through-flow nacelle of CFM-56 with the long cowl

Based on the experimental campaigns^{8,45}, the typical installation penalty for a CFM56 through-flow nacelle on the DLR-F6 is approximately +30dc at $C_L = 0.5$ (Figure 2.13). The installation effect was tested over a range of incidence and it was mostly constant over the range of lift coefficients (C_L) from 0.3 to 0.6. The installation drag was sensitive to engine horizontal position with a maximum benefit of -8dc for the movement of the nacelle upstream from the wing by approximately $dx/c_{loc} = 0.2$ at $C_L = 0.5$. The

sensitivity of the installation effect to the vertical engine position was also tested. The movement of the engine downward from the wing by approximately $dz/c_{loc} = 0.05$ at $C_L = 0.5$ resulted in a benefit of -2dc. The experimental accuracy was determined to be 1dc within a single test⁴⁵. However, a discrepancy of 7dc was noted between different experimental campaigns⁴⁵.

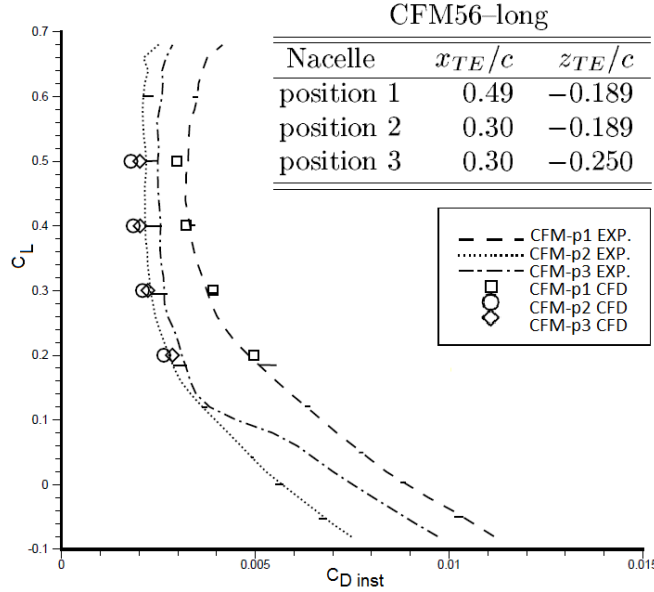


FIGURE 2.13: The installation drag ($C_{D\ inst}$) for the three configurations with CFM-56 long cowl⁴⁵. The experimental results at $M_\infty = 0.75$ compared with computations

2.2.1.2 A Common Research Model

The CRM model has been developed by NASA with the help of interested third parties⁴⁶. It was an answer to the arising need for a contemporary-subsonic-transport-aircraft model for CFD simulations and wind tunnel experiments. The model was tested in two different facilities in order to compare the experimental results. This became a basis for the validation of numerical simulations. The geometry has been designed to represent a vehicle of nominal cruise Mach number of 0.85 and nominal lift coefficient of $C_L = 0.50$ at a Reynolds number of 40 million per reference chord. The full model description was provided by Vassberg et al.⁴⁶ In the current project, it was chosen to compare the CRM to the Airbus A330, which is the most relevant existing airplane (Figure 2.14). This analogy between A330 and CRM makes CRM a valuable tool for studies of aerodynamic performance of a relatively modern airframe.

A series of experiments were conducted with the use of various CRM configurations. The clean wing configuration denoted as Wing Body (WB) and it has been used as a baseline for all the investigations. The effects of empennage on the aircraft aerodynamics have been assessed by investigating WBT0, WBT+2 and WBT-2, where the last number represents the inclination of tailplane¹⁴. From the perspective of the aerospace-propulsion

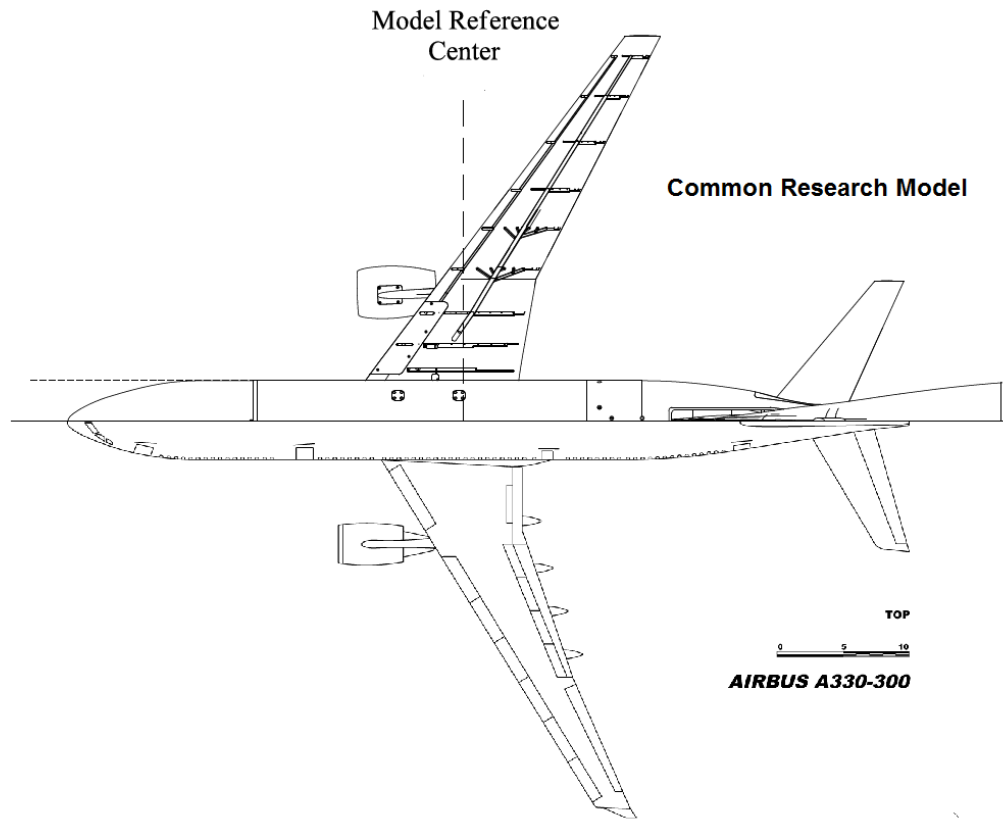


FIGURE 2.14: A comparison between Airbus A330 and the CRM; based on Vassberg et al.⁴⁶ and Airbus Recovery Manual⁴⁷

studies, the comparison of clean wing configuration (WB) against the Wing/Body/Nacelle/Pylon (WBNP)¹⁵ is the most interesting part of the CRM studies. The nacelles were designed as simple through-flow nacelles which were mounted under the wing with the use of adequate pylons⁴⁶. The experimental data for WB and WBNP configuration is available for 3 Reynolds numbers, based on the aerodynamic chord of the subscale model: 5, 19.8, 30 million. The $Re = 5 \times 10^6$ was the nominal case with most of the available data. Notably smaller dataset for larger Reynolds numbers of 19.8×10^6 and 30×10^6 were obtained¹⁵ to investigate the effect of Reynolds number. From the perspective of the current study the comparison between WBNP and WB configurations was of primary interest, and therefore the attention was dedicated to determination of the drag and lift effects due to the installation of nacelles on the CRM.

Based on the experimental results^{14,15}, the typical installation penalty for a through flow nacelle was approximately 25dc, as measured at $C_L = 0.5$ and $M_\infty = 0.85$. The reduction of Mach number to $M=0.83$ had a modest effect on the installation drag with a typical penalty of 25dc at $C_L = 0.5$. For both Mach numbers, the installation penalty

is mostly constant within the range of lift coefficients from 0.2 to 0.5. The further increase of aircraft incidence results in a sudden reduction of the installation penalty at constant lift. The reported repeatability for airframe drag ($C_{D\ tot}$) within the tests is within two standard deviations of $\pm 2dc$ for a WB configuration tested at $M_\infty = 0.85$ and $Re = 5 \times 10^6$.

2.2.2 Turbine powered simulators

Since the through-flow nacelle technique cannot address the effect of the jet on the airframe, it is desirable to represent the engine by ejection of air flow at the back of the nozzle to create similar efflux condition. In the simplest approach to propel cold air through the simulator a rotating fan can be installed within the nacelle^{48,49}. Although the jet is cold, the fan allows for the control of the nozzle pressure ratio. The impact of the nozzle pressure ratio on the external cowl and its drag can be assessed. For example, some tests were completed for a under-wing podded nacelle (Figure 2.15) on a high-wing transport aircraft model⁴⁸. The test was conducted for a NACA-1 external cowl sized to represent turbofan engine with a mass flow capture ratio of 0.7 operated at Mach number between 0.6 and 0.8. The through-flow nacelle geometry was compared with the configuration with dual cold stream (Figure 2.15). The intake flow was redirected to the core nozzle, whereas a fan present in the by-pass duct propelled the air provided from the separate feed through the pylon. The fan was operated to provide the jet expansion rate set to 1.5, the presence of the jet caused an increase of the suction peak at 25% of wing chord. As a result, a relatively strong un-swept shock appeared. However, the impact on the overall aircraft drag was not quantified.

A broadly similar study of a cold-jet was carried out in ONERA wind tunnels with the use of SNECMA nacelles⁴⁹. Unfortunately, only an extract of the conducted research is publicly available. Three types of nacelles were used, namely short cowl and long cowl with a single cold jet, and a short cowl with two separate cold jets. Jet interference was reported to have an insignificant influence on drag terms. The increase of jet expansion from 1.0 to 2.0 resulted in relatively modest difference of a few drag counts (dc). A moderate influence on the lift coefficient was present. The lift coefficient loss for different cases varied between 0.02 and 0.05. Loss of lift was observed through notable changes in pressure distributions on the wing at the spanwise location of the nacelle. During the tests, the nacelles were mounted on struts, therefore no effect of the pylon was assessed for these configurations.

The best known way to represent the realistic efflux of a turbofan engine in the wind tunnel is to mount a micro gas turbine under the model aircraft. The method is called the turbine powered simulators (TPS) technique. Even though not every similarity criterion is satisfied, it still has been proven to be the most reliable method. Functionally, the TPS units consist of a fan in the inlet and a turbine to propel it. Pressurized cold air

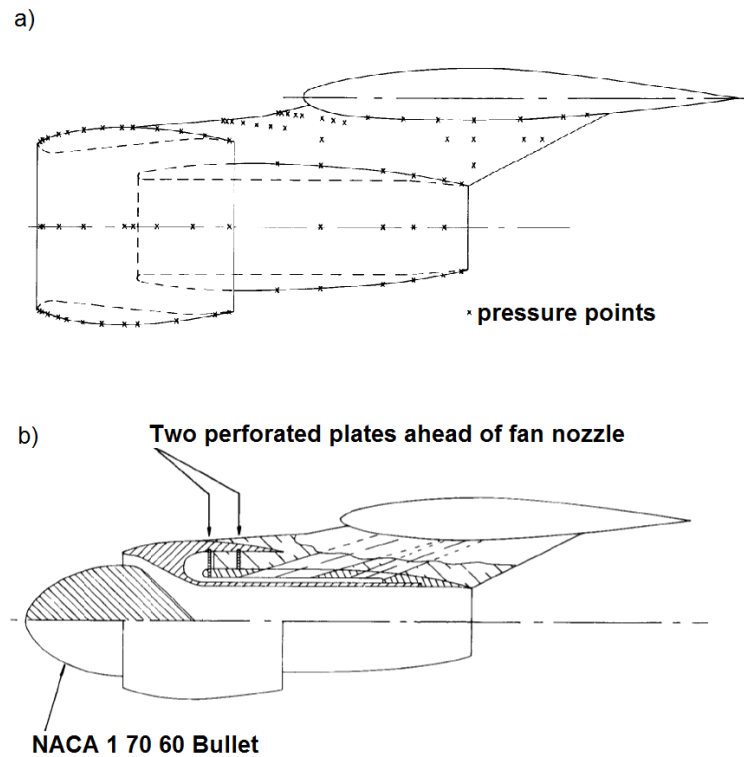


FIGURE 2.15: a) Through-flow nacelle compared with b) cold-jets simulator⁴⁸

is typically fed internally through the airframe model to the TPS turbine and the core flow is independent of the external fan flow. The use of TPS allows for determination of the installation drag penalty, as well as quantification of effects on the aerodynamics of the model scale aircraft⁵⁰ that are dependent on the engine power setting. For example, Wells et al.⁵⁰ tested an over-wing-nacelle configuration for a range of Mach numbers from 0.6 to 0.78. It was reported that favorable aerodynamic interference can be achieved, as the pressure distribution on the suction surface was altered and resulted in weaker upstream displaced shock. Moreover, the jet effects were assessed to be between 6% and 8% of drag value for the wing-body configuration. The effect of the jet was directly related to the fan pressure ratio (FNPR) used and the increase of fan pressure ratio caused an increase of the overall installation drag.

Similar TPS technique was also applied within other study⁵¹ for the tests at low speeds ($M_\infty = 0.18$) with the high lift devices deployed. A high-lift airframe broadly similar to the A340 was investigated. Even though the flow conditions are substantially different for the considered configuration, it is important to note that the vertical movement of the engine had a negligible effect on the drag. However, the close-coupling of the TPS to the airframe affected the lift substantially and caused a greater loss of lift due to the reduced gap between the nacelle and the inboard slat. In addition, the effect of the jet induced interference increased from 15dc to 20dc for the close-coupled position.

2.2.2.1 ALVAST

As part of DUPRIN and ENIFAIR projects⁵², the tests of a relatively modern airframe with the TPS were carried out. The ALVAST is an airframe broadly similar to a A320 aircraft in 1:10 scale⁵³. It was fitted with three sizes of turbine powered simulators also in 1:10 scale⁵³ (Figure 2.16). The engines fan diameters were 0.163m 0.198m and 0.254m for the turbofan simulator (TF), very high by-pass ratio simulator (VHBR) and ultra-high by-pass ratio simulator (UHBR) respectively and with corresponding by-pass ratios⁵³ of 5, 9.2 and 15.7. Prior to engine installation, the favourable engine locations were determined through the use of an Euler code⁵⁴. In the context of the current work, the ALVAST is the most pertinent publicly available dataset, due to the relatively modern design of the airframe and the use of separate-jet exhaust TPS units. However, the geometry of ALVAST is not publicly available to re-create the study.



FIGURE 2.16: The comparison of ALVAST airframe⁵⁴ with the turbofan simulator (TF), very high by-pass ratio simulator (VHBR) and ultra-high by-pass ratio simulator (UHBR).

The engine thrust calibration was carried out to determine the relation between the thrust and simulator rotational speed⁵⁵. Furthermore, a test campaign in the S1MA wind tunnel⁵⁵ was carried out for Mach numbers 0.34, 0.6, 0.7 and 0.75 for lift coefficients from 0.3 to 0.65. In total four power settings were considered from a “through flow condition” to the maximum power setting available⁵⁵. Overall, the discrepancies in thrust coefficient (C_V) corresponded to 1dc on the ALVAST model and were considered satisfactory⁵⁵. The discrepancies in the measurement of the static pressure⁵⁵ with the use of pressure taps on the surfaces were less than $\Delta C_p = 0.01$.

The results at $M=0.75$ with “start of cruise” power setting are of greatest relevance with fan pressure ratios of 1.58, 1.44 and 1.25 for TF, VHBR and UHBR respectively⁵⁴. The nozzles of TF and VHBR operated with supercritical fan nozzle pressure ratio (FNPR)^{53,54}. The total aircraft drag ($C_{D_{tot}}$) for each configuration is a product subtraction of respective engine thrust (F_N^*) from the force measured on the balance in

the wind tunnel ($F_{balance}$)^{53,55}. The value of engine thrust was known from the calibration⁵⁵, however, a correction for the effect of the freestream Mach number is applied to account for the difference in base pressure⁵³. Overall, the thrust contains all the beneficial forces and scrubbing losses in the jet downstream of the critical cross-section⁵³. It was stated that the pressure ($F_{x\ pres}$) and viscous ($F_{x\ visc}$) forces on the core-cowl depend on the shape of the core⁵⁴, however, no optimisation of the shape of the core-cowl was undertaken⁵³. As the last step, the installation drag of the configuration was determined⁵⁵.

During the tests, the overall installation drag ($C_{D\ inst}$) of 18dc⁵² was determined for the configuration with the turbofan (TF, $BPR = 5$). The effect for ultrahigh-by-pass ratio engine was more severe with approximately 35dc (around 12% of clean-wing aircraft drag). An evaluation of engine power setting was undertaken between the “through flow configuration” and the “start of cruise” configuration⁵⁴. The difference ($C_{D,J}$) between the “start of cruise” and the “through flow configuration” was more significant for greater by-pass ratios (Figure 2.17). It was evaluated that the $C_{D,J}$ of 8dc for the regular by-pass ratio engine ($BPR = 5$) was doubled for ultrahigh-by-pass configuration ($BPR = 15$). The increase in $C_{D,J}$ is explained as between the configurations “the massflow change [and the concurrent reduction of MFCR] influences in particular the intake lip spillage drag, [which is] the main portion of the jet interference drag.”⁵⁴.

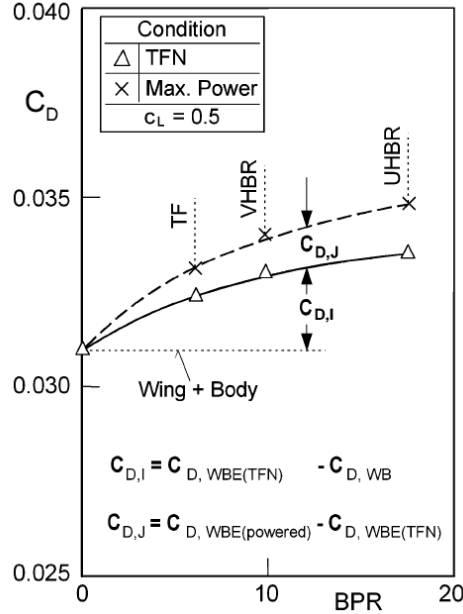


FIGURE 2.17: Installation drag ($C_{D,I}$) and jet interference ($C_{D,J}$) in relation to by-pass ratio; $M_\infty = 0.75$; subscale⁵⁴

2.3 Numerical studies of propulsion system integration

This section highlights the precious experience with the use of numerical methods in engine integration studies. The aim of the section was to assess the accuracy of available methods, present the current state of art, and to provide a list of considerations that have resulted from previous studies.

2.3.1 Lower fidelity Computational Fluid Dynamics

The three dimensional (3D) nature of the engine-airframe integration requires that only 3D computational methods are considered. Amongst one of the simplest are panel methods, which consider 3D geometry represented by surface meshes. However, such panel methods are not capable of calculating the non-linear flow features present at transonic speeds. Therefore, full potential solvers became of interest. A variety of full potential codes were developed with the most notable Boeing's Tranair⁵⁶. Even though, no viscosity and no vorticity effects can be taken into account, a great advantage in computational cost makes codes like Tranair still pertinent. For example, Gisin et al.⁵⁷ used Tranair in conjunction with the ISES viscous boundary layer code to compute a parametric study of a prismatic wing and over-wing nacelle. The configuration was tested at $C_L = 0.4$ and $M_\infty = 0.75$ for different axial and vertical nacelle positions and it was observed that the drag was predominantly sensitive to the axial position.

To increase the computational fidelity, the next step was to use the Euler equations, which provided more accurate solutions of transonic flows. The accuracy of the Euler methods for the prediction of the installation effects was tested⁵⁸ against the available dataset for the DLR-F6 geometry from the ONERA wind tunnel tests (Section 2.2.1.1). The computation were compared with the experimental data for the cruise condition at $M_\infty = 0.75$, $AoA = 0.98^\circ$ and $C_L \approx 0.5$, $Re = 3 \times 10^6$. It has been reported (Figure 2.18) that the Euler computation captured the main flow features of the flow such as “correct shock location for inviscid flow, vortex sheets behind the wing as well as the development of the inviscid jet”⁵⁸. However, the Euler computations over-predicted the lift coefficient at a constant angle of attack as compared with the experiment. The over-prediction was explained as “the missing de-cambering effect of the boundary layer”⁵⁸. The detailed aerodynamic effects on the wing in terms of the pressure distribution were broadly captured. As a result, the peak suction at the leading edge on the upper side of the wing was under-predicted by $\Delta C_p = +0.2$, which was followed by the over-prediction of the second peak of suction on upper side by $\Delta C_p = -0.3$. At the same time, the peak acceleration on the lower side of the wing at the leading edge was under-predicted by $\Delta C_p = +0.2$. To accommodate the viscosity effects in the boundary layer, the Euler code was coupled with a 3D boundary layer code to provide a correction for the displacement thickness in the boundary layer. However, the computational effort

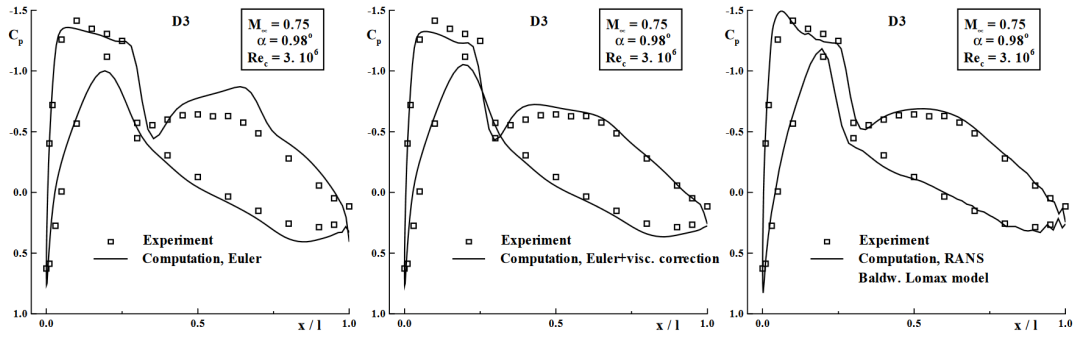


FIGURE 2.18: Pressure distribution (C_p) at the wing cross-section inboard of the nacelle at DLR-F6 configuration⁵⁸; $M_\infty = 0.75$; $C_L \approx 0.5$; $Re = 3 \times 10^6$

increased by 2-3 times. As a result (Figure 2.18), an improved prediction of shock location was achieved and the usage of the boundary layer code alleviated the incorrectly predicted expansion after the shock. The accuracy of the computation of the peak suction remained unaffected with $\Delta C_p = +0.2$ and for the second peak of suction on upper side the result improved to an over-prediction by $\Delta C_p = -0.1$, as compared with standard Euler code. Meanwhile, the peak acceleration on the lower side of the wing was captured more accurately with $\Delta C_p = +0.1$, as compared to standard Euler code. Although the main flow features are captured by the Euler approach with the boundary layer correction, the research concluded that “the use of RANS viscous modelling is advised when possible”⁵⁸.

2.3.2 Navier-Stokes CFD

To enhance the drag prediction, the computation of the Navier-Stokes equations is required to model the effects of viscosity and vorticity. The computational time for RANS was estimated to increase about 10 times as compared with Euler solvers^{58,59}. The development of RANS computation was addressed in a project MEGAFLOW^{9,60,61}, of which a substantial part was dedicated to the problem of aero-engine installation.

The computations for the DLR-F6 with through flow nacelles (Section 2.2.1.1) were completed with the use of CFD^{60,61} at $M_\infty = 0.75$ and $Re = 3 \times 10^6$. Due to the time consuming mesh generation and difficulties with creation of structured meshes for some of the configurations, unstructured meshes with prismatic boundary layers were used⁴⁵. Amongst other solvers tested, the computation was performed with TAU⁶¹ which applied the second order central scheme for the discretisation of convective fluxes⁴⁵. Overall, the DLR-F6 configurations with CFM-56 long-cowl (Figure 2.19a), CFM-56 short-cowl (Figure 2.19b) and a very-high by-pass ratio (VHBR) nacelle (Figure 2.19c) were considered. The numerical results were compared with the measurements (Section 2.2.1.1). As for DLR-F6 with CFM-56 long-cowl (Figure 2.19a), an offset of approximately $-20dc$ between the computations and the experimental data was observed (Figure 2.19d) and

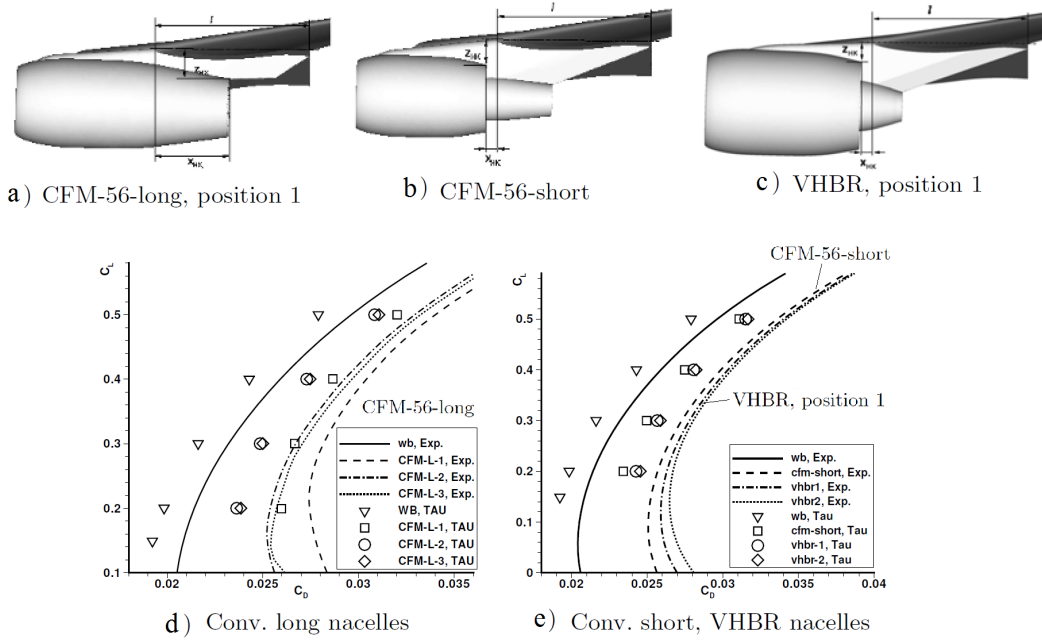


FIGURE 2.19: The geometries of a) CFM-56 long-cowl b) CFM-56 short-cowl c) VHBR nacelles; The lift-drag polars for the numerical and experimental results for the DLR-F6 configuration for d) CFM long-cowl nacelles and e) short-cowl nacelle compared with very-high by-pass ratio (VHBR) nacelles⁹;

approximately $-16dc$ offset was noted for the DLR-F6 clean-wing configuration⁴⁵. The accuracy in determination of drag absolute values corresponds to 6-8% of the total drag⁴⁵ of the clean wing airframe. Even though there was an offset in the prediction of absolute values of the aircraft drag, the installation drag was broadly captured and ranged from 20 to 35dc for the CFM56 long-cowl⁴⁵. As compared with the measurements, the installation drag was under-estimated⁴⁵ from $-3dc$ to $-5dc$ (Figure 2.20a) at a typical cruise condition with $C_L = 0.5$ and $M_\infty = 0.75$ depending on nacelle position. The accuracy in determination of the installation drag was approximately 10% or as reported by Brodersen et al.⁴⁵ “an accuracy of less than 1% regarding the installation drag in comparison with the experimental data can be achieved” when referred to the aircraft total drag. The through-flow nacelle configuration of the CFM-56 with a short-cowl was tested (Figure 2.19b). As compared with the measurements (Figure 2.19e), the under-prediction of -23dc was computed for the total drag of the CFM-56 with a short-cowl. The evaluated discrepancy in the installation drag of -10dc was reported⁹ at a condition with $C_L = 0.5$ and $M_\infty = 0.75$, based on the lift-drag polars (Figure 2.19e). This accuracy corresponded to 25% when compared with the magnitude of the installation drag $\Delta C_{D\ inst} = 40$ for the CFM-56 short-cowl. Moreover, a through-flow nacelle configuration of the very-high by-pass ratio (VHBR) short-with a cowl was tested (Figure 2.19c). As compared with the measurements (Figure 2.19e), the under-prediction of the total drag for the VHBR-configuration by -25dc was computed. As a result, a discrepancy in the installation drag for the VHBR was -9dc at a typical cruise condition with $C_L = 0.5$

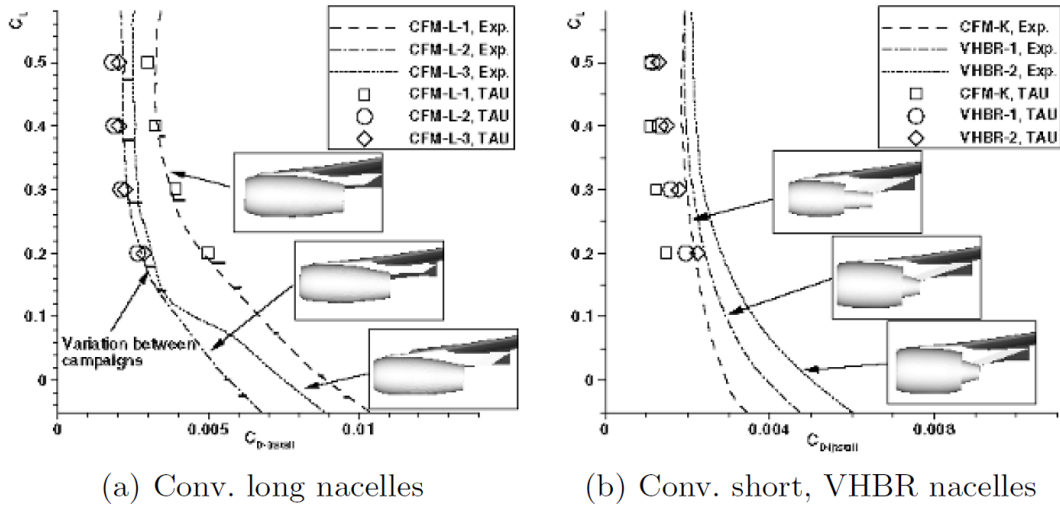


FIGURE 2.20: The installation drag ($\Delta C_{D\ inst}$) for numerical and experimental results for the DLR-F6 configuration for a) range of long-cowl nacelles b) short-cowl nacelle compared with very-high by-pass ratio (VHBR) nacelle⁹

and $M_\infty = 0.75$ (Figure 2.20b). Further study by Devine et al.⁶² reported that the accuracy of the drag prediction within the 2nd Drag Prediction Workshop with use of the DLR-F6 geometry was not satisfactory and the discrepancy reached 40dc for WB and 36dc for WBNP for some participants. Even though the overall accuracy was questioned, it was considered as a reasonable vehicle for use in drag reduction studies. With the improvement in the computational power, the use of finer meshes and application of more complex turbulence models allowed for the prediction of the $\Delta C_{D\ inst}$ to reach 39dc numerically⁶² as compared to 43dc experimentally⁴⁵. Moreover, it was proposed that the potential improvements in the installation drag ($\Delta C_{D\ inst}$) due to a pylon redesign are possible⁶². The use of a compression pylon reduced the installation drag by 6dc as compared with baseline symmetric pylon and the reduction is notable in the context of $\Delta C_{D\ inst} \approx 40$. The reduction was a result of the downstream movement of the shock on the inboard side and it provided a better pressure recovery at the aft of the pressure side of the wing. The implication of the study is that a detailed bespoke design of the pylon can significantly improve the aerodynamic performance and the reduce the installation drag.

Apart from the overall installation effects, the detailed effect of installation on the local aerodynamics needs to be considered to determine if the physics of interaction between the wing and the nacelle are captured correctly. The computations indicate that the nacelle installation caused the upstream movement of the shock on the upper side of the wing⁴⁵. In addition, the nacelle installation resulted in flow acceleration at the leading edge of the lower side of the wing inboard of the pylon, due to the virtual flow channel created between the nacelle pylon and fuselage. The peak suctions on the upper and the lower side of the wing were over-predicted by $\Delta C_p = -0.15$ on the inboard side of the pylon. The shock position was computed $\Delta x/c = -0.02$ upstream as compared with

the measurements. Furthermore, the impact of engine position on pressure distribution was analysed⁴⁵ and it was determined that the local aerodynamics are predominantly sensitive to the horizontal engine displacement and modestly sensitive to the vertical engine displacement.

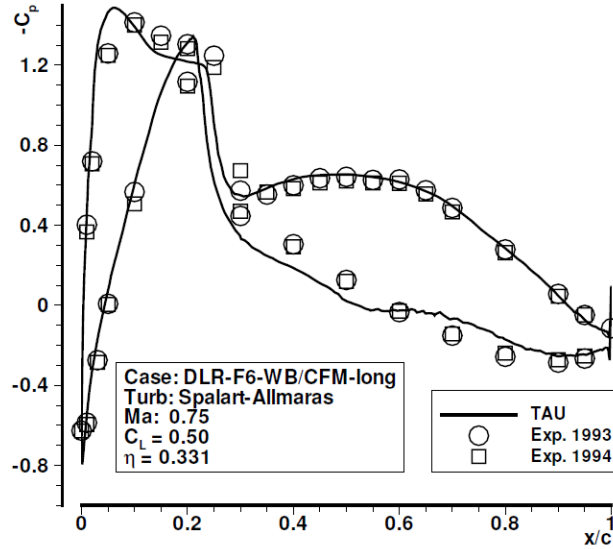


FIGURE 2.21: The pressure distribution (C_p) for numerical and experimental results for the DLR-F6 configuration with the CFM-56 long-cowl nacelle⁹ at inboard cross-section; $M_\infty = 0.75$; $C_L = 0.5$

The ALVAST configuration with turbofan TPS (Section 2.2.2.1) was tested numerically at a cruise condition at $M_\infty = 0.75$ and $C_L = 0.5$ ^{60,61}. Most of the publications concentrate on the detailed aerodynamic effects of the engine installation in terms of changes to the pressure distribution^{9,58–61}. However, the numerical accuracy in determination of the aircraft total drag ($C_{D\ tot}$) and the quantification of the installation drag ($\Delta C_{D\ inst}$) can be determined based on results presented by Rossow et al¹¹. The agreement for the computation of $C_{D\ tot}$ at cruise condition $M_\infty = 0.75$ and $C_L = 0.5$ is approximately $-16dc$ for configurations without turbine power simulators and approximately $-26dc$ for configuration with both TF and VHBR powered configurations (TPS). As compared with the total drag of $300dc$ for the clean-wing configuration, the accuracy is from 5% to 8% of $C_{D\ tot}$. As compared with the experimental $\Delta C_{D\ inst}$ of $34dc$ and $36dc$ for the TF and VHBR respectively, the computations predict $22dc$ and $24dc$ for the TF and VHBR. As a result, the accuracy of approximately $-12dc$ in determination of $\Delta C_{D\ inst}$ is noted, which corresponds to approximately 30% of the installation drag.

The application of numerical methods allowed for a deeper analysis of the flowfield. As compared with the clean wing configuration, a sequence of acceleration and deceleration patterns occurred, which may be attributed to the pylon influence^{58,59} and were suggested as the main cause of the aerodynamic interference. All the key flow features due to engine installation such as an increase in the suction peak on the upper side of

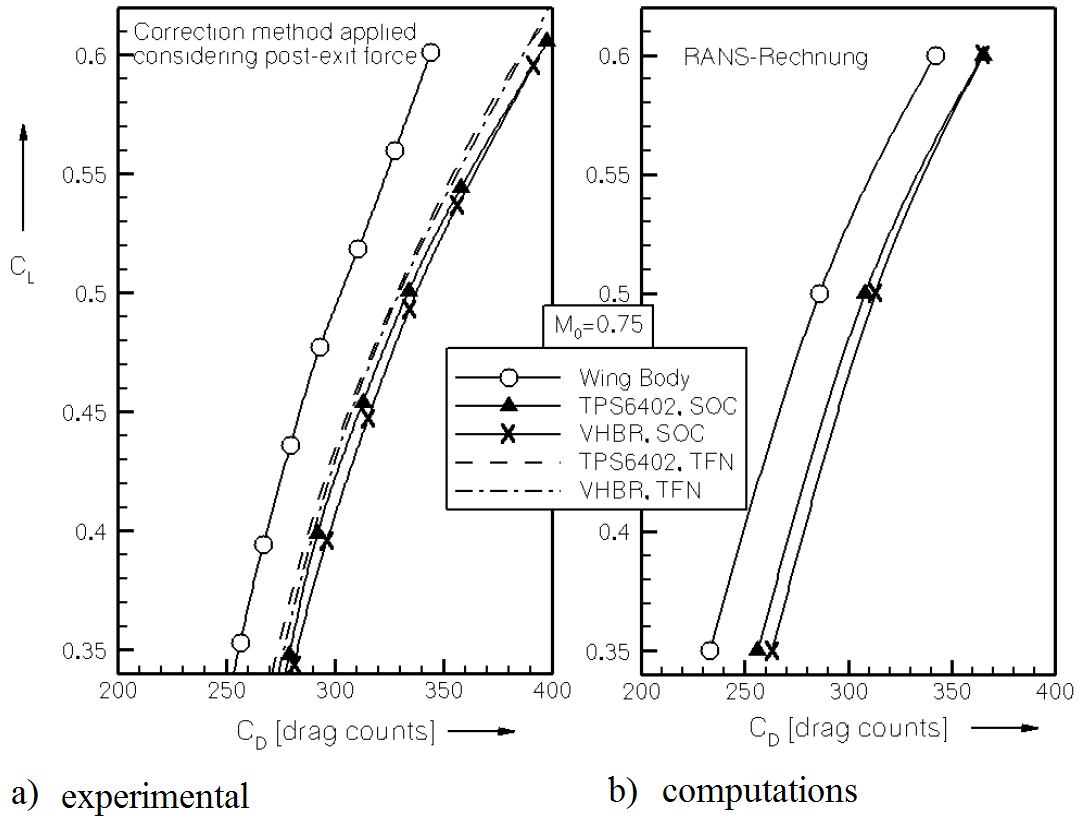


FIGURE 2.22: The lift-drag polar for a) experimental and b) numerical results for the ALVAST configuration with the TF and VHBR turbine powered simulators¹¹; $M_\infty = 0.75$

the wing, the upstream shift of the shock and a reduced value of pressure at the wing leading edge on the lower side of the wing were captured correctly as compared with the measurements. Nevertheless, some of the numerical deficiencies were noted^{58,59}. For example, the ΔC_p at peak suction at the leading edge of the upper side of the wing for the TF powered configuration was under-predicted by +0.1 (Figure 2.23a). Meanwhile, at the wing lower side the peak suction was over-predicted by -0.2 (Figure 2.23a). The peak suction for the VHBR configuration were also compared with the experimental results (Figure 2.23b) and a discrepancy of $\Delta C_p = +0.1$ and $\Delta C_p = -0.1$ was noted for the upper and lower side of the wing respectively^{58,59}. The apparent, over-prediction of the suction on the lower side of the wing in the computations requires a greater compensation in incidence for the incurred loss of lift to maintain the same lift coefficient as in the measurements^{58,59}. Moreover, the UHBR configuration was analysed and the pressure distribution on the wing lower side inboard from the pylon indicated an over-prediction of the minimum pressure coefficient of $\Delta C_p = -0.4$ as compared with experiment⁹. Furthermore, some deviations close to the wing trailing edge are observed and are compared to the deviations noted in analysis of the DLR-F6 configuration. It was assumed that flow separations are responsible for this effect⁹. To test the sensitivity of the pressure distributions to the engine power setting, the through-flow condition

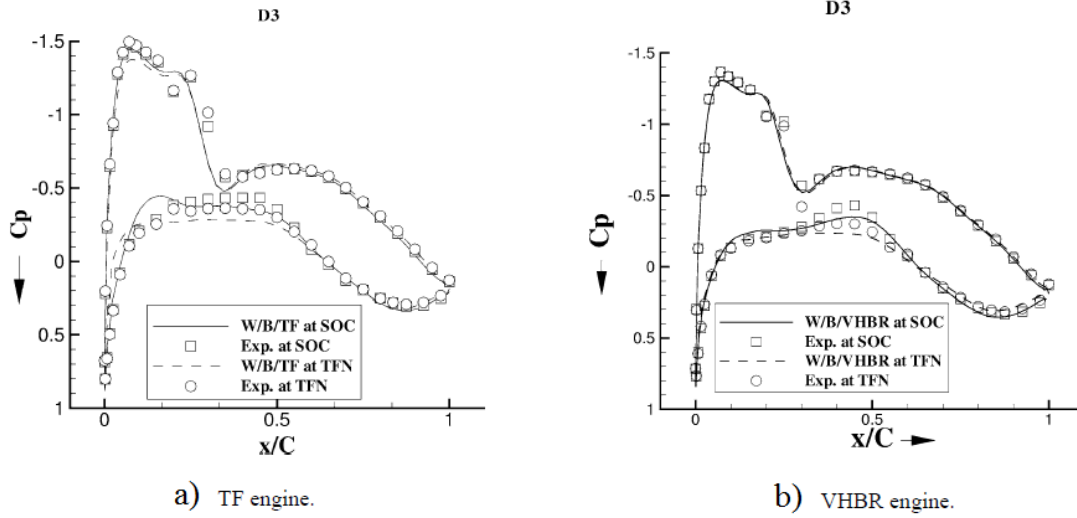


FIGURE 2.23: The pressure distribution (C_p) for experimental and numerical results⁵⁹ of ALVAST airframe with a) TF engine and b) VHBR engine on the inboard side of the installation; $M_\infty = 0.75$; $C_L = 0.5$

(“TFN”) with the fan pressure ratio close to unity was considered and compared to the “Start of Cruise” power setting (“SoC”). Similarly as in the experiment, the same shock location for the TF at “TFN” and “SoC” condition was predicted at $x/c=0.35$ ^{60,61} and it is within $\Delta x/c = -0.05$ (Figure 2.23a). Also, the peak acceleration at the leading edge of the upper side of the wing was computed with the same accuracy of $\Delta C_p = +0.1$ for both “TFN” and “SoC” condition. However, the computations indicated an increased acceleration $\Delta C_p = -0.2$ at $x/c=0.1$ on the lower side of the wing for “SoC” condition of the TPS^{60,61} as compared with experiment. Meanwhile, for the through-flow condition no over-prediction of the acceleration at $x/c=0.1$ on the lower side of the wing was observed as compared with the experiment (Figure 2.23a). Furthermore, both “SoC” and “TFN” conditions present an increased value of pressure $\Delta C_p = +0.1$ at $x/c=0.4$ on lower side of the wing^{60,61}.

In total four engine positions were considered (Figure 2.24a) for the numerical study on the VHBR engine^{58,59}, however, only pressure distributions on the wing were reported. Overall, a small dependency of interference effects on the vertical position was observed (Figure 2.24b), whereas the pressure distribution was significantly dependent on the horizontal position of the engine (Figure 2.24c). The overlap position of the engine (Figure 2.24c) resulted in the local acceleration of the flow inboard of the pylon at the leading edge. This was due to reduced cross-sectional area of the channel between the engine and the fuselage. The upstream movement of the engine was suggested as a measure to alleviate detrimental interference effects on the wing. Furthermore, it was stressed that no optimisation of the geometry for any of the positions was carried out and that the relation of aerodynamic effects and engine position depends on various factors, as e.g. engine mixer type (internal, external), nacelle and core-cowl shape and extension, pylon design and size^{58,59}.

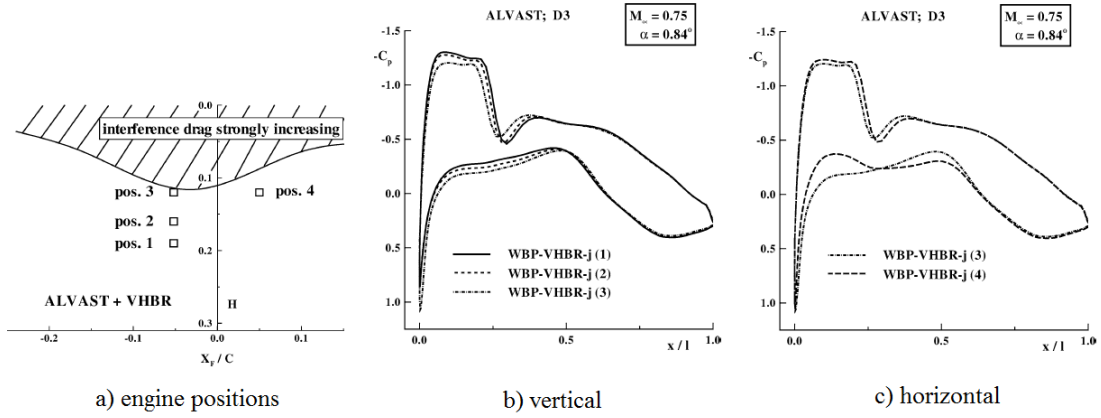


FIGURE 2.24: a) Considered engine positions for the VHBR engine on the ALVAST airframe⁵⁹ and the pressure distribution (C_p) for b) range of vertical engine positions and c) range of horizontal engine positions; $M_\infty = 0.75$; $C_L = 0.5$

A more modern study with a “twin-jet civil aircraft configuration”⁶³ with turbine powered simulator was completed by Zhang et al.⁶³ at $M_\infty = 0.785$; $Re = 24 \times 10^6$; $C_L = 0.55$. However, the sparse amount of information provided limits the use for the current project. The study considered a power configuration and compared it with the through-flow nacelle configuration. The presence of the jet was found to increase the drag by 1.6dc. It was explained⁶³ to be a result of the accelerated flow inboard of the pylon on lower wing surface due to the presence of the jet. Apart from the effect on drag, the loss of lift was reported and the change of the wing-loading to an aft-loaded wing. The reported accuracy for the computation as compared with experiment is 6dc for absolute values of total aircraft drag, as based on the through flow nacelle measurements.

2.3.3 Guidelines for numerical drag prediction

To address the need for guidelines and best practice rules in the prediction of drag with the use of CFD, a range of Drag Prediction Workshops (DPW) were carried out^{12,64}. A range of aircraft geometries were investigated to set generic guidelines and improve the overall accuracy of drag determination. The DLR-F4 configuration was tested in the first DPW and the DLR-F6 configuration (Section 2.2.1.1) was tested in the second and the third Drag Prediction Workshops. The second drag prediction workshop¹² included the study on the nacelle installation effects. Moreover, the fourth and the fifth DPW took place with use of NASA Common Research Model⁴⁶ (Section 2.2.1.2). Based on the experience from the DPW series a range of conclusions and a set of numerical guidelines were derived^{13,65–67}.

2.3.3.1 Meshing guidelines

The prediction of drag for configurations of civil aircraft was proven to be significantly dependent on the grids⁶⁵. The dominant sources of error are related to the spatial discretisation. However, the mesh type, topology, grid resolution and grid-quality need to be taken into account⁶⁵. Based on the mesh size only, the individual participants of the Drag Prediction Workshops⁶⁵ have demonstrated a monotonic grid convergence behaviour, particularly for attached-flow cases. However, in many instances, “the demonstration of apparent grid convergence is contradicted by results obtained with the same code on grids of different construction”⁶⁵. The structural hexahedral meshes proved to be a reliable way to reduce the number of points without losing the grid quality due to sharp angle elements⁶⁶. Moreover, the convergence study for structured and unstructured meshes were compared and the conclusion was “higher errors associated with the unstructured-mesh results at a given level of resolution”⁶⁵. The deficiency of unstructured meshes was also linked to an “inadequate resolution of the fuselage”⁶⁵. In the MEGAFLOW project⁶¹, hexa meshing was initially used. Due to the time consuming process of structured mesh generation that can take up to 4 months for a new geometry configuration⁴⁵, it was decided to substitute structured meshes by hybrid meshes⁶¹.

Summarising the second DPW, it was considered not fully satisfactory⁶⁶. Due to flow separations in the modelled flow the influence of grid became significant and it eventually led to a large scatter in the reported data^{65,66}. The grid refinement is expected to help alleviate the aforementioned issues as the grid sizes tend to increase due to the greater computational power available. For example within the period of 10 years mesh sizes increased from 2 million element to 10 million elements⁶⁶. A further increase to meshes of 35-40 million was anticipated and currently can exceed 100 million. Overall, the differences in grid resolution affect primarily the local quantities such as pressure coefficients or skin friction as reported by Brodersen, Eisfeld et al.⁶⁶, as well as Brodersen, Rakowitz et al.¹³ The most noticeable effect of grid resolution is the location of shock areas and the computed shock sharpness⁶⁵, it was noted that even slight changes of shock location and strength based on grid resolution affect the computed drag values appreciably⁶⁵. Furthermore, the changes to drag were explained by the resolution ahead of the shock and the effect of the spatial resolution on the thickening of the boundary layer through the convection of numerically upstream-generated entropy⁶⁵.

Nevertheless, the grid resolution is not the only criterion that leads to accurate drag prediction. It is believed that the “mesh generation for computational aerodynamics will remain somewhat of an art based on practitioner experience and intuition, at least for the near term”⁶⁵. Therefore, additional meshing criteria were suggested to guide meshing practitioners. For example, it was determined that the boundary layer flow can be only resolved with elements of high aspect ratio⁶⁵ such as 1:1000 or 1:10000. Furthermore, the boundary layer should be resolved with at least 10-16 layers in the

physical boundary layer and overall of 25 prismatic layers should be present in the mesh^{9,13}. It was also proposed to extend the number of prismatic layers to 40 layers⁶⁶. The first cell height should be based on $y^+ = 1$ ⁶⁵ or even set⁶⁶ to be $y^+ = 0.9$. The growth ratio of elements normal to the surface should be 1.25 or even 1.2 if possible⁶⁵, whereas the lateral growth ratio should not exceed 1.5. In addition, some detailed cell spacing criteria for the wing were developed⁶⁵, such as local spacings at wing leading and trailing edge not greater than 0.1% of local chord, spanwise spacing at root and tip not greater than 0.1% of semispan, fuselage spacing at the nose not greater than 2% of c_{ref} , and more than 8 nodes for the wing blunt trailing edge⁶⁵. Moreover, the mesh resolution and quality needs to be improved for the “regions of separated flow [which] are much more demanding and less well understood than for cases in which the flow is fully attached”⁶⁵. For those more demanding aerodynamic problems, the grid quality needs to be taken into account based on the element aspect ratio and orthogonality for structured meshes.⁶⁵ Finally, the domain size was also tested to determine the impact of the far-field boundary condition on the computed results. It has been established that a far-field boundary condition needs to be located by 100 c_{ref} away from geometry⁶⁵.

2.3.3.2 Turbulence modelling

The fact that the configurations tested in the 1st and 2nd DPW contained substantial regions of turbulent separated flow made those flow configurations good test cases for turbulence models. The regions of separated flow are known to be very sensitive to turbulence model fidelity⁶⁵. Amongst many models tested, the most common models were the Spallart-Almaras and Mentner’s $k - \omega$ SST. The $k - \omega$ SST model was developed with aeronautical applications in mind and was deliberately sensitized to flow separation⁶⁶. The DLR-F6 configuration with a long-cowl nacelle was used for computations with both turbulence models at $C_L = 0.5$ and $M_\infty = 0.75$. As compared with the S-A model, the sensitivity of $k - \omega$ SST to separation was found to be beneficial¹³ for the cases with the trailing edge separation on the wing. The solution of S-A had a nonphysically fully-attached flow up to the trailing edge. It was stated that “the application of $k - \omega$ SST turbulence models produces a more adequate solution of the flow separation”¹³. However, it has to be kept in mind that the greater sensitivity of $k - \omega$ SST model to the flow separation might result “in earlier lift breakdown and a larger flow separation for the maximum incidence” as appeared on other test cases⁶⁶ such as DPW-W1 and DPW-W2 wing-only configurations tested at angle of attack 3°.

To improve the accuracy of drag prediction, the modelling of flow transition became of interest⁶¹. It has been shown for the DLR-F4 case⁶¹, that the modelling of transition can reduce the discrepancy in the drag prediction from +20dc for fully turbulent models to +6dc when with transition modelling. Thus, the computations for the DLR-F6 with the CFM-56 long cowl were completed at $C_L = 0.5$ and $M_\infty = 0.75$ with and without transition modelling on two solvers¹³. Both solvers under-estimate the total drag

($C_{D\ tot}$) as compared with the fully turbulent model. With the use of the transitional model $-10dc$ accuracy for WB and $-3dc$ for WBNP was achieved¹³ and it corresponds to 3% and 1% accuracy for WB and WBNP respectively as referred to the aircraft total drag. However, due to the suspicion of inadequate transition specification, a larger flow separation was reported. Thus, the “fully-turbulent computation shows a more plausible result”¹³. Overall, even though in some cases the modelling of transition offers an improved accuracy, it is very sensitive to the method of fixing the transition location. As a result fully turbulent models are preferred.

An alternative way to improve the turbulence modelling was found to move toward more complex turbulence models. Eisfeld et al.⁶⁷ proposed the hybrid SSG/LRR-omega model, which is a differential Reynolds stress turbulence model⁶⁷. Instead of the Boussinesq hypothesis of linear relation of Reynolds stress tensor and the strain tensor, a non-linear tensor expansion in terms of the mean strain rate and rotation tensor is used⁶⁷. Overall an improved accuracy of the model was reported as the absolute drag accuracy for the DLR-F6 configuration improved from typically -30dc to values between -9dc and -6dc depending on the lift coefficient⁶⁷. The advanced turbulence modelling resulted in the improved prediction of the separation pattern on the upper wing surface. The shock location was captured within $\Delta x/c = -0.03$ as compared with the experiment⁶⁷.

2.4 Drag extraction theory

The aim of this section is to provide the reader with an introduction to thrust and drag book-keeping and the fundamental aerodynamics. Thrust and drag assessment is a difficult task due to the relatively complicated aircraft geometry, as well as the problematic split between the engine and aircraft perspectives of the aerodynamics. Two approaches can be applied to the aircraft drag assessment and those are the body-perspective and the fluid-perspective approaches⁶⁸. Nevertheless both cause some difficulties. A brief comparison of the body perspective (Section 2.4.1) with the fluid perspective (Section 2.4.2) is considered based on the current state of knowledge. Greater details on the method implemented in current research are provided in Section 4.4. Some additional information about drag methods not directly pertinent to the current research was located in Appendix A.

2.4.1 Near-field drag extraction theory

The full body-perspective requires the integration of pressure and viscous forces along all the geometry of the aircraft and engine including the internal engine flow. It is practically impossible in terms of complex jet-engine internal geometry to perform such an integration. Therefore, other intermediate measures can be taken in a form of defining the streamtube of a jet engine and treating it as a part of the geometry. The pressure force that acts on the boundary of the stream tube is then integrated. The drag of the engine-airframe configuration is a summation of all the external surfaces of the nacelle and airframe and on the surfaces of the pre-entry and the post-exit streamtubes⁶⁹ (Figure 2.25). The front streamtube is defined from the upstream infinity plane to the highlight point on the cowl. The exit streamtube is defined from the nacelle trailing edge down to the downstream far-field at infinity⁷⁰. This approach excludes complicated internal turbomachinery from the integration process and minimises the problems with wake integration.

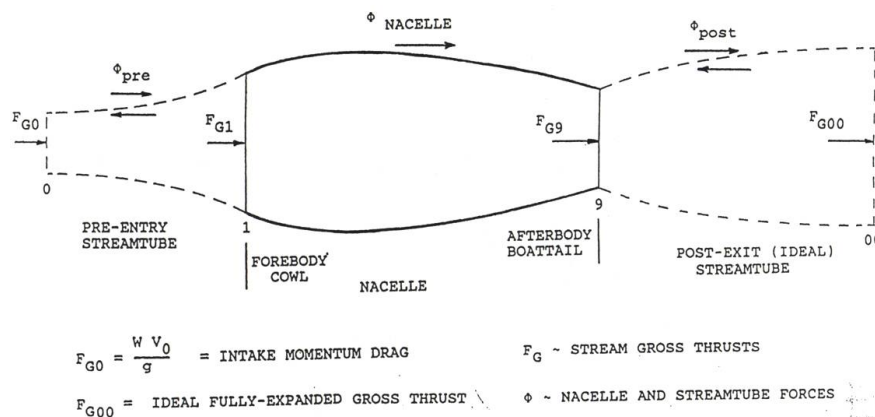


FIGURE 2.25: Jet-engine streamtubes and streamtube forces⁶⁹

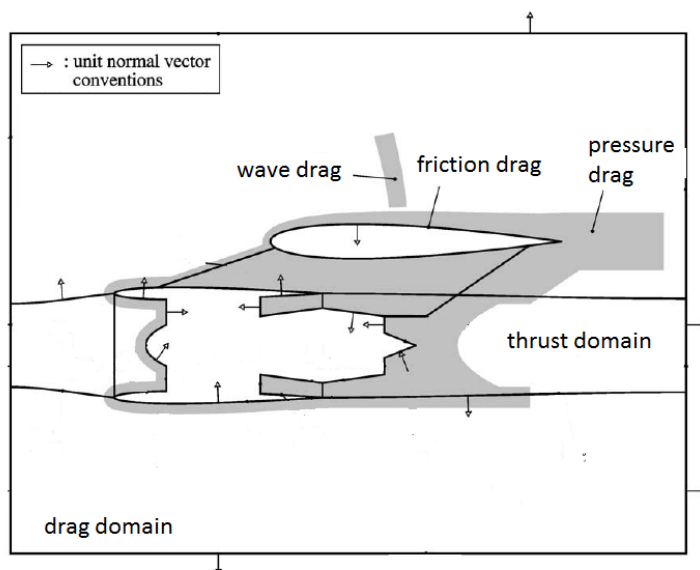


FIGURE 2.26: Jet-engine aircraft domain decomposition based on Destarac et al.⁷¹

However, the presence of the wing and pylon complicates the task of drag and thrust determination even more. Further analytical analysis for correct thrust and drag assessment might be carried out as presented by Destarac et al.⁷¹. The computational domain needs to be split into a thrust and a drag domain (Figure 2.26). The pylon structure lies in both thrust and drag domain and is typically split into two parts. The part of the pylon washed by the jet lies in the thrust domain, whereas the remaining part of the pylon is in the drag domain. In all cases an extreme care has to be taken when assessing drag components in order to avoid the double accounting of forces.

2.4.2 Far-field drag extraction theory

The far-field method for drag extraction determines the overall drag from the perspective of the fluid. The far-field method was successfully used for the calculation of the airframe drag of un-powered configuration and the method offers the breakdown of the overall drag into friction drag, vortex drag and wave drag⁷⁰. The decomposition of the far-field drag indicates the physical origins of the drag and in this sense is useful for overall understanding. The far-field method requires the definition of an extensive outer domain for wake integration. In the context of engine installation studies, Brodersen et al.¹³ carried out a notable study to determine the accuracy of far-field method applied to solutions computed on two different solvers (TAU and elsA)¹³. As compared with the near-field force integration, the values of absolute aircraft drag were 7dc more accurate for the far-field method. Nevertheless, the evaluation of the overall installation drag resulted in broadly the same answer for the two methods¹³. As compared with the installation penalty of 43dc in experiment the far-field in 41.2dc and 46.5dc for TAU

and elsA respectively¹³. Meanwhile, the near-field method resulted in 41.7 and 48dc for TAU and elsA respectively¹³

The application of a far-field method to the powered configuration requires additional care in determination of the pre-entry thrust⁷². To separate the thrust from drag an additional control volume within the domain was created. The far-field method is used within the created control volume and the momentum balance for the pre-entry stream-tube is evaluated to determine the pre-entry thrust. The method was verified by comparison with the near-field approach and the agreement between the methods is within 1.7dc which is 2.3% of the nacelle drag.

2.5 The summary of literature review

In summary, the magnitude of installation effect on commercial type aircraft ranges from 20 to 40 dc, which corresponds to approximately 8-15% of the total aircraft drag. Over the range of studies, the engine size and the change of engine position axially or vertically were found to be the most influential factors. The installation effect can be predicted numerically, however, the best accuracy reached approximately 3-5dc. Moreover, the suitable validation test cases were identified for the airframe configuration (CRM) and the isolated engine with separate exhaust (DSFRN). To aid the practical development of the numerical methods, the detailed meshing criteria were outlined. The findings from the literature review have a fundamental effect on the choice of the project scope.

CHAPTER 3

Scope of the project

“To raise new questions, new possibilities,
To regard old problems from a new angle,
Requires creative imagination
And marks real advance in science.”

A. Einstein

In this chapter, the gap in knowledge (Section 3.1) is identified as an outcome of the literature review. Furthermore, an overall strategy for the project is outlined in a form of a project roadmap (Section 3.2). Based on the broader project perspective, the detailed scope of the current project is highlighted in terms of the test matrix of undertaken configurations (Section 3.3) which are investigated within the current research.

3.1 The gap in knowledge

Based on the literature review, no publicly available study was found to systematically address the engine installation effects for realistic turbofan engines on commercial aircraft. The past studies typically addressed the evaluation of drag for modern turbofan engines, however the quantification of the effect of engine installation on engine thrust was depreciated. Therefore, there is a requirement for a detailed breakdown of the overall effect of engine installation on the lift, drag and thrust force and creation of a complete system for evaluation to account for all the effects with regards to key parameters such as an engine position, engine size and engine power setting. An enhancement of thrust and drag book-keeping method and the quantification of the interference on component level is necessary to propose a simplified model for the prediction of overall effect of installation for large turbofans at a preliminary design stage.

3.2 The project roadmap

To quantify the overall effect of engine installation of lift, drag and thrust of an aircraft and to develop an integrated drag prediction tool for under-wing installed engines, a detailed strategy had to be developed. The work had to cover a broad range of activities for both isolated and wing-installed engines. Overall, the activities involved in the project have been grouped into four main branches (Figure 3.1) as the fundamental building blocks of the research. The first branch was the isolated engine studies, which included conceptual design of engine cycle and engine performance analysis, as well as CFD computations and validation of those CFD methods. The impact of the pre-entry and post-exit streamtube on the nacelle aerodynamics was assessed for different engine power settings. The second branch of the project was focused on studies for an isolated airframe that involved the validation of numerical methods, as well as an incidence study, a Reynolds number study and a flight Mach number study. The third, and key, branch of the project concerns studies on engine installation. There are two main aspects of work on engine installation that can be investigated such as the impact of the engine on the wing aerodynamics and the assessment of engine performance during a cruise phase of flight. The outcome of the cruise analysis for chosen configurations was the quantification of changes to the integrated fuel burn. The fourth branch is called Drag Methods. Within this last branch the thrust and drag book-keeping is clarified and within this framework a suitable drag extraction method was determined and applied for both the engine and the aircraft studies.

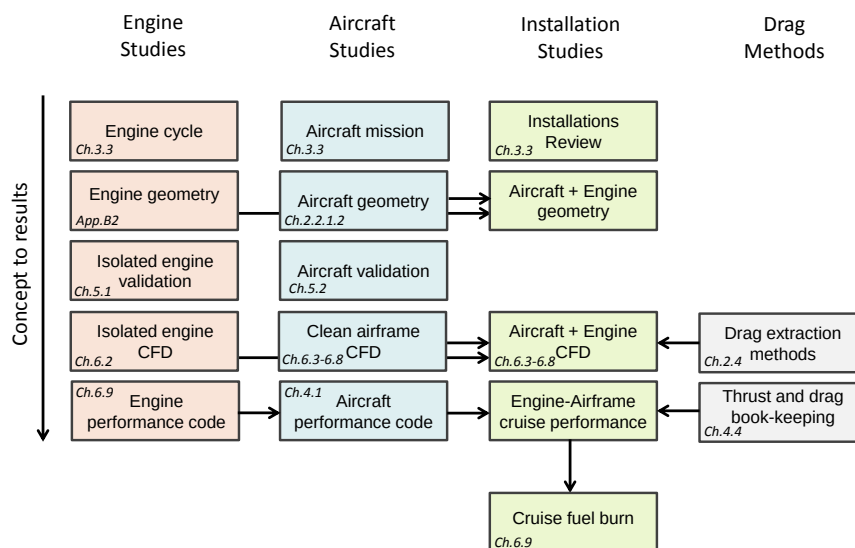


FIGURE 3.1: Project Roadmap; Main four branches of project are grouped vertically: Engine Studies, Aircraft Studies, Installation Studies and Drag Methods; The thesis chapters are annotated on the work blocks.

3.3 The test matrix

A typical mission at an altitude of $H = 10668m$ (35000 ft) and a cruise Mach number of 0.82 was chosen based on the overall project scope to assess the impact of engine installation for the cruise conditions of a typical long-haul flight. The range of aircraft angles of attack from 0° and 5° was considered. For this given mission specification, a typical twin-engine wide-body 250-300 seater aircraft was considered and the NASA Common Research Model (CRM) was chosen for the study due to the broad similarity to the A330 (Section 2.2.1.2).

The literature review indicated that the greatest sensitivity of engine installation effects was to the vertical and axial engine position under the wing. Therefore, the axial (Δx) and vertical (Δz) displacements of the engine are considered (Table 3.1, Figure 3.2). The nomenclature for the positions was chosen to be similar to the chessboard with letters A, B, C, D denoting the columns and rows denoted by numbers from one to three. The displacement of the trailing edge point of the topline on engine nacelle was measured with regards to the intersection point of the wing leading edge with the engine installation plane. The installation plane was chosen to be normal to the horizontal axis of the airframe coordinate system (y) and it was defined based on the original CRM geometry at the horizontal location of the trailing edge point of the topline on the through-flow nacelle ($y_{inst} = 9.806\text{ m}$ from the aircraft axis). A range of engine positions were considered based on the location of the fan cowl trailing edge in relation to wing leading edge (Figure 3.3). The relative axial and vertical position of the engines was referred to the local wing chord (c_{loc}), which was defined as the wing chord of the wing on the engine installation plane. The axes of the coordinate system for the engine displacement are parallel to the corresponding longitudinal (x) and vertical (z) axes of the airframe. To provide better context for the work and to establish the region of interest for under-wing engine installation, a revision of engine axial and vertical position on existing aircraft was performed (Figure 3.3) which expanded the work of Oliveira et al.⁷³. The original reference⁷³ did not quote the accuracy on the evaluation of the engine positions. Based on the publicly available technical drawings of popular long-range aircraft, the position of the nacelle trailing edge with regards to the wing leading edge was quantified. However, the accuracy of drawings was also not specified, thus the data needs to be treated with caution. For additional guidance, the historical boundary of increased aerodynamic interference⁷³ (dashed line) was included. The region above the line located around $\Delta x = -0.05$ and $\Delta z = -0.05$ was traditionally considered as a region of high penalties attributed to intensive aerodynamic interference caused by the close-coupling of the engine with the airframe. The presence of existing engine installations within these boundaries clearly indicates that the application of more refined design methods can lead to a plausible design of engine installation within those boundaries.

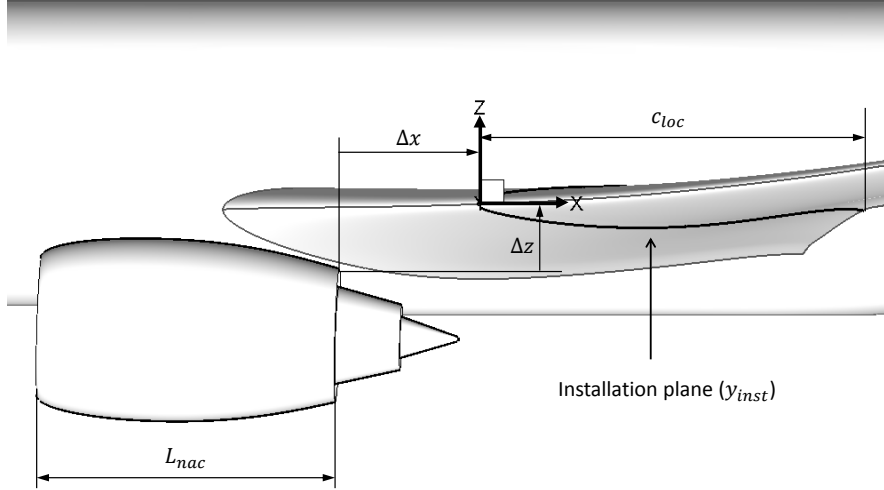


FIGURE 3.2: Sketch of key installation parameters

Within the current study, it was decided to consider two engine sizes for the specified aircraft mission. Given the recommendations from the industrial partner, the study should encompass a typical turbofan and the second engine with an increase in fan diameter by 23%. The increase in diameter by approximately 20% is expected to exhibit a notable increase in the engine drag based on the literature⁵⁴. To determine the realistic engine boundary condition for the numerical study, an initial specification of the engines was determined. Based on the performance of the CRM with through-flow nacelles at $M_\infty = 0.82$, a net thrust requirement at an altitude of 10668m (35000 ft) was estimated. For a given mission profile, two engine thermodynamic cycles need to be considered to determine the full range of boundary conditions over a cruise phase of flight. It was chosen to consider an engine representative of a typical modern turbofan-engine ($BPR = 10$ and $OPR = 50$) as a baseline engine. The larger engine should represent a relatively high by-pass ratio turbofan-engine ($BPR \approx 17$ and $OPR \approx 60$). In addition to the design point considerations, the study encompasses the performance over the entire cruise phase from the start of cruise to the start of descent for the chosen geometrical configurations (Table 3.1). The thrust requirement over such a range of operating points ranges from 100% at the start of cruise down to approximately 40% at the start of descent condition. The engine performance results and the details of the engine design are discussed in the Section 6.2. All the considered geometrical configuration at all engine operating conditions were considered over a range of aircraft incidence ($AoA_{A/C}$) from 0° to 5° . The simulation of the relatively wide range of aircraft incidence was intended to enable altering the aircraft angle of attack during the entire cruise phase of flight.

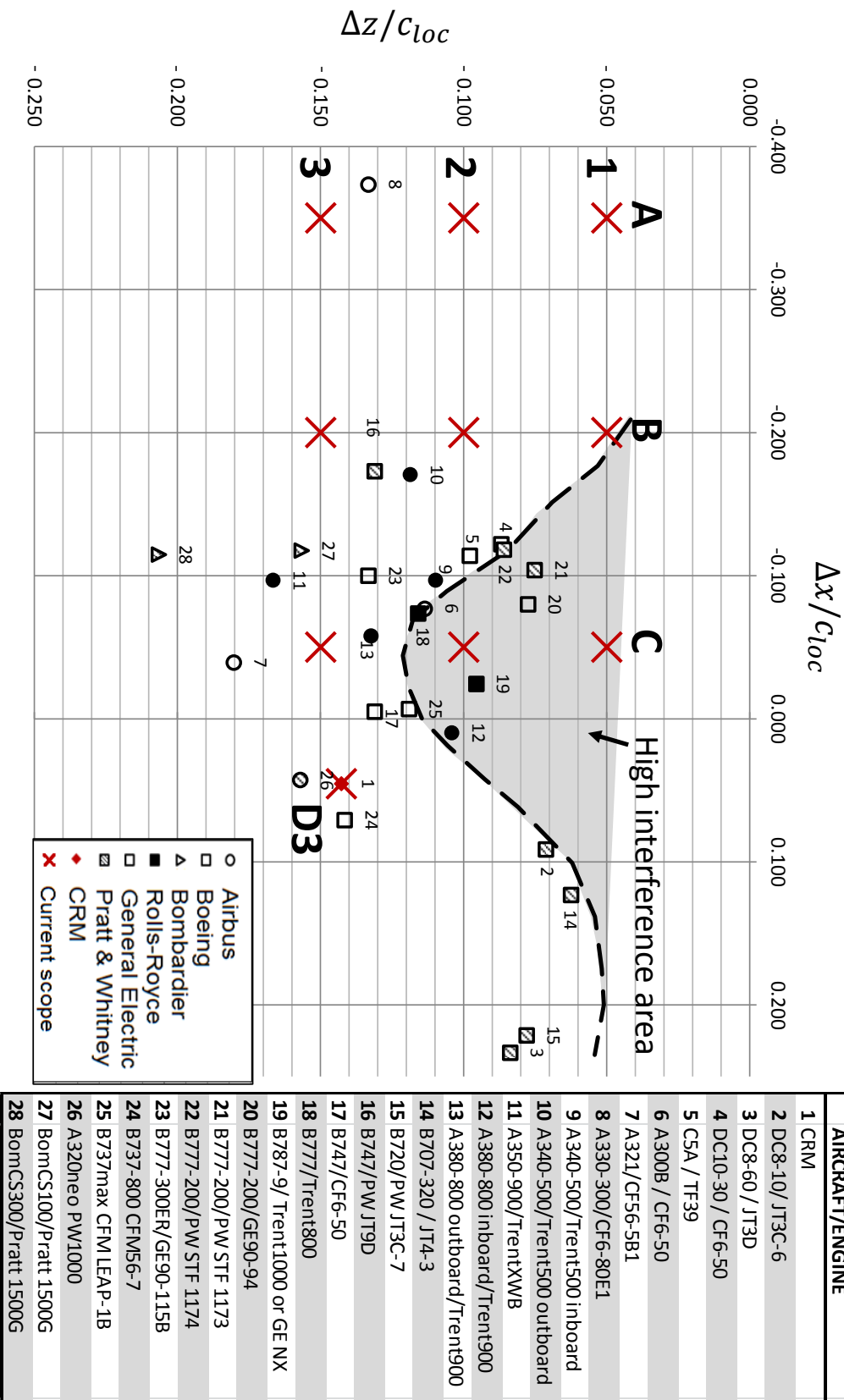


FIGURE 3.3: Map of investigated engine positions for the two engines E1 and E2 in the context of existing engine installations based on Oliveira et al.⁷³

Configuration	dx/c_{loc}	dz/c_{loc}	$F_{N \text{ cruise}}$	$AoA_{A/C}$
CRM (D3)	0.045	0.143	100%	$0^\circ - 5^\circ$
A1	-0.35	-0.05	100%	$0^\circ - 5^\circ$
B1	-0.2	-0.05	100%	$0^\circ - 5^\circ$
C1	-0.05	-0.05	100%	$0^\circ - 5^\circ$
A2	-0.35	-0.1	100%	$0^\circ - 5^\circ$
B2	-0.2	-0.1	100%	$0^\circ - 5^\circ$
C2	-0.05	-0.1	100%	$0^\circ - 5^\circ$
A3	-0.35	-0.15	100%	$0^\circ - 5^\circ$
B3	-0.2	-0.15	100%	$0^\circ - 5^\circ$
C3	-0.05	-0.15	40%-100%	$0^\circ - 5^\circ$

TABLE 3.1: Range of investigated engine positions for engine E1 and E2 over the selected range of engine power settings ($F_{N \text{ cruise}}$) and aircraft incidence ($AoA_{A/C}$)

CHAPTER 4

Methodology

“By three methods we may learn wisdom:
First, by reflection, which is the noblest;
Second, by imitation, which is the easiest;
And third by experience, which is the bitterest.”
Confucius

In this chapter, the reader is provided with the details of the methodology used in achievement of the research aim. As the first section, the description of aircraft performance modelling is outlined (Section 4.1). The section on engine design (Section 4.2) highlights the methods used in engine performance calculation and generation of the engine geometries. The methodology chapter is concluded by an extensive description of the methods used in Computational Fluid Dynamics (Section 4.3) and the description of current thrust and drag system for a re-trimmed aircraft (Section 4.4).

4.1 Aircraft performance

A simplified approach of fuel burn analysis at the cruise phase is considered. A point mass model of the aircraft is considered at the constant altitude and the constant flight Mach number. The balance of forces on the aircraft needs to be obtained to result in steady state flight as a basis of the calculations. The initial mass of the aircraft at the start of the cruise phase is determined based on the BADA methodology⁷⁴ (Section 4.1.1) for a typical 250-seater aircraft. The climb phase was modelled⁷⁴ as independent of the type of engine used for the cruise calculation. As a result, the fuel burnt during the climb phase was deducted from the maximum take-off mass of the airplane and the aircraft mass at start of cruise for baseline configuration was established. To account for the weight penalty associated with the larger engine, the weight estimation of the propulsion system had to be evaluated (Section 4.1.2). The weight of considered engines

was evaluated and an evaluated weight penalty for the larger engine was added to the total aircraft mass at the start of cruise.

4.1.1 Base of Aircraft Data (BADA)

Base of Aircraft Data (BADA) is an extensive database, which specifies the aircraft performance parameters and airline procedure parameters for approximately 438 aircraft types^{74,75}. The model was developed by Eurocontrol and aimed for the use in trajectory simulation and prediction algorithms within the field of Air Traffic Management. The performance coefficients included, allow for the calculation of thrust, drag and fuel flow and specification of nominal cruise, climb and descent speeds. The detailed definitions of each of the coefficients and then full explanation of the method is contained in the User Manual⁷⁵.

Based on the parametrised engine thrust and airframe drag models, the BADA method uses the Total Energy Model (TEM)⁷⁵ to resolve the forces that act on the aircraft and to calculate the horizontal and vertical accelerations. TEM “equates the rate of work done by forces acting on the aircraft to the rate of increase in potential and kinetic energy” and expressed by Equation 4.1. The formulation expresses the rate of work by engine thrust in the direction of aircraft velocity (F_N), the aerodynamic drag (D) and the true airspeed (V_{TAS}). The increase of potential energy is defined with the use of aircraft mass (m), gravitational acceleration (g) and the vertical speed ($\frac{dH}{dt}$), while the increase of the kinetic energy is defined with the use of aircraft mass (m), true airspeed (V_{TAS}) and the rate of change of airspeed ($\frac{dV_{TAS}}{dt}$). Based on the physical corrections for the influence of the flight Mach number and on the assumptions of the aircraft operation⁷⁵ the rate of climb or descent (ROCD) can be evaluated (Equation 4.2), where T is an ambient temperature and $f[M]$ is a function of Mach number dependent on the flight phase and aircraft operating conditions.

$$(F_N - D) \cdot V_{TAS} = mg \frac{dH}{dt} + mV_{TAS} \frac{dV_{TAS}}{dt} \quad (4.1)$$

$$ROCD = \frac{T - \Delta T}{T} \left[\frac{(F_N - D) \cdot V_{TAS}}{mg} \right] \times f[M] \quad (4.2)$$

The lift coefficient of the aircraft (C_L) was derived from the simple body point mass model with the correction for the bank angle between the airframe axis and the horizontal reference of the ground (α_{bank}). The drag coefficient was derived based on drag-lift polars of an aircraft, where $C_{D0,CR}$ and $C_{D2,CR}$ are the coefficients derived for a particular aircraft based on the datasets within the BADA. Due to the similarity of the CRM to the Airbus A330, the A330 model was used within the study.

$$C_L = \frac{2mg}{(V_{TAS})^2 A_{ref} \cos(\alpha_{bank})} \quad (4.3)$$

$$C_D = C_{D0,CR} + C_{D2,CR} \times (C_L)^2 \quad (4.4)$$

Finally, the mass of the aircraft will be reduced sequentially by the amount of fuel burnt by the aircraft. The fuel consumption is determined (Equation 4.5) based on the engine thrust (F_N) and thrust specific fuel consumption (SFC), where SFC is a function (Equation 4.6) of true airspeed (V_{TAS}) and the aircraft dependent coefficients C_{f1} and C_{f2} . Overall, the aircraft mass at any point of the flight can be determined.

$$\dot{m}_f = SFC \times F_N \quad (4.5)$$

$$SFC = C_{f1} \times \left(1 + \frac{V_{TAS}}{C_{f2}}\right) \quad (4.6)$$

4.1.2 Estimation of weight for the propulsion system

The mass of the propulsion systems were estimated based on the statistical correlations for engine mass⁷⁶ and nacelle mass⁷⁷. The bare engine mass is estimated based on the WATE++ method⁷⁶. This method is a statistical approach to engine weight estimation (Equation 4.7) based on the overall pressure ratio (OPR), core massflow ($\dot{m}_{core}$) and a set of coefficients (Equations 4.8-4.10) based on the engine by-pass ratio (BPR). This method has been calibrated based on a range of engines with a reported uncertainty⁷⁶ of $\pm 10\%$ on the mass estimation for engines ranging in thrust from 120 to 400 kN.

$$m_{eng} = a \left(\frac{\dot{m}_{core}}{100 \text{ lbs/s}} \right)^b \left(\frac{OPR}{40} \right)^c \cdot \frac{0.454 \text{ kg}}{\text{lbs}} \quad (4.7)$$

$$a = \left(-6.590 \times 10^{-1} \right) BPR^2 + \left(2.928 \times 10^2 \right) BPR + 1915 \quad (4.8)$$

$$b = \left(6.784 \times 10^{-5} \right) BPR^2 - \left(6.488 \times 10^{-3} \right) BPR + 1.061 \quad (4.9)$$

$$c = \left(-1.969 \times 10^{-3} \right) BPR + 0.0711 \quad (4.10)$$

The WATE++ tool was tested for the evaluation of engine mass for the two generic engines used in the project (Section 6.2). The weight estimation for the baseline engine (E1) engine was assessed against known engines of a similar class (BPR, OPR, thrust) and a discrepancy of -3% on engine mass was noted, which was considered to be satisfactory within the performance of statistical methods⁷⁸. The use of WATE++ evaluated an increase of engine mass of +64% for the prospective large turbofan (E2).

	$BPR [-]$	$OPR [-]$	$m_{core} [kg/s]$	$m_{eng} [kg]$
RR Trent 1000	10	52	X	5,765
GE NX	9.6	41	X	5,816
E1	10.7	46.1	90.0	5,595
E2	17.8	55.8	77.6	9,196

TABLE 4.1: Comparison of the WATE++ estimations for the E1 and E2 engines with the open domain data⁴ for RR Trent 1000 and GE NX

The nacelle mass estimation is based on Jenkinson et.al⁷⁷ (Equation 4.11). This uses a piecewise linear model based on the take-off thrust of the engine and the weight of thrust reversers was also taken into account while establishing the method. The uncertainty of the method was found⁷⁹ to be $\pm 10\%$. Moreover, the nacelle mass was estimated for the considered engines and it was checked if it equated to 25% of the total engine mass as indicated for modern technology levels⁸⁰. From the quadrature method for the assessment of uncertainties⁸¹ the overall combined uncertainty on the propulsion system mass is $\pm 3.4\%$.

$$m_{nac} = \begin{cases} 6.8 \times F_{N, SLS} & (for F_{N, SLS} < 600kN) \\ 2760 + (2.2 \times F_{N, SLS}) & (for F_{N, SLS} > 600kN) \end{cases} \quad (4.11)$$

4.1.3 Cruise calculation

The cruise calculation was performed based on the particular case of Total Energy Model (Section 4.1.1), where the balance of aircraft thrust and the drag was obtained to maintain constant altitude (H) and constant flight Mach number (M_∞). Initially, the range of the aircraft and the number of steps for the cruise calculation were determined (Figure 4.1). To provide relatively small intervals, the cruise phase was divided in to 40 segments to consider steps shorter than 20 minutes⁸² of steady flight. The initial mass of the aircraft ($m_{A/C}$) was determined based on the result from the BADA climb model (Section 4.1.1). To include the mass of the propulsion system (Section 4.1.2) in the cruise considerations, the difference in weight penalty for the larger engine is added to the mass of the baseline aircraft configuration at the start of cruise ('SoC'). With the use of the aircraft mass for 'SoC' the weight, and therefore the lift requirement ($L_{A/F}$) of the airframe was established. Based on the required lift, the corresponding drag value ($D_{A/F}$) for the clean-wing airframe was evaluated and corrected (F_{inst}) based on the results from the CFD for the considered configurations. For each configuration, the overall lift including the vertical component of thrust was considered. Furthermore, the standard net thrust requirement (F_N) was set based on the drag of the aircraft. However, the drag of the configuration contains the thrust dependent terms and therefore it was an iterative process to balance the thrust and drag at a given lift coefficient. Once the

balance of forces on the aircraft was achieved the first step of the cruise computation was computed and the fuel burn was evaluated. The mass of fuel burnt at that step was subtracted from the ‘SoC’ aircraft mass and the procedure was repeated over the next step distance. The procedure was continued until the traveled distance for the aircraft reached the prescribed range. Finally, the integrated fuel burn value for the engine-aircraft configuration was reported.

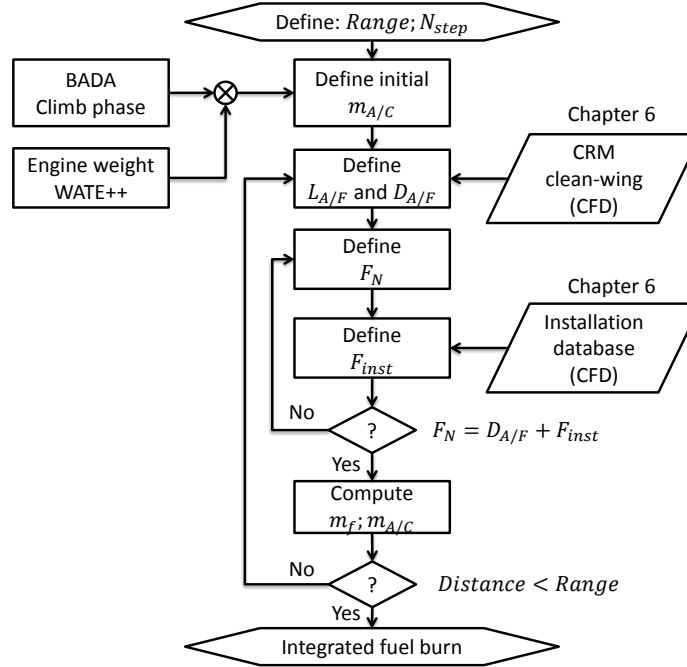


FIGURE 4.1: Flow chart for the integrated fuel burn calculation during the cruise phase of flight; the CFD results discussed in Chapter 6

4.2 Engine design and performance methods

The cycle design for the turbofan engine was completed with the engine performance code with an off-design capability (Section 4.2.1). Based on the engine performance model, the design of the engine geometry was carried out (Section 4.2.2) to represent a relatively modern turbofan.

4.2.1 Engine performance modelling

The engine performance modelling was performed with the use of Turbomatch^{83,84}. Turbomatch is a zero-dimensional code for aerothermal analysis used in Cranfield for performance calculation. The code has been validated against the experimental data with the agreement of 6% for engine thrust and engine massflows by McMillan⁸³. Moreover, the code had been successfully used for a wide range of gas turbine engines^{85–87}.

The code uses discrete component maps for the determination of each component performance. The method solves for the mass and energy balance between the various engine components (Figure 4.2). Thermodynamic engine cycles can be defined with major engine components' efficiencies. The cycles can be determined for existing engines as well as for possible future cases. The important feature of Turbomatch is the off-design analysis. During the off-design calculation, Turbomatch refers to components' performance maps to assess engine performance as engine operating conditions or engine throttling changed. An aircraft mission profile can be defined and the thrust requirement during certain flight modes set, based on the aircraft performance tool. As a result, a full engine performance model can be developed for a required specification to provide a range of realistic CFD boundary conditions for an engine. Within the current scope of work, the engine is assumed to be operating exclusively at steady-state conditions for each mission operating point.

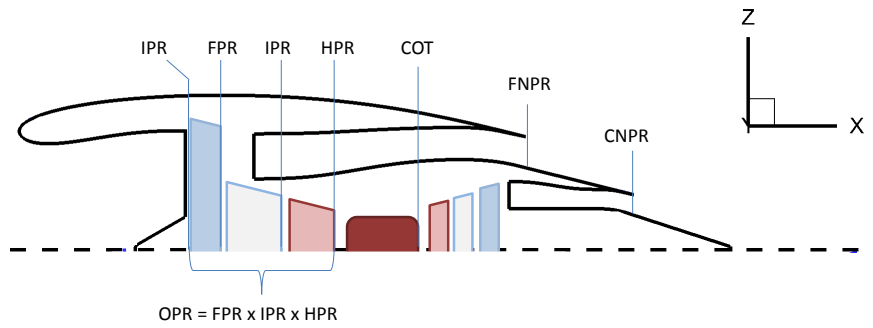


FIGURE 4.2: Sketch of the three-spool engine architecture with key performance parameters

Based on the airframe performance at the cruise Mach number, a net thrust requirement at a cruise altitude was estimated. To determine the exact cycle of the baseline engine, a set of required performance characteristics for an engine were defined. Based on the thrust requirement and assumed fan diameter, an initial engine sizing and architecture were determined. It was decided to use a three-spool architecture for the engines in current research (Figure 4.2). Decisions on engine technology level are based on open source data for engines of comparable thrust class⁴. The bounds for the combustor outlet temperature (COT), or compressor pressure ratios for each shaft (FPR, IPR, HPR), as well as current component isentropic efficiencies were specified to meet modern technology level⁸⁸. A detailed fine-tuning of the engine cycles was completed with the use of an optimisation routine called “Automatch” (see Appendix B.1). Automatch fine-tunes engine cycles to provide the best SFC for a required intake massflow, engine thrust with respect to the technology constraints.

4.2.2 Engine geometry generation

To determine the engine installation effects for modern engine turbofan engines, it was necessary to design engine geometries for the considered engine thermodynamic cycles. The achievement of this goal required a reliable engine geometry generation tool to be developed. A spreadsheet based tool with an outline of engine geometry was initially developed by Christie⁸⁹. Within the current project, the existing tool was enhanced by the implementation of parametric design guidelines for engine nacelles and core cowls to broaden the range of possible designs. Moreover, an enhancement in the geometry definition was implemented by the use of the Class Shape Transformation (CST) curves^{90,91} for the nacelle aerolines. As a result a Geometric Engine Modeller (GEM) python routine was created. Furthermore, the fully parametric definition of nacelle geometry was developed and implemented by Heidebrecht et al.^{92,93}. The parameterisation of the nozzle designs and implementation of the CST curves in the exhaust ducts was carried out by Goulos et al.^{94,95}. As a result, the Geometric Engine Modeller Including Nozzle Installation (GEMINI) framework (Appendix B.2) was developed to rapidly design and evaluate the aerodynamic performance of the axisymmetric aero-engines with separate-jet exhaust. Furthermore, GEMINI was used to design the axisymmetric engines used within current study (Figure 4.3), thus the more detailed structure of framework is outlined in Appendix B.2. The engine fan face, by-pass duct (BPD) entry and low pressure turbine (LPT) exit are sized based on the engine massflows from the performance model and the design is completed with remaining aerolines (see Appendix B.2). Based on the 2D definition of engine geometry, the 3D geometry was created by revolution around the engine axis.

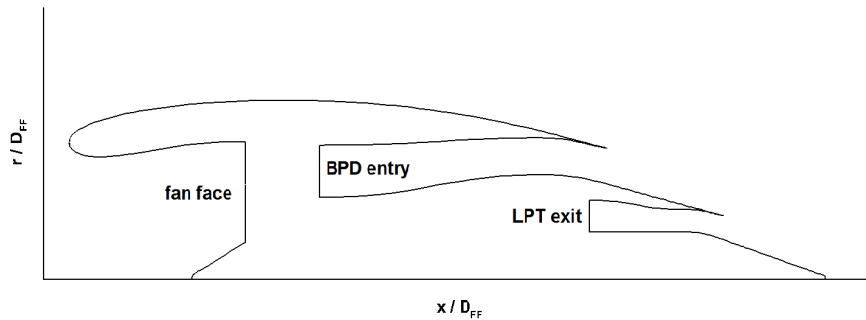


FIGURE 4.3: An example of the engine geometry created by GEMINI

4.3 Computational Fluid Dynamics Methodology

The core of the current research is largely completed by the means of the Computational Fluid Dynamics (CFD). Within this section a description of the methodology used in the assessment of engine installation effects is described. Based on the revision of the current state of the art (Section 2.3), a coherent strategy was developed that consisted

of the choice of the domain size (Section 4.3.1), meshing strategy (Section 4.3.2), solver settings (Section 4.3.3) and turbulence modelling (Section 4.3.4).

4.3.1 Computational Domain

To avoid the pressure interaction from the far-field boundary, it is necessary for the CFD to use considerable domain sizes. Therefore, a range of computations were performed to determine the impact of the domain size on the calculated value of drag and thrust for the isolated nacelles and engines. Greater details on the computed studies are located in Section C.1.

Configuration	Dimensionality	Domain type	Tested sizes	Selected size
ARA	2D - axisym.	Semi-circular	60, 90, 120, 150 D_{max}	60 D_{max}
DLR-TPS	2D - axisym.	Semi-circular	75 D_{max}	75 D_{max}
DSFRN	3D - sym.	Hemi-spherical	20, 40, 60 D_{max}	60 D_{max}
Iso. Eng.	3D - sym.	Hemi-spherical	10, 30 50 D_{max}	50 D_{max}
CRM	3D- sym.	Hemi-spherical	100 c_{ref}	100 c_{ref}

TABLE 4.2: Domain specification for ARA Cowls¹⁹, DLR-TPS³², DSFRN³⁴, isolated engines and CRM⁴⁶ configurations

For the studies on isolated nacelles, the study showed that a domain size of 60 nacelle maximum diameters (D_{max}) was sufficiently large to ensure no spurious effects on the flow field. The $60D_{max}$ was used for the validation work on the intake spillage characteristics (Figure 4.4a) The effect of the domain size on the numerical simulation of the exhaust system was investigated and overall, the $60D_{max}$ domain was adopted (Figure 4.4b) for the further study on DSFRN (Section 5.1.3). For the studies on isolated engines, the extent of the domain size was tested and it was showed that the domain size of 50 nacelle D_{max} is sufficiently large to ensure no spurious effects (Figure 4.4c).

For the aircraft studies, the domain size was based on the experience of the 4th Drag Prediction Workshop⁹⁶. The aircraft was placed in the numerical domain with a size of $100c_{ref}$ (Figure 4.4d). For the engine-aircraft configuration the domain was also in the size of $100c_{ref}$.

4.3.2 Meshing strategies

Throughout the study, only structured multi-block meshes are considered. Based on the complexity of the geometry, an adequate blocking strategy was required (Table 4.3). The simple 2D axisymmetric meshes for the ARA Cowls and DLR-TPS consisted of a external O-grid meshes with the central H-mesh block and a boundary layer C-grid around the geometry. The more complex 3D mesh topology of the DSFRN was created (Table 4.3) as a external O-grid in the far-field with the central H-grid and two C-grid

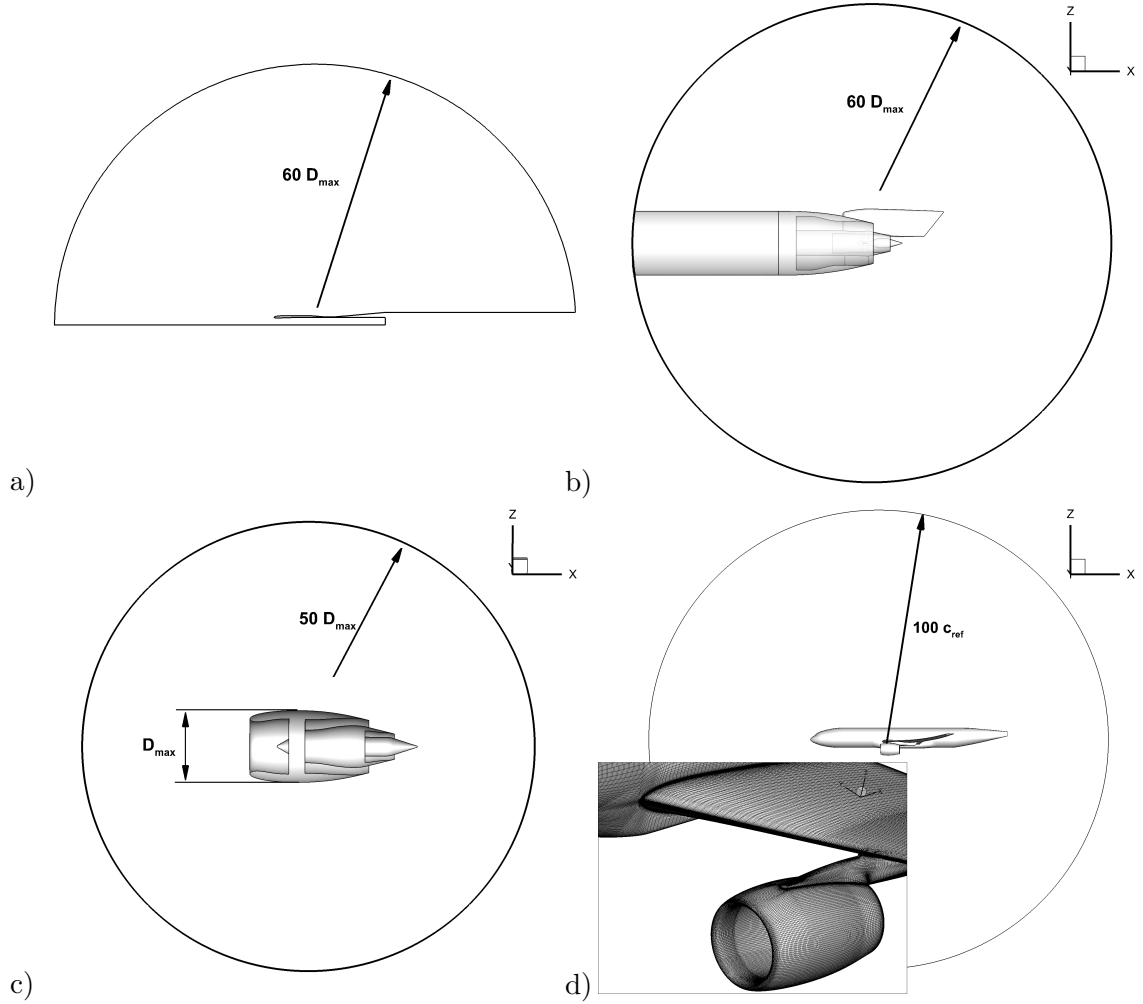


FIGURE 4.4: The domain size for a) the isolated ARA fan-cowl b) the isolated DSFRN exhaust system c) the isolated engine and d) the aircraft configurations

structures for the fan and core cowls. An additional O-grid around the pylon geometry and by-pass bifurcations was implemented. The most complex geometry considered within the study consisted of airframe, engine and pylon geometry (CRM WBT0EP; Table 4.3). As a result, the typical O-grid around the far-field was created with the H-grid central block. Furthermore, the O-grid structure was created around the fuselage and it intersected with the independent O-grids around the wing and the tail. The two C-grid topology was followed for the under-wing engine and pylon structure was surrounded with an additional independent O-grid (Table 4.3). All meshes were created with the boundary layer blocks, where the first cell height was set to satisfy the $y^+ = 1$ criterion.

For the aircraft configuration (WBT0), based on the gridding guidelines from DPW^{13,64,65} (Section 2.3.3.1), a “medium” density mesh (10.3×10^6) for the CRM clean-wing configuration (WBT0) was created with the O-H-O-O topology. The node distribution and element growth ratios were based on the DPW4 gridding guidelines⁶⁴, however, and additional care was dedicated to the achievement of mesh smoothness and orthogonal

Configurations	Mesh topology	Number of blocks	y^+
<i>ARA</i>	O-H-C	13	1
<i>DLR – TPS</i>	O-H-C-C	34	1
<i>DSFRN</i>	O-H-C-C-O	311	1
<i>Iso.Eng.</i>	O-H-C-C	123	1
<i>CRM WBT0</i>	O-H-O-O	154	1
<i>CRM WBT0NP</i>	O-H-O-O-C-O	327	1
<i>CRM WBT0E</i>	O-H-O-O-C-C	373	1
<i>CRM WBT0EP</i>	O-H-O-O-C-C-O	439	1

TABLE 4.3: Mesh topology for ARA Cowl¹⁹, DLR-TPS³², DSFRN³⁴, isolated engines and CRM⁴⁶ configurations

quality. Moreover, to enable the enhanced wall treatment, the boundary layer mesh had 25 elements in the physical boundary layer and the first cell height was set to result in $y^+ = 1$. In total four mesh densities were considered and the more detailed study (Section 5.2.1) indicated that the “fine” mesh resolution (16.1×10^6) is advisable. For the meshes with through flow nacelles (TFN), a structured mesh with a comparable blocking strategy to the clean wing configuration (O-H-O-O) was followed and to accommodate the presence of through flow nacelles the C-grid blocking around the nacelle was arranged as concentric with the engine axis. The meshing of the TFN geometry was based on the experience from the previous studies on isolated studies^{79,97,98}. Thus, the following criteria have been added: 40 elements for the nacelle lip, maximum axial spacing on the nacelle $\Delta x = L_{nac}/110$ and minimum 15 elements in the gully between the nacelle and the wing. Those criteria have been merged with the DPW4 gridding guidelines⁶⁴ and the near-wall treatment was facilitated with a $y^+ = 1$. The outcome of the validation activity (Section 5.2.1) indicated that the “fine” mesh resolution (16.1×10^6) is advisable for the installation studies.

Configurations	“coarse”	“medium”	“fine”	“superfine”
ARA	x	0.050	0.040	0.072
DLR-TPS	x	x	0.752	x
DSFRN	8.520	10.744	14.065	x
Iso. Eng.	x	3.023	9.871	33.464
CRM WB	6.872	10.223	14.711	x
CRM WBNP	15.731	22.477	29.712	x
CRM WBT0	6.937	10.655	16.479	24.663
CRM WBT0N	x	x	30.994	x
CRM WBT0NP	x	x	31.316	x
CRM WBT0E	x	x	35.287	x
CRM WBT0EP	x	x	43.653	x

TABLE 4.4: The mesh size in million elements for ARA Cowl¹⁹, DLR-TPS³², DSFRN³⁴, isolated engines and CRM⁴⁶ configurations

To accompany the studies of under-wing nacelles, the isolated nacelle had to be computed

to enable a comparison between the nacelle with and without the presence of the wing. As a result, a 7.2×10^6 element mesh for the through-flow nacelle in isolation was created by following the “fine mesh” nacelle meshing rules as for the WBT0NP mesh. Furthermore, the separate jet geometry is of key interest in the current research. The intake and nacelle meshing strategy were the same to that adopted for the TFN “fine mesh”. The nacelle meshing guidelines were established based on previous studies⁹⁹. However, the bypass duct, core duct, nozzles and core cowl required a modified blocking strategy which also facilitated the boundary layer meshes and followed. Based on the previous experience^{79,97,98}, the meshing criteria for the separate-exhaust system were established and incorporated into the existing meshing criteria for nacelles. Moreover, the 3D engine blocking for an isolated engine with separate exhaust was created that facilitated the boundary layer blocks. As a result a mesh with 9.8×10^6 elements was used for an isolated baseline engine with separate exhausts and the meshing strategy is coherent with the “fine” mesh resolution in the aircraft studies. The details of grid studies on isolated engines are described in Appendix C.2

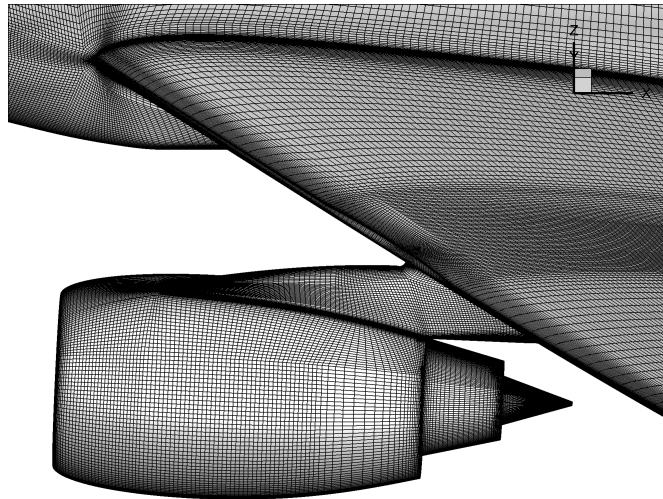


FIGURE 4.5: An example of the “fine” mesh for the aircraft configuration with the engine and the drag-domain pylon

For the full engine geometry under the wing, the meshing strategy was similar to that adopted for the TFN under the wing. As for the isolated separate jet configuration, the mesh had to facilitate the boundary layer meshes for the bypass duct, core duct, nozzles and core cowl. The inclusion of the separate jet meshing rules resulted in the nominal mesh of 35×10^6 elements for the configuration of CRM with engine (Figure 4.5). The mesh was derived from the fine mesh for the clean wing configuration (WBT0).

4.3.3 Solver settings and boundary conditions

The aerodynamic analyses were performed with the use of a compressible RANS method. An implicit flow solver (FLUENT 15.0.7)¹⁰⁰ was used for the entire study with second order discretisation for all terms. The key settings of the solver are as follows:

- double precision
- steady state computation
- Reynolds Averaged Navier Stokes (RANS) solver
- density based solver
- Roe flux-difference splitting scheme
- Green Gauss node based gradient method
- second order spatial discretisation for flow and turbulent terms

The investigated problems are expected to have no transient features, therefore the steady state computation was chosen. To maintain high accuracy the double-precision for the computation was used. The density-based solver was chosen, as the computed problems are at relatively high Mach number and have a considerable density gradients present in the flowfield. The suggested choice¹⁰⁰ for that type of problems is density based solver. The Green Gauss node based gradient method was chosen as it offered more stable convergence in comparison with quicker but more susceptible least squares method.

The domain was simulated with a pressure far-field boundary condition with the specified freestream Mach number and static pressure and static temperature based on International Standard Atmosphere (ISA) and altitude. A centreline symmetry boundary condition was used as half of the aircraft model was computed. In configurations with an engine, the fan face was set to pressure outlet with a target mass-flow option and pressure inlets were used for fan nozzle entry and low pressure turbine exit. All remaining walls were modelled as viscous, adiabatic wall boundary condition. The boundary conditions at the engine inlets and outlets were dictated by the engine performance model.

4.3.4 Turbulence model

Based on the results from 4th Drag Prediction Workshop (DPW)⁹⁶, the $k - \omega$ *SST* turbulence model was used as default turbulence model in the study¹⁰¹. In addition, an independent turbulence study (Section 5.2.2) to compare $k - \omega$ *SST* against other

common turbulence models such as Spalart-Allmaras, $k - \epsilon$ and $k - \omega$ SST $\gamma - \theta$ was completed. The SST $k - \omega$ turbulence model¹⁰¹ is a two-equation eddy-viscosity model. The shear stress transport (SST) formulation combines the modelling of the freestream flow by the $k - \epsilon$ with the modelling of the boundary layer by the standard $k - \omega$. Typically, the SST $k - \omega$ model is advised for its good behaviour in adverse pressure gradients and separating flow. However, the model is also known to produce too much turbulence in stagnation regions and regions with strong acceleration¹⁰².

The use of a $k - \omega$ formulation in the inner parts of the boundary layer makes the use of the model practical down to the wall through the viscous sub-layer. Thus the SST $k - \omega$ model can be used as a Low-Re turbulence model without any additional damping functions. The $k - \omega$ model is one of the commonly used turbulence models. It is a two equation model and it includes two transport equations to represent the turbulent properties of the flow (Equation 4.12-4.13)¹⁰⁰. This allows a two equation model to account for history effects like convection and diffusion of turbulent energy. The first transported variable is turbulent kinetic energy (k), which determines the amount of energy in the turbulence. The second transported variable is the specific dissipation (ω), which determines the scale of the turbulence. Within the system of equations, the turbulent viscosity (μ_t) requires to be modelled. The G_k and G_ω are production terms as in standard $k - \omega$ model for the k and ω , respectively. The Y_k and Y_ω describe the dissipation due to the turbulence and the D_ω is a cross-diffusion term. Meanwhile, the S_k and S_ω are the source terms¹⁰⁰.

$$\frac{\partial \rho k}{\partial t} + \frac{\partial \rho k u_j}{\partial x_j} = \frac{\partial}{\partial x_j} \left[\left(\mu + \frac{\mu_t}{\sigma_k} \right) \frac{\partial k}{\partial x_j} \right] + G_k - Y_k + S_k \quad (4.12)$$

$$\frac{\partial \rho \omega}{\partial t} + \frac{\partial \rho \omega u_j}{\partial x_j} = \frac{\partial}{\partial x_j} \left[\left(\mu + \frac{\mu_t}{\sigma_\omega} \right) \frac{\partial \omega}{\partial x_j} \right] + G_\omega - Y_\omega + D_\omega + S_\omega \quad (4.13)$$

In the free-stream, the $k - \omega$ SST formulation switches to a $k - \epsilon$ behaviour and thereby avoids the common $k - \omega$ problem that the model is too sensitive to the inlet free-stream turbulence properties. The $k - \epsilon$ model is also one of the common turbulence models. Similarly, it is a two equation model and it includes two transport equations to represent the turbulent properties of the flow. The first transported variable is turbulent kinetic energy (k). The second transported variable in this case is the turbulent dissipation (ϵ) and it determines the scale of the turbulence. The $k - \epsilon$ model has been shown to be useful for free-shear layer flows with relatively small pressure gradients¹⁰². Similarly, for wall-bounded and internal flows, the model gives good results only in cases where mean pressure gradients are small. Typically, it is advised not to use $k - \epsilon$ model for the flows containing large adverse pressure gradients.

Part of the validation of the computational method for the aircraft calculation involved the use of the transition SST model (Section 5.2.2). The Transition SST model solves

two additional equations as compared with the two-equation SST model. The two additional transport equations (Equations 4.14-4.15)¹⁰⁰ are for the intermittency and the transitional momentum thickness Reynolds number. The intermittency determines the activation of the SST model. As a result, the intermittency of zero suppresses the production of turbulent kinetic energy and the flow is effectively laminar. In the opposite case, when the intermittency equals unity, the SST model is fully active. The transport of intermittency is described by Equation 4.14, where $P_{\gamma 1}$ and $E_{\gamma 1}$ are transition sources, and where $P_{\gamma 2}$ and $E_{\gamma 2}$ are relaminarisation sources. The momentum thickness Reynolds number controls the transition criterion, against which the local vorticity Reynolds number ($\frac{\rho y^2 S}{\nu}$) is compared. If the local quantity is greater than the critical Reynolds number defined by the transported scalar Reynolds number, the intermittency begins to grow and transition happens. The transport for the transition momentum thickness Reynolds number ($R\tilde{e}_{\theta t}$) is described as in Equation 4.15, where $P_{\theta t}$ is the source term.

$$\frac{\partial(\rho\gamma)}{\partial t} + \frac{\partial(\rho U_j \gamma)}{\partial x_j} = P_{\gamma 1} - E_{\gamma 1} + P_{\gamma 2} - E_{\gamma 2} + \frac{\partial}{\partial x_j} \left[\left(\mu + \frac{\mu_t}{\sigma_\gamma} \right) \frac{\partial \gamma}{\partial x_j} \right] \quad (4.14)$$

$$\frac{\partial(\rho R\tilde{e}_{\theta t})}{\partial t} + \frac{\partial(\rho U_j R\tilde{e}_{\theta t})}{\partial x_j} = P_{\theta t} + \frac{\partial}{\partial x_j} \left[\sigma_{\theta t} (\mu + \mu_t) \frac{\partial R\tilde{e}_{\theta t}}{\partial x_j} \right] \quad (4.15)$$

4.3.5 Iterative convergence

A number of actions were taken to define the iterative convergence (Appendix C.3). As a result, the iterative convergence was monitored based on the scaled residuals of continuity, momentum and turbulent terms, as well as based on the convergence of the the wall forces and mass flows on the inlets and outlets to the domain. The typical convergence resulted in residuals levelling below 10^{-4} . Meanwhile, the amplitude of the oscillation in wall forces did not exceed an equivalent of 0.2 aircraft dc for all the computations, which is 0.1% of total aircraft drag. The convergence for the pressure inlets and outlets converged with no oscillation greater than approximately 0.004% of the respective massflows.

4.4 Current thrust and drag book-keeping system

The full thrust and drag book-keeping system for the engine and airframe configuration was developed to account for all the forces present in the domain. Firstly, the detailed accounting for the engine (Section 4.4.1) is introduced and later it is incorporated to the complete accounting system for the aircraft (Section 4.4.2).

4.4.1 Thrust and drag book-keeping system for the engine

In the current work, a thrust and drag book-keeping system was developed^{97,98} in agreement with publicly known systems^{103,104}. As a result, a modified near-field method for separate jet engines was used⁸⁹. The domain was split into a drag and thrust domain. The forces that acted in the thrust domain (Figure 4.6), i.e. inside the streamtube, were denoted as θ and the forces in the drag domain were denoted as ϕ . Furthermore, gauge stream forces of the flow are evaluated at characteristic engine stations from upstream to downstream infinity and are denoted F_G .

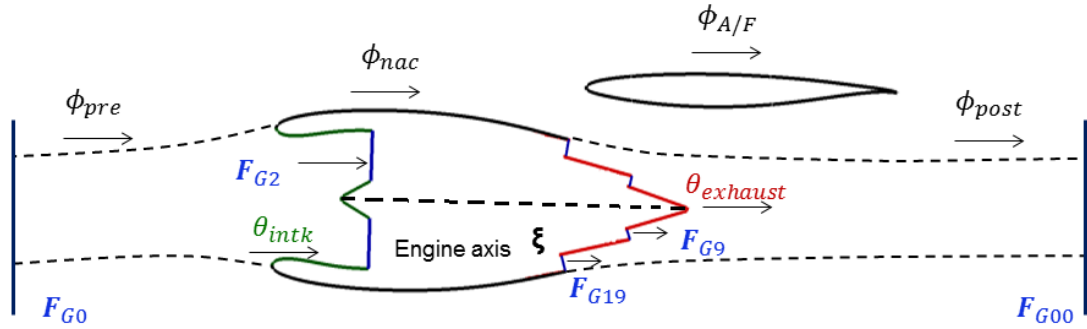


FIGURE 4.6: Decomposition of modified Near Field Method forces acting on the entry streamtube and nacelle

To account for all of the forces that act on an isolated or wing-installed engine, the Net Propulsive Force (NPF) was considered as the difference of the overall engine thrust (F_{ov}) and nacelle drag (D_{nac}) (Equation 4.16). From the balance of forces for the post-exit domain, the unknown values of F_{G00} and ϕ_{post} from Equation 4.16 were substituted by the known terms of stream forces (F_{G9} , F_{G19}) and fluid forces that act on the exhaust surfaces ($\theta_{exhaust}$). As a result, the NPF was expressed in terms of standard net thrust (F_N , Equation 4.17) and fluid forces exerted on exhaust surfaces ($\theta_{exhaust}$), forces on the nacelle (ϕ_{nac}) and force on the pre-entry streamtube (ϕ_{pre}) (Equation 4.18). By the introduction of modified standard net thrust (F_N^*) and modified standard drag for the nacelle (D_{nac}^*) (Equations 4.19 and 4.20), the NPF can be expressed in terms of F_N^* and D_{nac}^* alone (Equation 4.21).

$$NPF = F_{ov} - D_{nac} = F_{G00} - F_{G0} - \phi_{pre} - \phi_{nac} - \phi_{post} \quad (4.16)$$

$$F_N = F_{G9} + F_{G19} - F_{G0} \quad (4.17)$$

$$NPF = F_N - \theta_{exhaust} - \phi_{pre} - \phi_{nac} \quad (4.18)$$

$$D_{nac}^* = \phi_{pre} + \phi_{nac} \quad (4.19)$$

$$F_N^* = F_N - \theta_{exhaust} \quad (4.20)$$

$$NPF = F_N^* - D_{nac}^* \quad (4.21)$$

For the analysis of the nozzle performance, the gross propulsive force of the engine in the engine axis (GPF_ξ) was considered (Equation 4.22) as a sum of gauge stream forces along the engine axis at the fan nozzle ($F_{G19|\xi}$) and core nozzle ($F_{G9|\xi}$) exits. To take into account all the pressure and viscous forces on the surfaces of the core-cowl and the plug, the modified gross propulsive force along the engine axis (GPF_ξ^*) was defined (Equation 4.23). Eventually, the velocity coefficients (Equations 4.24-4.25) were calculated based on the air mass flows in the by-pass duct ($\dot{m}_{19}$) and the core flow mass flow ($\dot{m}_9$) and the estimated ideal velocities at the nozzle exits (V_9^{ideal} and V_{19}^{ideal} , Equation 4.26), where p_{tot}/p_∞ is the ratio of total pressure in the duct to the ambient static pressure. The standard velocity coefficient (Equation 4.24) was based on the GPF_ξ , whereas the modified velocity coefficient (Equation 4.25) was based on the GPF_ξ^* . To determine the magnitude of irrecoverable losses in the by-pass and core ducts, the discharge coefficients for the by-pass duct (C_d^{BPD}) and the core duct (C_d^{Core}) were defined (Equations 4.27-4.28). The discharge coefficient is a ratio of the real mass flow rate obtained in the CFD computation ($\dot{m}_{19}^{CFD}$) referenced by the ideal massflow rate ($\dot{m}_{19}^{ideal}$) that was anticipated based on the flow properties at the beginning of the duct (Equation 4.29).

$$GPF_\xi = F_{G9|\xi} + F_{G19|\xi} \quad (4.22)$$

$$GPF_\xi^* = F_{G9|\xi} + F_{G19|\xi} - \theta_{exhaust|\xi} \quad (4.23)$$

$$C_V = \frac{GPF_\xi}{\dot{m}_{19}^{CFD} \cdot V_{19}^{ideal} + \dot{m}_9^{CFD} \cdot V_9^{ideal}} \quad (4.24)$$

$$C_V^* = \frac{GPF_\xi^*}{\dot{m}_{19}^{CFD} \cdot V_{19}^{ideal} + \dot{m}_9^{CFD} \cdot V_9^{ideal}} \quad (4.25)$$

$$V^{ideal} = \sqrt{\left(\frac{2\gamma RT_t}{\gamma - 1}\right) \cdot \left(1 - \left(\frac{p_\infty}{p_{tot}}\right)^{(\gamma-1)/\gamma}\right)} \quad (4.26)$$

$$C_d^{BPD} = \frac{\dot{m}_{19}^{CFD}}{\dot{m}_{19}^{ideal}} \quad (4.27)$$

$$C_d^{Core} = \frac{\dot{m}_9^{CFD}}{\dot{m}_9^{ideal}} \quad (4.28)$$

$$\dot{m}^{ideal} = A_{th} \cdot p_t \cdot \left(\frac{p_\infty}{p_{tot}} \right)^{\frac{1}{\gamma}} \cdot \frac{2\gamma}{(\gamma - 1) * R * T_t} \cdot \left(1.0 - \left(\frac{p_\infty}{p_{tot}} \right)^{\frac{\gamma-1}{\gamma}} \right)^{0.5} \quad (4.29)$$

4.4.2 Thrust and drag book-keeping system for the aircraft

To compute the balance of forces in drag aero-axis (Figure 4.7) for an engine and an aircraft an overall horizontal force ($F_{X'}$) (Equation 4.30) is considered as the difference between net propulsive force (NPF) and airframe drag ($D_{A/F}$). The airframe drag ($C_{D A/F}$) is an integration of pressure and viscous forces that act in the drag direction on the surface of the wing, body and tail. Similarly, the overall vertical force ($F_{Z'}$) is considered (Equation 4.31) as a summation of the lift on the airframe ($L_{A/F}$), lift on the nacelle (L_{nac}^*) and the vertical component of thrust ($GPF_\xi^* \sin(AoA_{eng})$). Furthermore, to obtain the force coefficients based on the aero forces, the freestream reference condition was used, where ρ is the mass density of the fluid, V_0 is the velocity of the fluid and A_{ref} is the aircraft reference area. As a result, the overall vertical force coefficient ($C_{Z'}$) was computed (Equation 4.32).

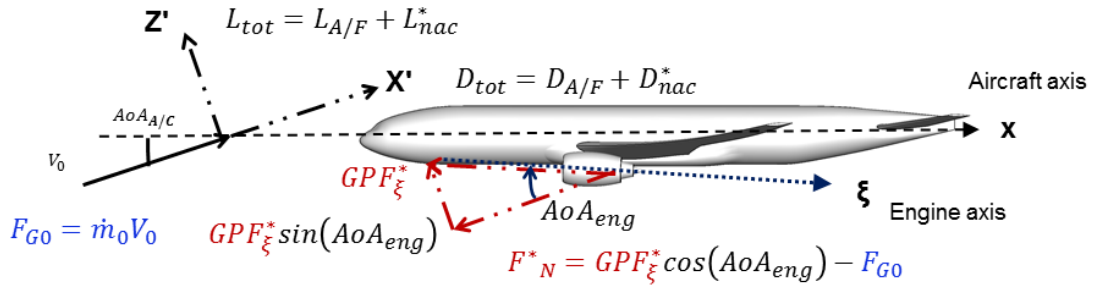


FIGURE 4.7: The sketch of coordinate systems for the aerodynamic forces, aircraft and engine

$$F_{X'} = NPF - D_{A/F} = F_N^* - D_{nac}^* - D_{A/F} \quad (4.30)$$

$$F_{Z'} = L_{A/F} + L_{nac}^* + GPF_\xi^* \sin(AoA_{eng}) \quad (4.31)$$

$$C_{Z'} = \frac{F_{Z'}}{(0.5\rho A_{ref} V_0^2)} \quad (4.32)$$

To assess the overall interference effect for the wing-installed engine configuration, a superposition model (SM) of clean-wing and isolated engine was created. Within this SM estimate, the values of lift and drag of the clean-wing aircraft are summed with the isolated engine thrust, drag and lift data at corresponding incidence ($AoA_{A/C}$). As a result, the overall horizontal force of the superposition model was obtained ($F_{X' SM}$). Furthermore, the aerodynamic interference F_{interf} was the difference between the installed configuration and the SM at equal overall lift force (Equation 4.33). The overall aerodynamic interference consists of the individual effects of installation on drag and thrust terms. The individual effects were referred to as the airframe interference drag (Equation 4.34), the engine interference drag (Equation 4.35) and the interference thrust (Equation 4.36). To consider the force coefficients, the nacelle drag coefficient ($C_{D^* nac}$) was defined (Equation 4.37) based on the nacelle drag and the aircraft total drag coefficient (Equation 4.38) was defined based on the sum of the drag forces for the nacelle and the airframe.

$$F_{interf} = F_{X' inst.} - F_{X' SM} = \Delta F_N^* - \Delta D_{A/F} - \Delta D_{nac}^* \quad (4.33)$$

$$\Delta D_{A/F} = D_{A/F installed} - D_{A/F iso} \quad (4.34)$$

$$\Delta D_{nac}^* = D_{nac installed}^* - D_{nac iso}^* \quad (4.35)$$

$$\Delta F_N^* = F_N^* installed - F_N^* iso \quad (4.36)$$

$$C_{D^* nac} = \frac{D_{nac}^*}{(0.5\rho A_{ref} V_0^2)} \quad (4.37)$$

$$C_{D tot} = C_{D A/F} + C_{D^* nac} \quad (4.38)$$

To determine the installation effect of the engine on the airframe, the installation drag (D_{inst}) was computed (Equation 4.39) as a difference between the engine-airframe configuration and the clean-wing at a constant value of the overall vertical force. The D_{inst} were decomposed to the modified nacelle drag of an isolated engine ($D_{nac iso}^*$) and the interferences on the airframe ($\Delta D_{A/F}$) and on the nacelle (ΔD_{nac}^*). The installation

drag considered the forces only within the drag domain. To define all the effects of engine installation at a constant value of the overall vertical force (Equation 4.40), the thrust of an isolated engine ($F_{N \text{ iso}}$) and the drag of the clean-wing airframe ($D_{A/F \text{ iso}}$) were subtracted from the overall horizontal force ($F_{X'}$). As a result, the F_{inst} is comprised of modified nacelle drag of an isolated engine ($D_{nac \text{ iso}}^*$), interference on the nacelle (ΔD_{nac}^*), interference on the airframe ($\Delta D_{A/F}$) and the interference on thrust (ΔF_N^*), as well as the force on the exhaust ($\theta_{exhaust \text{ iso}}$). The use of F_{inst} that included $\theta_{exhaust \text{ iso}}$ was essential, due to the fact that the engine performance code did not account for the $\theta_{exhaust \text{ iso}}$ in the net thrust definition.

$$\begin{aligned} D_{inst} &= D_{tot \text{ installed}} - D_{tot \text{ clean}} \\ &= D_{nac \text{ iso}}^* + \Delta D_{nac}^* + \Delta D_{A/F} \end{aligned} \quad (4.39)$$

$$\begin{aligned} F_{inst} &= F_{X'} - F_{N \text{ iso}} - D_{A/F \text{ iso}} \\ &= -\theta_{exhaust \text{ iso}} - D_{nac \text{ iso}}^* + \Delta F_N^* - \Delta D_{nac}^* - \Delta D_{A/F} \end{aligned} \quad (4.40)$$

The alternative definition of installation drag (X_{inst}) was proposed within the ALVAST studies^{53,55} (Appendix A.2), and it was adopted for the purposes of direct comparison (Equation 4.41). To obtain X_{inst} (Equation 4.41), the modified thrust of an isolated engine ($F_{N \text{ iso}}^*$) and the drag of the clean-wing airframe ($D_{A/F \text{ iso}}$) were subtracted from the overall horizontal force ($F_{X'}$) at a constant value of the overall vertical force. As a result, the X_{inst} comprised of modified nacelle drag of an isolated engine ($D_{nac \text{ iso}}^*$), interference on the nacelle (ΔD_{nac}^*), interference on the airframe ($\Delta D_{A/F}$) and the interference on thrust (ΔF_N^*).

$$\begin{aligned} X_{inst} &= F_{X'} - F_{N \text{ iso}}^* - D_{A/F \text{ iso}} \\ &= D_{nac \text{ iso}}^* + \Delta D_{nac}^* + \Delta D_{A/F} - \Delta F_N^* \end{aligned} \quad (4.41)$$

To visualise the installation drag (D_{inst}), it was presented (Figure 4.8) in form of the installation drag coefficient ($\Delta C_{D \text{ inst}}$) as a difference between the total drag polars ($C_{D \text{ tot}}$) at a constant value of the overall lift coefficient ($C_{Z'}$). The $\Delta C_{D \text{ inst}}$ was a result of an increase of the drag due to the installation considered at a constant angle of attack ($\Delta C_{D \text{ inst}|AoA}$) and the portion of the lift induced drag that arose due to the re-trimming of the aircraft and compensation for the incurred loss of lift due to the installation ($\Delta C_{L \text{ inst}|AoA}$).

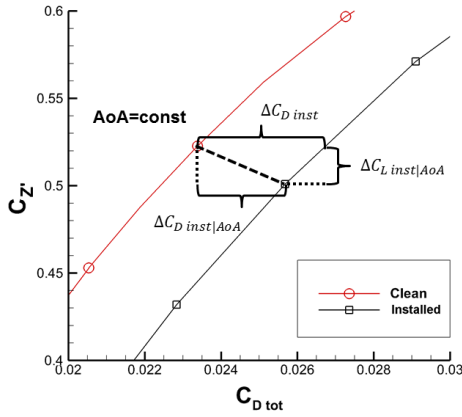


FIGURE 4.8: Definition of installation drag at constant angle of attack ($\Delta C_{D\ inst|AoA}$) and at constant overall vertical force ($\Delta C_{D\ inst}$); definition of the loss of lift due to the installation ($\Delta C_{L\ inst|AoA}$)

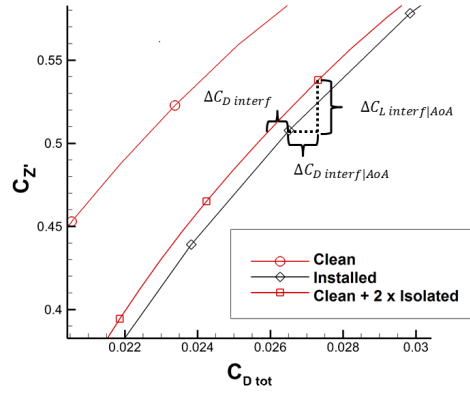


FIGURE 4.9: Definition of interference drag at constant angle of attack ($\Delta C_{D\ interf|AoA}$) and at constant overall vertical force ($\Delta C_{D\ interf}$); definition of the interference loss of lift due to the installation at constant angle of attack ($\Delta C_{L\ interf|AoA}$)

4.5 Summary

In summary, a full numerical methodology was established for the evaluation of the engine installation effect by the means of the RANS computation. The methodology is a subject to the validation in the following Chapter 5. Moreover, the methodology for the calculations of engine performance and airframe performance with the use of zero-dimensional codes was briefly outlined. Overall, the methodology allows for the quantification of the engine installation effect and the evaluation of integrated cruise fuel burn.

Validation of numerical methods

“Verification: solve the equations right;
Validation: solve the right equations”

P. J. Roache

The validation of numerical methods for the computation of aerodynamic effects due to the presence of an engine was considered. A range of geometric configurations needed to be analysed, and therefore isolated engines and nacelles, an isolated airframe, and a combined airframe-nacelle configurations were simulated and the results were compared against respective experimental data. The numerical campaign for the isolated engines involved the assessment of the CFD method for the calculation of thrust and drag characteristics (Section 5.1) of isolated engines over a range of power settings. Furthermore, the evaluation of numerical methods in the determination of the total drag for an aircraft configuration was completed (Section 5.2). The interest was to evaluate the ability of CFD to determine the installation drag and the loss of the lift due to the installation of a through-flow nacelle (Section 5.3). The use of a numerical method enabled the decomposition of the total aircraft drag into component drag contributions and the analysis of the aerodynamic interference incurred due to the mutual interaction between the components (Section 6.1). The chapter was concluded with a brief summary and conclusions (Section 5.4) that impact the further work.

5.1 Validation cases for isolated engines

5.1.1 ARA Cows

To validate the capability of numerical methods to assess the isolated nacelle drag, drag rise Mach number and spillage characteristics, a range of computations for the Aircraft Research Association’s (ARA) cows²⁴ were completed. The experimental range of Reynolds numbers from 1.5 to 2.24×10^6 was considered where the Reynolds number

was based on nacelle maximum diameter. Based on the Reynolds number the other flow characteristics were specified as in the experiment. The total pressure was constant at 101325 Pa and static temperature was constant at 285.7 K with the remaining flow conditions computed for the range of freestream Mach numbers (M_∞) and mass flow capture ratios (MFCR). The standard flow properties for dry air were used with $\gamma = 1.4$ and $R = 287.1 \text{ J/(kg K)}$.

To assess the ability of CFD to determine a drag rise Mach number, a range of M_∞ from 0.4 to $M=0.9$ for a single $MFCR = 0.7$ were computed (Figure 5.1a). In addition, the spillage characteristics were assessed for a typical M_∞ of 0.85 and a range of $MFCR$ from 0.4 to 0.78 (Figure 5.1b). To test the drag characteristics for different cowl designs, a total of two cowls from the experimental study¹⁹ were chosen, namely Cowl 1 and Cowl 3. The drag reported in the experiment consisted of the modified nacelle drag ($C_{D^* \text{ nac}}$) until the nacelle trailing edge and the sting force (ϕ_{sting}) until the measurement rake (Equation 5.1). The sum of $C_{D^* \text{ nac}}$ and ϕ_{sting} will be referred to as corrected drag ($C_{D'}$).

$$C_{D'} = C_{D^* \text{ nac}} + \phi_{sting} \quad (5.1)$$

The absolute accuracy in the determination of the corrected nacelle drag ($C_{D'}$) over the range of Mach numbers was determined for both Cowls 1 and 3 (Figure 5.1a). In the pre-drag rise region, there was an under-prediction in the order of -5% and over-prediction of +3% of the $C_{D'}$ for the CFD results as compared with the measurements for the Cowl 1 and Cowl 3, respectively. As expected, the CFD results for the Cowl 1 align with the CFD results for the Cowl 3 in the pre-drag rise region and it is explained by a similar design with only difference in the value of the forebody radius. The discrepancy between the experimental datasets for both cowls indicates the potential magnitude of the error due to repeatability. Post drag-rise (Figure 5.1a), the accuracy for the prediction of the $C_{D'}$ is 7% for the Cowl 1 and 3% for the Cowl 3. Furthermore, the absolute accuracy in the determination of $C_{D'}$ over a range of MFCRs was evaluated (Figure 5.1b).

The drag characteristics ($C_{D'}$) from the experimental results for Cowl 1 and Cowl 3 were broadly constant across the range of Mach numbers from 0.7 to 0.8 (Figure 5.1a). The insensitivity of the experimental results to modest changes in drag values can be explained by an insufficient amount of Pitot probes in the measurement rig to indicate such subtle changes in the flow below the drag rise condition. The numerical results showed an initial slight decrease in $C_{D'}$ as M_∞ increased from the 0.7 to 0.8. This effect can be explained by the subtle increase of Reynolds number and an associated reduction in friction drag at conditions below drag rise. As well as determining the sensitivity of the nacelle drag to both $MFCR$ and M_∞ , a key parameter for nacelle aerodynamic performance is the critical drag rise Mach number. It defines the Mach number at which the rise of the drag value occurs due to the rapid increase in the wave drag in the transonic flow. The analysis of a slope of the drag curve against the gradient criterion

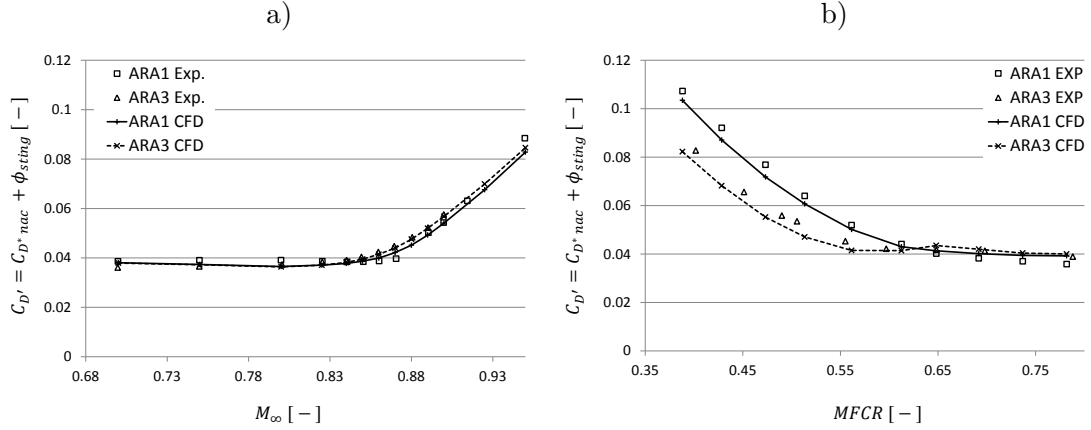


FIGURE 5.1: a) $(C_{D'})$ as a function of Mach number for ARA cowls 1 and 3 at $MFCR = 0.7$; $AoA_{eng} = 0^\circ$; b) $(C_{D'})$ as a function of MFCR for ARA cowls 1 and 3 at $Re = 2.2 \times 10^6$; $M_\infty = 0.85$; $AoA_{eng} = 0^\circ$; CFD compared with measurements²⁴

(Figure 5.1a) provides an adequate assessment of drag rise Mach number (M_{DR}). The drag rise Mach number was identified¹⁰⁵ as the value of the free-stream Mach number, where the gradient of drag as a function of Mach number equals $dC_{D'}/dM = 0.05$. This definition of drag rise Mach number was applied to both numerical and experimental datasets. A gradient was computed based on the experimental resolution and a piecewise linear fit through the gradient results was completed to determine the drag rise Mach number based on the imposed criterion. With an increase in Mach number from 0.7 in the experiment, both cowls exhibited a typical drag rise characteristic with M_{DR} of 0.86 and 0.84, for Cowl 1 and Cowl 3, respectively. This was expected based on the design of Cowl 1 which has a higher M_{DR} due to the larger leading edge radius. The CFD showed broadly the same drag characteristics for both cowls as compared with the experiment. The CFD captures the relatively constant drag as the Mach number increases. Moreover, the drag rise location broadly matched the experimental results. Although the CFD simulations mainly captured the M_{DR} , the calculated M_{DR} are -0.015 greater than the measured value for Cowl 1 and $\Delta M_{DR} = -0.01$ less for Cowl 3.

To analyse the accuracy of the drag coefficient computation across the range of MFCRs, the calculations with constant $M_\infty = 0.85$ were considered (Figure 5.1b) for a range of MFCRs from 0.4 to 0.73. For a constant M_∞ of 0.85, as the MFCR was reduced, the spillage drag increased for both Cowl 1 and Cowl 3 (Figure 5.1b). Unexpectedly, Cowl 1 with the larger leading edge radius of curvature, was notably more sensitive to the reduction in MFCR whereby at an extremely low MFCR of 0.40 the $C_{D'}$ increased from 0.04 to 0.1. The effect was milder for the Cowl 3 where the measured $C_{D'}$ increased from 0.04 to 0.08 for a MFCR of 0.40. Even though, the increase of forebody radius led to a reduction in spillage drag as compared with the NACA-1 profiles²⁴. Cowl 3 with lower forebody radius than Cowl 1 exhibited better performance due to the reduced wave drag. The conclusions of Langley et al.²⁴ were confirmed by an analysis of the

pressure distribution from current computations at $M_\infty = 0.85$ and $MFCR = 0.47$ (Figure 5.2). Even though a higher peak Mach number by $\Delta C_p = -0.05$ is noted for Cowl 3 due to the greater curvature at the leading edge ($x/D_{max} = 0.25$), the further distribution of curvature resulted in a deceleration of the flow prior to the shock and resulted in a weaker shock as compared with Cowl 1. The increase in $C_{D'}$ with reducing $MFCR$ gives rise to a critical mass flow capture ratio ($MFCR_{CR}$). A similar gradient criterion was considered for the determination of the $MFCR_{CR}$ and was defined as $-dC_{D'}/dMFCR = 0.05$, as an analogy to the gradient criterion¹⁰⁵ for the drag rise Mach number. The $MFCR_{CR}$ was numerically evaluated for the CFD results within current study as $MFCR_{CR}=0.65$ for Cowl 1 and $MFCR_{CR} = 0.59$ for Cowl 3. In comparison with the experimental results, the CFD results indicated that the $MFCR_{CR}$ occurred at a lower MFCR and the differences between the experimental and numerical results were $\Delta MFCR_{CR} = +0.025$ and $\Delta MFCR_{CR} = +0.02$ for Cows 1 and 3, respectively.

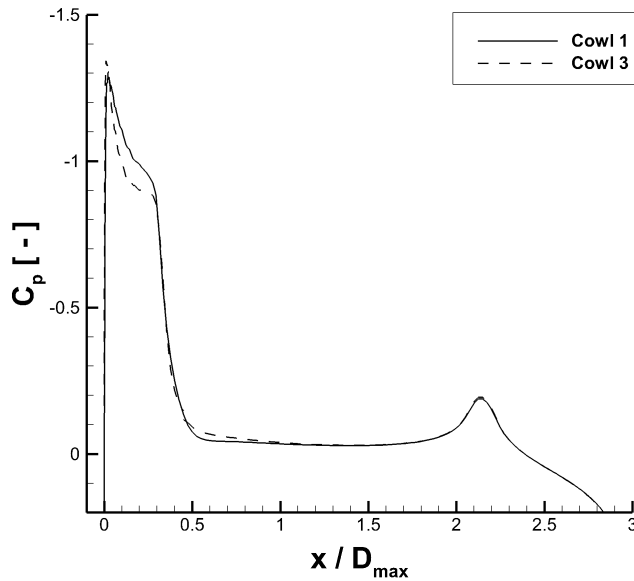


FIGURE 5.2: Pressure distribution (C_p) on the ARA Cows 1 and 3 at $M_\infty = 0.85$; $MFCR = 0.47$; $AoA_{eng} = 0^\circ$

Overall, it was found that current numerical methods broadly captured the key nacelle drag characteristics and provided accuracy of 1.5% for the assessment of the critical $MFCR$ and $\Delta M_{DR} = \pm 0.015$ for the drag rise Mach number. These results are considered to be satisfactory for the use of the CFD for the assessment of nacelle drag within the context of creation of preliminary design tools.

5.1.2 DLR TPS

An experimental test case of a turbine powered simulator (TPS)^{32,33} was chosen to validate the numerical method for the determination of engine thrust characteristics.

However, the axisymmetric geometry of the TPS case is publicly available with the exclusion of the pylon geometry, which allowed for 2D axisymmetric simulations. The test case was computed for a range of operating conditions and compared to a publicly available experimental dataset. The conditions from the experimental setup were recreated³³. As a result, the TPS was simulated for a range of the fan pressure ratio (FPR) that varied from 1.2 to 1.6 at highest available freestream Mach number of 0.17. The range of FPR from 1.2 to 1.6 corresponds to fan nozzle pressure ratios (FNPR) of approximately 1.21 to 1.64. However, an accurate conversion is not possible based on the provided dataset.

The 2D axisymmetric simulation of the TPS was performed and as expected it resulted in approximately +9% larger massflow than for the 3D model in the experiment. Unfortunately, the exact 3D geometry of the exhaust system with the pylon is not available. Therefore, to take into account the pylon blockage that was present in the experiment, it was estimated based on the angular blockage, specified by DLR^{32,33}, that approximately -8% of the by-pass nozzle area was blocked by the pylon. Therefore, a correction applied to the computed CFD data was applied to account for the reduction in massflow and the loss of thrust due to the pylon. A correction of the numerical results by a reduction in the massflow of the fan nozzle by -8% was applied. (Figure 5.3a)). The reduction in the accounted massflow, translates almost linearly to the computation of the overall gross thrust (F_{Gross})(Figure 5.3b)). The overall discrepancy in the F_{Gross} after the applied correction was +5% within the tested range of fan pressure ratios (FPR).

Furthermore, the analysis of the case at the highest power setting (FPR=1.6) was performed to assess the capabilities of CFD in capturing the detailed flow-field (Figure 5.4a)). The numerical solution was compared against the experimental data in terms of the isentropic Mach number in the by-pass and core ducts (Equation 5.2)(Figure 5.4b)), where the total pressure at the entry to the corresponding duct was used as a reference. In general, the CFD correctly captures the trends of the distribution of isentropic Mach number. In the by-pass duct, the numerical solution presents a region of a modest M_{isen} of 0.45 in the central section, which is followed by a rapid increase in M_{isen} to 0.95 closer to the exit plane of the nozzle (Figure 5.4a)). The discrepancy in determination of local minimum for the isentropic Mach number (Figure 5.4b)) in the by-pass duct is roughly 10% of M_{isen} at $x/D_{max} = 1.55$. Even though the flow was gradually accelerated till the nozzle exit, the by-pass nozzle remained un-choked (Figure 5.4a)). The numerical solution indicated presence of very weak shocklets near to the nozzle exit as a result of local acceleration (Figure 5.4a)). As compared to the experimental data, the discrepancy in isentropic Mach number (Figure 5.4b)) was roughly 5-6% at the surfaces of the core cowl and external plug. Overall, the numerical computations captured the key features of the flow. Given the fact that the detailed geometry was not available, the basic performance characteristics of the separate-jet exhaust system were predicted reasonably well.

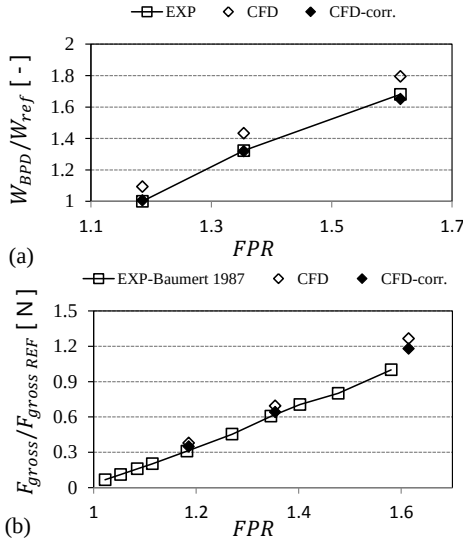


FIGURE 5.3: The DLR TPS test case at $M_\infty = 0.17$ at $FPR=1.2$ for (a) Mass flow in the by-pass duct (W_{Bypass}) as fraction of reference massflow (W_{ref}) compared with experimental results³²; (b) Overall gross thrust of the TPS (F_{Gross}) as fraction of reference gross thrust from the experiment ($F_{Gross REF}$) compared with experimental results³³

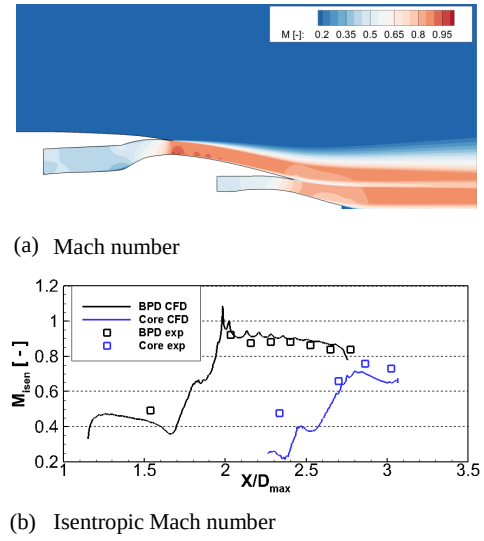


FIGURE 5.4: (a) Contours of Mach number for the DLR TPS test case at $M_\infty = 0.17$ at $FPR=1.6$ ($FNPR \approx 1.64$) and (b) Isentropic Mach number in the by-pass duct compared with measurements³³

$$M_{isen} = \sqrt{\left(\left(\frac{p_t}{p_\infty} \right)^{\frac{(\gamma-1)}{\gamma}} - 1 \right) \cdot \frac{2}{(\gamma-1)}} \quad (5.2)$$

5.1.3 Dual Separate Flow Reference Nozzle

A broader numerical campaign was established in collaboration with Otter et al.^{106, 107} to validate the ability of the numerical methods used in this study to calculate the nozzle performance metrics for separate-jet exhaust systems (Figure 5.5). The computations for the Dual Separate Flow Reference Nozzle (DSFRN) from this study^{106, 107} were compared with the experimental results³⁴. The ability of CFD to calculate nozzle performance metrics was quantified (Equation 5.3). Three key performance parameters were considered such as thrust coefficient (C_V) and discharge coefficients for both core (C_d^{Core}) and by-pass (C_d^{BPD}).

$$\Delta C(\%) = \frac{(C^{Exp} - C^{CFD})}{C^{Exp}} \cdot 100 \quad (5.3)$$

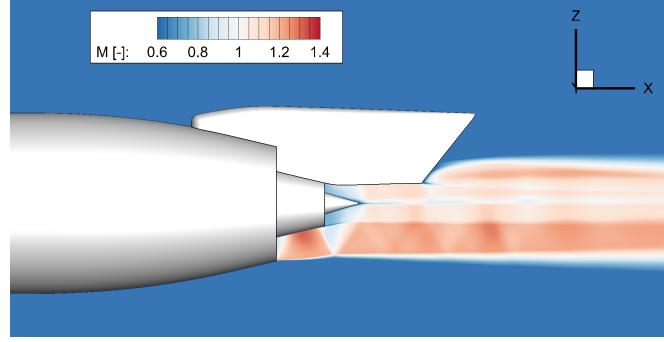


FIGURE 5.5: Contours of Mach numbers for DSFRN; $M_\infty = 0.01$, $FNPR = 2.6$

Across the range of fan nozzle pressure ratios (FNPR) from 1.4 to 2.8 (Figure 5.6c), the greatest deviation from the experimental C_V was 0.82% at a FNPR of 1.4 and the RMS deviation between the experimental and computational thrust coefficient was 0.31%. Due to the main interest in the cruise performance of engines, the FNPRs from 2.2 to 2.6 (Figure 5.5) for the DSFRN were the most significant. The agreement for C_V as compared with experimental data was better than 0.1% for the FNPRs between 1.4 and 2.6. To verify if the agreement was acceptable, the results were compared with the average values obtained in 2nd Propulsion Aerodynamics Workshop³⁵. The C_V was consistently closer to the experimental data as compared with the average of results from 2nd Propulsion Aerodynamics Workshop³⁵ (PAW2). As a result, the computations were in line with current CFD practice. Furthermore, for C_d^{BPD} the largest deviation was found to be 0.58% at a FNPR of 1.8 with an RMS deviation across the range of FNPR investigated of 0.48% (Figure 5.6a). Meanwhile, the greatest deviation of C_d^{Core} was 0.76% at a FNPR of 2.4 (Figure 5.6b) and an RMS deviation across the range of CNPR investigated of 0.45%. For the CNPRs in range from 1.3 to 1.7, the deviation is less than 0.3%. More results obtained during the campaign were presented in Appendix E.2 and Table E.3.

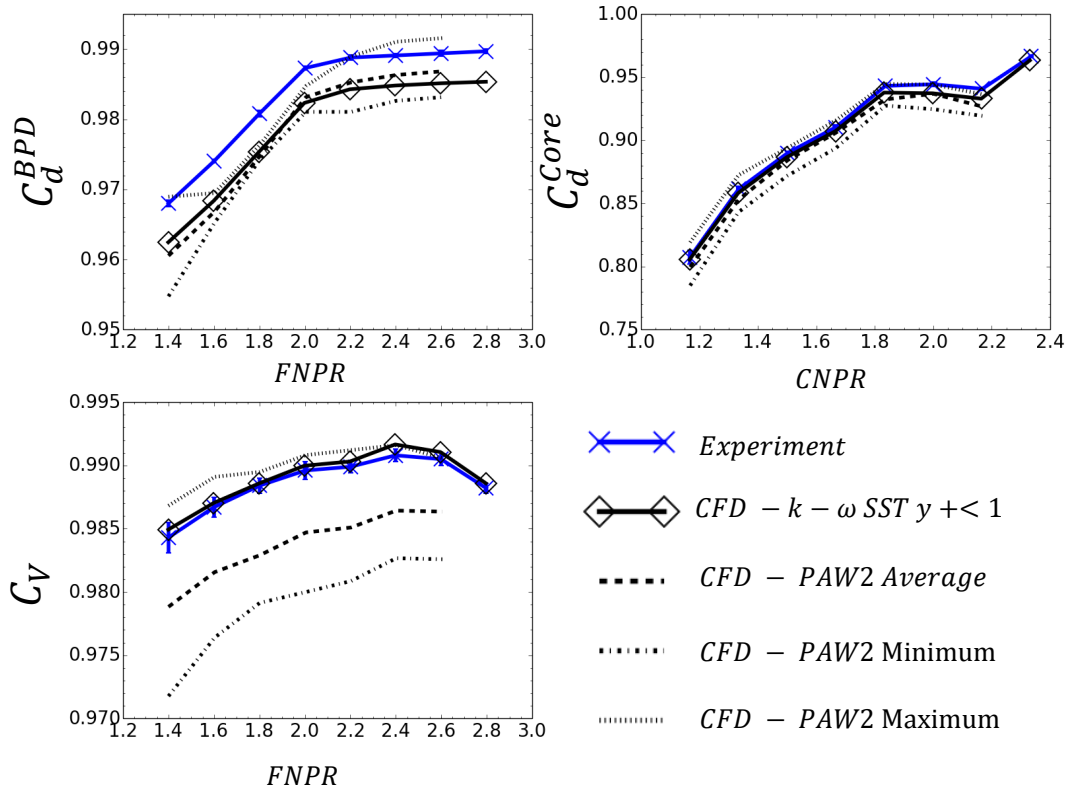


FIGURE 5.6: Variation of (a) C_d^{BPD} , (b) C_d^{Core} and (c) C_V with FNPR and CNPR for the DSFRN. Current numerical results¹⁰⁶ (solid black line and diamonds) compared with the experimental results (blue line) and the range of PAW2 results³⁵ (dashed lines)

5.2 Clean-wing CRM

For the aircraft studies, the NASA Common Research Model⁴⁶ was used and it is representative of a contemporary transonic transport aircraft. The original rigid wind tunnel geometry was used in the current study as the deflected geometry was not publicly available. The issue of aerodynamic twist of the wing was evaluated by Rivers et al.¹⁵ who reported that the twist of the wing due to the wing loading had no impact on the overall lift-drag polar. The computations were carried out for a range of angles of attack ($AoA_{A/C}$) from 0° to 4° in half a degree steps. The experimental setup was recreated¹⁵ for $Re = 5 \times 10^6$ and $M = 0.85$ to compare with the results of the 4th Drag Prediction Workshop (DPW)⁹⁶. Moreover, the sensitivity of the numerical method to the mesh density (Section 5.2.1), choice of the turbulence model (Section 5.2.2) and Reynolds number (Section 5.2.3) were determined.

5.2.1 Grid independence study

The impact of spatial resolution on aircraft drag $C_{D\ A/C}$ was evaluated under typical cruise conditions ($M_\infty = 0.85$) for a range of incidence from $AoA_{A/C} = 0^\circ$ to 4° . The

results were compared to the wide range of results from the 4th DPW as reported by Tinoco et al.⁹⁶ (Figure 5.7). In total four meshes were generated for the clean wing aircraft with the tail (WBT0). The refinement ratio between the meshes was 1.15 in each direction. The boundary layer mesh was set to have the same node distribution and a consistent first cell height of $y^+ = 1$ for all mesh densities. As a result, meshes of 6.9×10^6 , 10.3×10^6 , 16.1×10^6 , and 24.1×10^6 cells were created and referred to as “coarse”, “medium”, “fine” and “superfine” respectively. The study was computed with the fully turbulent $k-\omega$ SST model and compared with experimental data from Rivers et al.¹⁵ (Figure 5.7). In the experiment a set of trip dots was used to fix the location of transition. The trip dots were applied on the fuselage, on the nacelle, at 10% of wing chord for both sides of the wing and at 10% of tail chord on both sides of the tail¹⁵. At the cruise $C_{L A/F} = 0.5$ (Figure 5.7 and Table 5.1), the CFD typically

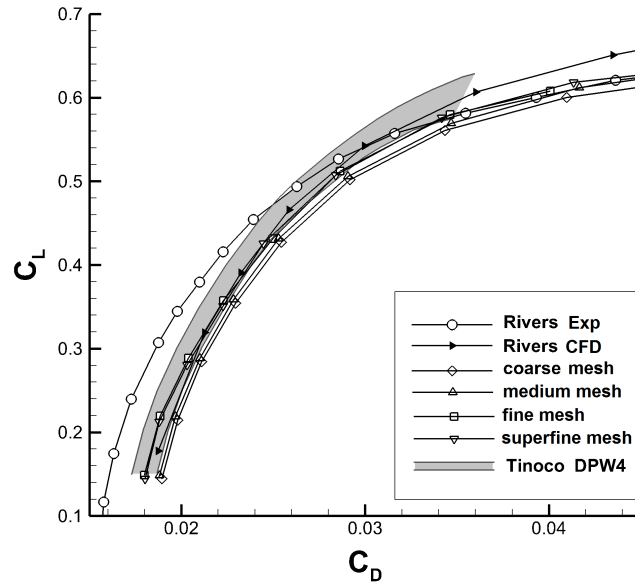


FIGURE 5.7: Computational drag polar from for clean wing CRM (WBT0) for ‘coarse’, ‘medium’ and ‘fine mesh’ compared with computational data for rigid geometry by Rivers et al and measurements¹⁵ using $k-\omega$ SST and S-A models; band of structured-mesh solutions from 4th Drag Prediction Workshop marked in grey⁹⁶; all data at $Re = 5 \times 10^6$ and $M_\infty = 0.85$; all current CFD with $k-\omega$ SST

over-estimated the $C_{D A/F}$. At this key condition the $\Delta C_{D A/F}$ between the computations and measurements monotonically reduced from +23dc to +20dc to +13dc for the “coarse”, “medium” and “fine” meshes, respectively. A further increase of mesh density to “superfine” resulted in no improvement in the drag accuracy with $\Delta C_{D A/F} = +13dc$ at $C_{L A/F} = 0.5$. At $C_{L A/F}$ from 0.2 to 0.4, the $\Delta C_{D A/F}$ increased to +29dc, 27dc, +21dc and 22dc for the “coarse”, “medium”, “fine” and “superfine”, respectively.

To quantify the grid convergence, the method advocated by Roache et al.¹⁰⁸ was followed (Appendix C.2) for a typical cruise condition $M_\infty = 0.85$ and $AoA = 2.5^\circ$ for all four meshes. The Richardson Extrapolation¹⁰⁹ was conducted to estimate the grid independent solution. The aircraft drag coefficient $C_{D A/F}$ reduced monotonically with

	$C_{D\ tot} [dc]$				
$C_{L\ tot}$	“coarse”	“medium”	“fine”	“superfine”	Exp.
0.2	196.2	194.3	186.1	186.5	166.9
0.3	215.1	213.1	206.8	208.4	185.9
0.4	245.1	242.3	238.4	237.1	217.1
0.5	290.8	287.6	280.1	280.4	267.2

TABLE 5.1: Aircraft total drag ($C_{D\ tot}$) at chosen total lift coefficients ($C_{L\ tot}$) for a range of mesh densities and compared with the experimental results¹⁵; $Re = 5 \times 10^6$; $M_\infty = 0.85$; CFD with $k - \omega\ SST$

increasing mesh size. Using a factor of safety of 1.25, the second order grid convergence index (GCI) at $AoA_{A/C} = 2.5^\circ$ was 6.54%, 5.99%, 4.06% and 3.05% for the coarse, medium, fine and superfine solutions, respectively. It was concluded that the “fine” mesh resolution is acceptable and the “superfine” mesh resolution presented too high computational cost for current standards. A full range of GCI studies was presented in Appendix C.2.4.

As well as considering the overall drag polar for the CRM, the computed wing surface pressure distributions were also compared with the measured data at wing cross-section at $y/(b/2) = 0.283$ for the aircraft design point of $C_{L\ A/F} \approx 0.5$ and $M_\infty = 0.85$ (Figure 5.8). In the experiment, the presence of the aerodynamic twist of the wing affected the pressure distributions. As the twisted geometry was not available, the comparison of CFD results with rigid geometry would not be representative. The used solver did not offer computations at a constant value of the lift coefficient and it was decided not to develop any additional user defined routines to perfectly match the $C_{L\ A/F}$. Thus, it was chosen to compare the pressure distribution at an almost constant value of lift coefficient, as a closest comparable condition with a comparable wing loading. Therefore, the experimental data for $C_{L\ A/F} = 0.494$ and corresponding $AoA_{A/C} = 2.90^\circ$ was compared with the numerical data at $C_{L\ A/F} = 0.51$ and $AoA_{A/C} = 2.50^\circ$, which was the closest comparison available. A cross-section of the wing at wingspan position of $y/(b/2) = 0.283$ was chosen, due to its greatest relevance to installation studies for through flow nacelles. Although good agreement was observed, the peak value of pressure distribution was slightly under-predicted by $\Delta C_p = -0.04$ and for both “medium” and “fine” meshes (Figure 5.8). Moreover, with the use of the “medium” mesh the computed shock location was shifted $\Delta x/c = +0.05$ downstream from its experimental location, whereas the shock location with the “fine” mesh was computed accurately at $x/c = 0.6$. The pressure coefficient at the shock location was evaluated more accurately with the “medium” mesh at $C_p = -0.65$ than with the “fine” mesh at $C_p = -0.73$ (Figure 5.8). It is worth noting that the experimental accuracy quoted by Rivers et al.¹⁵ is ± 0.0026 in terms of C_p . Overall, the CFD results showed good agreement with the key aspects of the measured data such as peak C_p and shock location. There was a

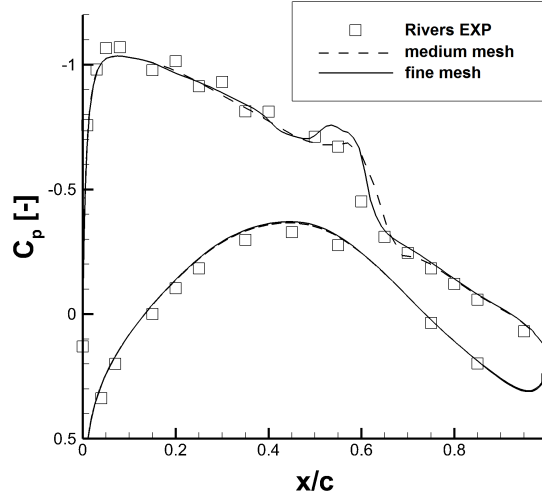


FIGURE 5.8: Pressure distribution C_p for clean wing CRM (WBT0) for two mesh resolutions ‘medium’ and ‘fine’ at $AoA_{A/C} = 2.5^\circ$; $C_{L\ A/F} = 0.51$; $y/(b/2) = 0.283$; both with structured grid and $k-\omega$ SST turbulence model; numerical results compared with experiments by Rivers et al.¹⁵ at $AoA = 2.90^\circ$; $C_{L\ A/F} = 0.494$; $y/(b/2) = 0.283$; All data at $Re = 5 \times 10^6$ and $M_\infty = 0.85$

small improvement in the agreement when a finer mesh was used, although overall the differences were modest. It can be concluded that “fine mesh” is advised.

5.2.2 Turbulence model study

A series of computations for the clean-wing CRM with the use of three different turbulence models was carried out (Figure 5.9). The models tested were the Spallart-Allmaras, $k-\omega$ SST and $k-\omega$ SST $\gamma - \theta$ transitional model. All three models were examined using the medium mesh. A notable agreement of $\Delta C_{D\ A/F} = +6dc$ was obtained for the transitional SST model at $C_{L\ A/F} = 0.5$. Moreover, a relatively low discrepancy of $\Delta C_{D\ A/F} = +13dc$ was observed between the $k-\omega$ SST $\gamma - \theta$ transition model and the measurements for a range of $C_{L\ A/F}$ from 0.1 to 0.4. The quoted discrepancy of $\Delta C_{D\ A/F} = +13dc$ was significantly better as compared with the computational results for $k-\omega$ SST and the discrepancy of $\Delta C_{D\ A/F} = +27dc$ in the same range of $C_{L\ A/F}$. However, it has to be noted that the transition in the model was not forced to match the experiment arrangement. As a result, the transitional $k-\omega$ SST $\gamma - \theta$ under-predicts drag above $C_{L\ A/F} = 0.55$. Therefore, modelling of the transition needs to be treated with caution to avoid any spurious effects due to the uncontrolled transition modelling. The Spallart-Allmaras model preformed relatively badly with $\Delta C_{D\ A/F} \approx 80dc$ and thus was not considered in further analyses.

In addition to the drag determination, the pressure distributions for $k-\omega$ SST and $k-\omega$ SST $\gamma - \theta$ were analysed to determine the impact of modelling transition (Figure 5.10). The pressure distributions for both models were in good agreement with the experimental data. Although, the peak C_p on the upper side of the wing was under-predicted by

+0.01 by both models and the shock position was moved downstream by $\Delta x/c = +0.05$ and $\Delta x/c = +0.07$ for $k-\omega$ SST and $k-\omega$ SST $\gamma - \theta$ respectively. Furthermore, the distribution of the skin friction contribution on the CRM wing for both turbulence models was analysed (Figure 5.11). A considerable changes between the models were observed along the wing leading edge with the chordwise position below $x/c < 0.2$. The use of fully turbulent $k-\omega$ SST model resulted in the peak value of $C_f = 0.0075$ at the wing leading edge and a gradual decline until the step reduction at a shock location at $x/c = 0.6$. The modelling of transition with the use of $k-\omega$ SST $\gamma - \theta$ resulted in a reduced peak value of C_f from 0.0075 to 0.006 at the wing leading edge. The peak was followed by an aggressive reduction of the skin friction coefficient to $C_f = 0.0012$ at $x/c = 0.1$ for the laminar section of the flow. At $x/c = 0.1$ the flow transition occurs and the C_f recovers a value in order of 0.0045. Downstream from the transition region, a gradual decline in the skin friction was observed to the step reduction at shock location at $x/c = 0.62$. Downstream of the shock, the prediction of the C_f was broadly the same for both models. The substantial reduction of the skin friction coefficient (C_f) at the wing leading edge explained the reduction in drag over-prediction for $k-\omega$ SST $\gamma - \theta$ model. The notable agreement of the drag results with the experiment was a consequence of the correct determination of the transition region by the turbulence model $k-\omega$ SST $\gamma - \theta$. The transition location in the computation was not specified and predicted with use of the Reynolds number based on the momentum thickness of the boundary layer.

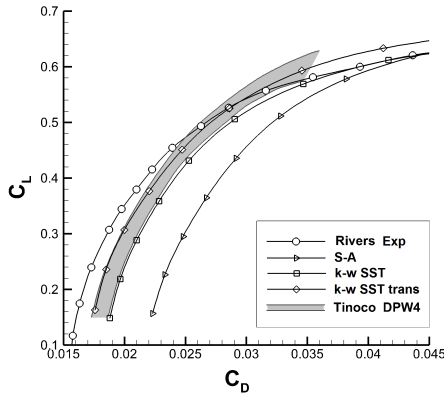


FIGURE 5.9: Computational drag polar from for rigid clean wing CRM (WBT0) for ‘medium mesh’ for different turbulence models: Spalart-Allmaras (S-A) and $k-\omega$ SST and transitional $k-\omega$ SST; all data compared with measurements¹⁵; all data at $Re = 5 \times 10^6$ and $M_\infty = 0.85$

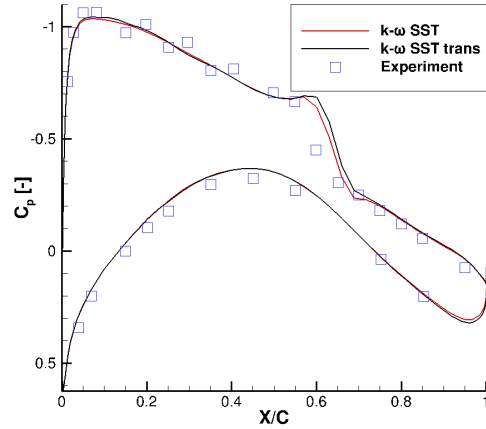


FIGURE 5.10: Pressure distribution C_p for clean wing CRM (WBT0) for “medium” mesh resolution at $AoA_{A/C} = 2.5^\circ$; $C_{L A/F} = 0.51$; $y/(b/2) = 0.283$; $k - \omega$ SST compared with $k - \omega$ SST $\gamma - \theta$ turbulence model; numerical results compared with experiments by Rivers et al.¹⁵ at $AoA = 2.91^\circ$; $C_{L A/F} = 0.52$; $y/(b/2) = 0.283$; All data at $Re = 5 \times 10^6$ and $M_\infty = 0.85$; based on del Vas¹¹⁰

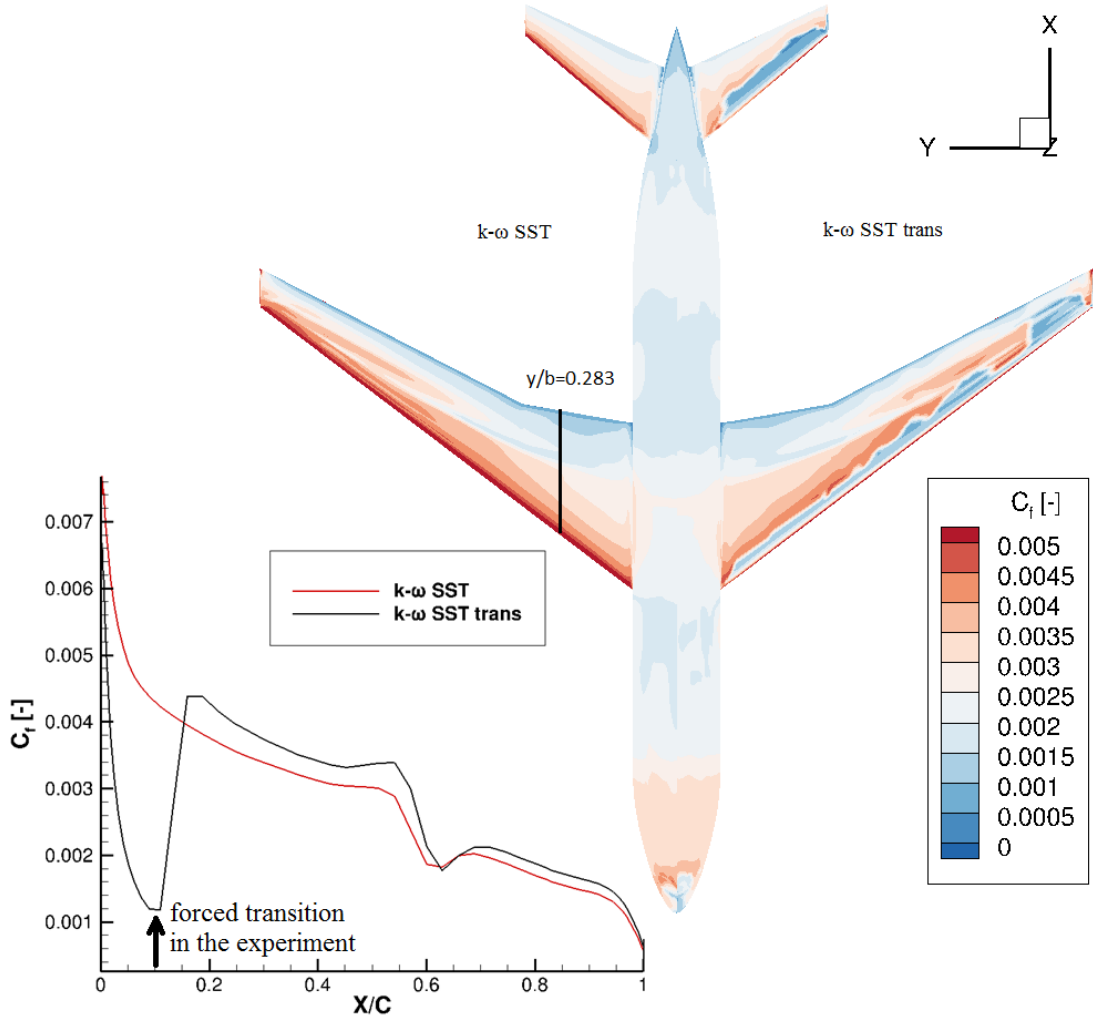


FIGURE 5.11: Contours of skin friction coefficient (C_f) for clean wing CRM (WBT0) for “medium” mesh for different turbulence models: $k-\omega$ SST and transitional $k-\omega$ SST $\gamma-\theta$; all data at $Re = 5 \times 10^6$ and $M_\infty = 0.85$; based on del Vas¹¹⁰

Overall, it was found that the modelling of transition had a negligible effect on the pressure distribution on the wing. The improvement in the drag prediction for the $k-\omega$ SST $\gamma-\theta$ model was predominantly a result of the reduced skin friction coefficient at the wing leading edge. The $k-\omega$ SST model was chosen as the baseline turbulence model for the current study, as it was consistent with the previous Drag Prediction Workshop computational activities. Moreover, the $k-\omega$ SST turbulence model was used throughout the study to avoid any spurious effects due to the uncontrolled transition modelling. It was decided not to use the fine tuning of the transition model for individual cases and this was not considered sufficiently robust. Also the good knowledge of the $k-\omega$ SST model allows for a critical assessment of the results due to known tendency to pre-mature flow separation⁶⁶.

5.2.3 Reynolds number study

A small study was carried out to investigate the impact of the Reynolds number on the accuracy of the CFD drag prediction. The drag-lift polars for three Reynolds number 5×10^6 , 19.8×10^6 and 30×10^6 were computed with the use of the “medium” mesh and compared with the experimental results¹⁵ (Figure 5.12). Based on the measurements, the increase of Reynolds number resulted in reduction of drag as expected due to the lower values of skin friction coefficient for high Reynolds number cases. Moreover, an increase of Reynolds number from 19.8×10^6 to 30×10^6 resulted in modest reduction of $-7dc$ as compared with $-20dc$ reduction from 5×10^6 to 19.8×10^6 . The computations broadly followed the trends observed in the experiment. For the lift coefficients ($C_{L\ tot}$) in range from 0.2 to 0.3, the accuracy in $C_{D\ tot}$ was $+27dc$ for the baseline case with Reynolds number 5×10^6 . The increase in Reynolds number caused an improvement in accuracy for the $Re = 19.8 \times 10^6$ to $+20dc$ but remained unaffected for $Re = 30 \times 10^6$ with approximately $+27dc$. However, at a typical cruise condition with $C_{L\ tot} = 0.5$ the accuracy for high Reynolds number improved to $+6.5dc$ and $+11.4dc$ for $Re = 19.8 \times 10^6$ and $Re = 30 \times 10^6$ respectively, as compared with the accuracy of $+20dc$ for the baseline. The summary of the agreement was presented in Table 5.2. Overall, the study suggests that CFD captured the key trend due to the increase in Reynolds number. In addition, the accuracy in drag prediction for high Reynolds numbers improved to approximately $+10dc$.

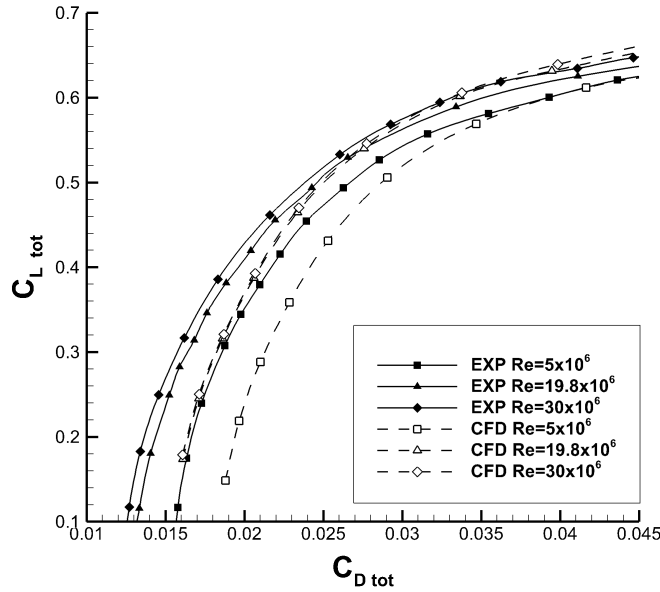


FIGURE 5.12: Computational drag polar for rigid clean wing CRM (WBT0) for ‘medium mesh’ for three Reynolds numbers 5×10^6 , 19.8×10^6 and 30×10^6 ; all data compared with measurements¹⁵; all data computed with $k-\omega$ SST at $M_\infty = 0.85$

$C_{L\ tot}$	$C_{D\ tot}\ [dc]$	$Re = 5 \times 10^6$	$Re = 19.8 \times 10^6$	$Re = 30 \times 10^6$
0.2	EXP	166.5	143.6	136.5
	CFD	194	164.4	163.6
	Δ	+27.5	+20.8	+27.1
0.3	EXP	185.8	164	157.4
	CFD	212.8	182.3	181.7
	Δ	+27	+18.3	+24.3
0.4	EXP	217	196.1	188.4
	CFD	241.4	210	209
	Δ	+24.4	+13.9	+20.6
0.5	EXP	267.1	246.8	240
	CFD	287.5	253.3	251.4
	Δ	+20.4	+6.5	+11.4

TABLE 5.2: Drag coefficient ($C_{D\ tot}$) for chosen values of lift coefficient ($C_{L\ tot}$) for the experimental¹⁵ and numerical results for the “medium” mesh cases at $M_\infty = 0.85$ and three Reynolds numbers 5×10^6 , 19.8×10^6 and 30×10^6

5.3 CRM with through-flow nacelle

The validation of numerical methods was also investigated for the CRM configuration with a through flow nacelle (TFN). Analyses for the CRM configuration with (WBNP) and without (WB) the nacelle and pylon were computed with three mesh resolutions to determine the accuracy in prediction of installation drag and to determine the sensitivity of the installation drag to the mesh density. To validate the computation for the aircraft at the condition closest to the chosen mission profile for the engine-airframe configurations, the Mach number of 0.83 was chosen from the experimental dataset¹⁵. However, to determine the sensitivity to Mach number, a computation with “medium” mesh was considered at a Mach number of 0.85. Finally, a test case with laminar turbulence model $k - \omega\ SST\ \gamma - \theta$ was considered.

5.3.1 The effect of TFN installation

The simulations for WB and WBNP configurations were compared with the experimental dataset at Mach number 0.83. The absolute agreement in the determination of the aircraft drag-lift polar ($C_{D\ tot} - C_{L\ tot}$) was determined (Figure 5.13). With the increase of mesh resolution for the WB configuration the difference between the computed and measured drag at $C_{L\ tot}=0.5$ reduced monotonically from 16.2 dc to 8.6dc for the coarse (6.9×10^6 elements) and fine mesh (14.7×10^6 elements) respectively. As for the WBNP configuration, the drag discrepancy reduced from 15.3dc for coarse mesh (15.7×10^6 elements) to 10.3dc and 8.4dc for medium (22.4×10^6 elements) and fine mesh (29.7×10^6 elements), respectively.

The placement of a TFN on the airframe had a notable effect on the airframe aerodynamics. The experimentally measured drag coefficient increased due to the presence of TFN by approximately $\Delta C_{D\ tot} = +25dc$ for lift coefficients in the range of $C_{L\ tot}$ from 0.15 to 0.4 at Mach 0.83 (Figure 5.13). The nominal cruise design lift coefficient is 0.5 and from a $C_{L\ tot}$ of 0.4 to 0.65 the increase in aircraft drag coefficient due to the installation of the TFN gradually reduced from approximately 25dc to a value of zero at $C_{L\ tot} = 0.65$ (Figure 5.13). As $C_{L\ tot}$ increased further beyond 0.65, the effect of the TFN was to reduce the overall $C_{D\ tot}$. Although there was an offset in drag levels of approximately 10dc between the computation and the experimental data (Figure 5.13), the CFD results captured these key elements of the impact of the TFN on the overall aircraft $C_{L\ tot}$ and $C_{D\ tot}$ (Figure 5.13). For example, the CFD showed a typical increase in $C_{D\ tot}$ of approximately +25dc at relatively low lift coefficients of $C_{L\ tot} = 0.15$ and the calculated point at which the change in $C_{D\ tot}$ reduced to zero is $C_{L\ tot} = 0.60$ in comparison with $C_{L\ tot} = 0.65$ for the experimental data. In addition, the CFD similarly showed that at greater values of $C_{L\ tot}$, there was an overall reduction in $C_{D\ tot}$ due to the presence of the TFN. The impact of the TFN on $C_{D\ tot}$ can be considered in more detail by evaluating the differences between the clean wing configuration (WB) and the TFN configuration (WBNP) at a constant lift coefficient (Equation 5.4 and Figure 5.14).

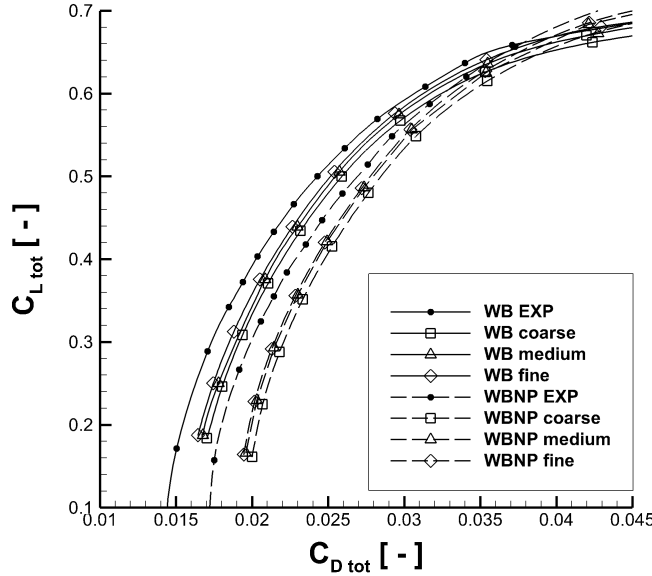


FIGURE 5.13: Lift-drag polar for CRM with (WBNP) and without a TFN (WB) for “coarse”, “medium” and “fine” meshes; computed with $k-\omega$ SST; numerical results compared with experimental by Rivers et al.¹⁵ at $Re = 5 \times 10^6$ and $M_\infty = 0.83$

$$\Delta C_{D\ inst||C_L} = [C_{D\ (WBNP)} - C_{D\ (WB)}]_{C_L=const} \quad (5.4)$$

The impact of the TFN on the overall aircraft drag showed that the magnitude of the effect of installation drag notably depended on the aircraft $C_{L\ tot}$ (Figure 5.14). At $M_\infty = 0.83$ and with the use of $k-\omega$ SST turbulence model the $\Delta C_{D\ inst}$ was relatively

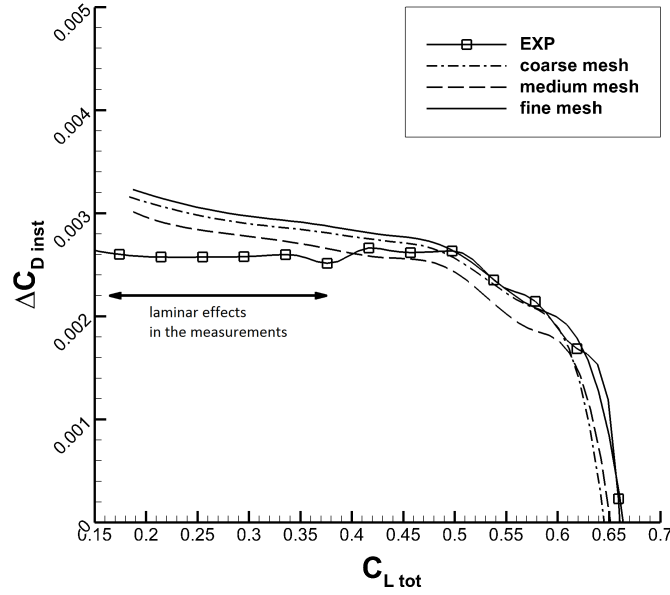


FIGURE 5.14: Installation drag coefficient ($\Delta C_{D \text{ inst}}$) for CRM WB and WBNP sub-scale as function of lift coefficient ($C_{L \text{ tot}}$); comparison between CFD coarse, medium, fine meshes and experimental data by Rivers et al.¹⁵; fully turbulent with $k - \omega$ SST; $M_\infty = 0.83$; $Re = 5 \times 10^6$

constant at 25dc up to a $C_{L \text{ tot}}$ of 0.5, beyond which it then reduces rapidly and eventually becomes a neutral change of $\Delta C_{D \text{ inst}} = 0.0$ at $C_L = 0.66$ for the “fine” mesh. The reduction of $\Delta C_{D \text{ inst}}$ was predominantly the result of an increased contribution of the TFN to the aircraft lift $C_{L \text{ tot}}$ with relatively constant drag contribution to $C_{D \text{ tot}}$ at higher aircraft incidence. The results for all the mesh resolutions were within 3dc from the experimental value of the installation drag over the range of lift coefficient from 0.4 to 0.6. The best accuracy was achieved for the fine mesh resolution with 1dc discrepancy across the range of interest ($C_{L \text{ tot}}$ from 0.4 to 0.6). The accuracy below $C_L < 0.4$ is significantly reduced to about 5dc due to the use of fully turbulent computations. It is expected that the portions of the wing for the configurations with and without the through-flow nacelle were exposed to the regions of laminar flow in the experiment. The discrepancy between the computations and the experiment was observed as the simulations were fully turbulent and by definition could not capture the subtle difference in the extent of the laminar regions over the wing with and without the through-flow nacelle. Moreover, the non-monotonic behaviour of the installation drag with respect to the mesh resolution was observed (Figure 5.14), as the “coarse” mesh presented comparable agreement with the experimental data as the “fine” mesh. The installation drag is a product of subtraction of two results for the configuration with and without the through-flow nacelle. Even though the results of the drag-polar are monotonic (Figure 5.13), the sensitivity of each configuration to the mesh refinement is different. As a result, the installation drag can be non-monotonic, due to an increased error on the computation of the total aircraft drag for the WB configuration with use of the “coarse” mesh. Overall, the use of “fine” mesh approach is advised for the engine-aircraft studies.

The 1dc discrepancy for the fine meshes is found to be an acceptable result.

Furthermore, the sensitivity of the effect of TFN installation to the turbulence model and Mach number was tested. The use of $k - \omega SST \gamma - \theta$ improved the accuracy in the drag determination for WB and WBNP configuration as compared with $k - \omega SST$. The modelling of transition resulted in a typical offset of 2dc between the computation and the measurements. However, the results of the transitional model $k - \omega SST \gamma - \theta$ offered worse agreement than a fully turbulent calculation as an opposite trend to the experimental data is observed for $C_{L\ tot} > 0.5$ (Figure 5.15). A free-transition model was used in the computations and thus the transition occurred at different x/c location of the wing for configurations with the TFN as compared to the clean-wing configuration.

The fully turbulent computation with the use of $k - \omega SST$ was completed for two Mach numbers $M_\infty = 0.83$ and $M_\infty = 0.85$ for the configurations with and without the TFN. At $M_\infty = 0.85$, the agreement between the CFD and the measurements was not as good across the low $C_{L\ tot}$ range with $\Delta C_{D\ inst} = +10dc$, although at a higher $C_{L\ tot}$ there was a notable reduction of $\Delta C_{D\ inst}$ to +1dc at approximately $C_{L\ tot}$ of 0.5 (Figure 5.16)

At $M_\infty = 0.85$, the CFD showed a reduction in $\Delta C_{D\ inst}$ with an increase in $C_{L\ tot}$, although the $C_{L\ tot}$ at which the $\Delta C_{D\ inst}$ began to reduce is lower by $\Delta C_{L\ tot} = -0.05$ than for $M_\infty = 0.83$.

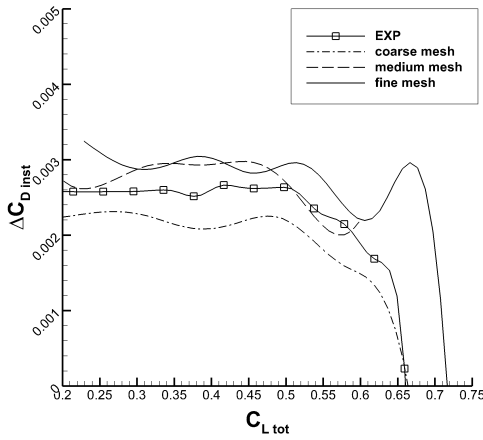


FIGURE 5.15: Installation drag coefficient ($\Delta C_{D\ inst}$) for CRM WB and WBNP subscale as function of lift coefficient ($C_{L\ tot}$); comparison between CFD coarse, medium and fine meshes and experimental data by Rivers et al.¹⁵; modelled transition with $k - \omega SST \gamma - \theta$; $M_\infty = 0.83$; $Re = 5 \times 10^6$

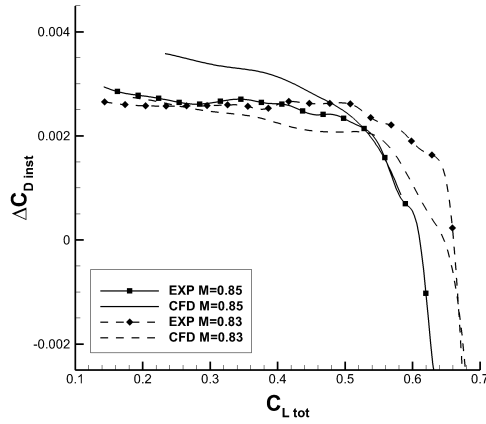


FIGURE 5.16: Installation drag coefficient ($\Delta C_{D\ inst}$) for CRM WB and WBNP subscale as function of lift coefficient ($C_{L\ tot}$); comparison between CFD medium mesh and experimental data by Rivers et al.¹⁵; $k - \omega SST$; $Re = 5 \times 10^6$

In addition to the effect on the drag, the presence of the TFN affects the lift coefficient. The effect on lift coefficient ($C_{L\ tot}$) was evaluated by considering the differences at a constant angle of attack ($AoA_{A/C}$) (Equation 5.5). The impact on $C_{L\ tot}$ as a function of AoA showed a detrimental loss of lift across the range of $AoA_{A/C}$ from 0° to about

3° (Figure 12). As the $AoA_{A/C}$ increases beyond 3° , there was a beneficial increase in $\Delta C_{L\ tot}$ to about $+0.02$. Overall, this indicated a change in lift coefficient from a loss of about 15% at $AoA_{A/C} = 0^\circ$ to an increase of lift coefficient of about 3% at AoA of 4.5° . The sensitivity of $\Delta C_{L\ tot}$ to Mach number was relatively modest. The $\Delta C_{L\ inst}$ as a function of $AoA_{A/C}$ (Figure 5.17) were generally captured well by the CFD with the characteristic constant $\Delta C_{L\ inst}$ at lower $AoA_{A/C}$ followed by an increase at an $AoA_{A/C}$ of approximately 3° for both $M_\infty = 0.83$ and $M_\infty = 0.85$.

$$\Delta C_{L\ inst||AoA} = [C_{L\ (WBNP)} - C_{L\ (WB)}]_{AoA=const} \quad (5.5)$$

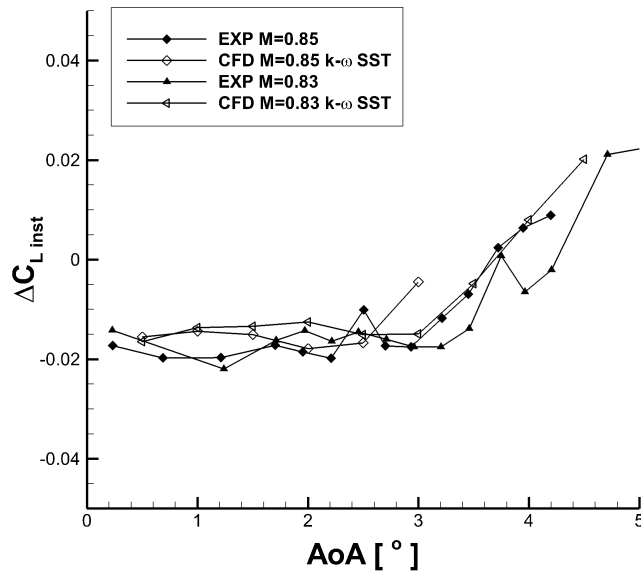


FIGURE 5.17: Installation lift coefficient ($\Delta C_{L\ inst}$) for CRM WB and WBNP subscale as function of lift coefficient; comparison between CFD medium mesh and experimental data¹⁵ for Mach number of 0.85

5.3.2 Pressure distribution

Clearly, the TFN can have a notable impact on the overall changes in $C_{D\ tot}$ and $C_{L\ tot}$. It was also of interest to consider how the TFN affects the local aerodynamics of the wing. The presence of the TFN altered the pressure coefficient (C_p) distribution mostly at measurement stations closest to the installation such as spanwise positions of $y/(b/2) = 0.201$, $y/(b/2) = 0.283$ and $y/(b/2) = 0.397$ (Figure 5.18). As a datum, it was chosen to present the C_p distributions for the computations and the measurements for the clean wing (WB) configurations at $M_\infty = 0.83$ and $C_{L\ tot} = 0.5$ (Figure 5.18). Based on the measurements, the effect of nacelle installation between the configurations with (WBNP) and without the nacelle (WB) was presented at a constant angle of attack of $AoA_{EXP} = 2.87^\circ$ (Figure 5.18). The purpose of this comparison was to indicate the loss of lift and the change in the pressure distribution due to the installation. Furthermore, the C_p distributions for the computations of WBNP configuration were considered (Figure 5.18). Ideally, the comparison between the computations and measurements for WBNP would be done at a constant lift coefficient, but the closest available data was for $C_{L\ tot\ (WBNP\ CFD)} = 0.483$ and $C_{L\ tot\ (WBNP\ EXP)} = 0.479$, respectively. Although there is a difference of $\Delta C_{L\ tot} = 0.004$, it was estimated with an linear interpolation that this equates to a difference in maximum ΔC_p of 0.0047. This was significantly smaller than the $\Delta C_p = 0.1$ due to the effect of the TFN installation.

Based on the experimental results, the inboard section of the nacelle ($y/(b/2) = 0.201$; Figure 5.18b) is subject to an increased local flow acceleration on the suction side as compared with the clean wing at the same cross-section. The increased suction starts at $x/c=0.1$ and can be quantified as an increase of $\Delta C_p = 0.1$, which increased the local peak isentropic Mach number from 1.42 to 1.46. The constant offset of $\Delta C_p = 0.1$ between the clean wing and TFN configuration held until the shock region. Further outboard (Figure 5.18c), at the cross-section closest to the engine installation ($y/(b/2) = 0.283$), the relative effects on the pressure distribution diminished. The key difference was that the position of the shock moved forward from $x/c = 0.45$ for the clean wing to $x/c = 0.35$ with TFN for the experimental data. There was an evidence of the impact of the TFN on the pressure side (Figure 5.18c), where there was a reduction of the pressure coefficient inboard from the pylon, followed by an increase of pressure coefficient after the nacelle trailing edge at $x/c = 0.15$. The change of pressure coefficient was approximately $\Delta C_p = \pm 0.1$. Further aft on the pressure side at $x/c = 0.7$ the pressure distribution for both configurations (WBT0 and WBT0NP) were broadly unaffected. Although the computations tended to under-predict the suction peak value of pressure coefficient by $\Delta C_p = -0.1$ (Figure 5.18b and Figure 5.18c), the under-prediction is consistent between the clean wing configuration and configuration with the TFN. Moreover, the location of the shock was captured correctly for the configurations with and without the nacelle. As well as for the suction side, the computation for the pressure side of the wing was in a good agreement with measurements. The CFD captured correctly the magnitude

of flow acceleration in the gully between the wing and the nacelle with maximum $\Delta C_p = +0.15$ at $x/c = 0.15$ (Figure 5.18b). Even though, in case of some cross-sections the experimental readings were not complete ($y/(b/2) = 0.397$, Figure 5.18d), it can be judged based on the agreement with the available experimental data (Figure 5.18d) that the results further outboard at $y/(b/2) = 0.397$ are found to be credible where comparisons can be made (Figure 5.18d). The presence of the installation was found to shift the shock location by $\Delta x/c = -0.1$ and causes a reduction was peak pressure coefficient of $\Delta C_p = -0.2$. The overall changes to the pressure distribution were linked to the loss of total lift for the TFN configuration in relation to clean wing. The change came from the loss of negative C_p on the suction side of the wing as well as stronger shock interaction. Overall, the CFD results indicated a good agreement with the experimental data.

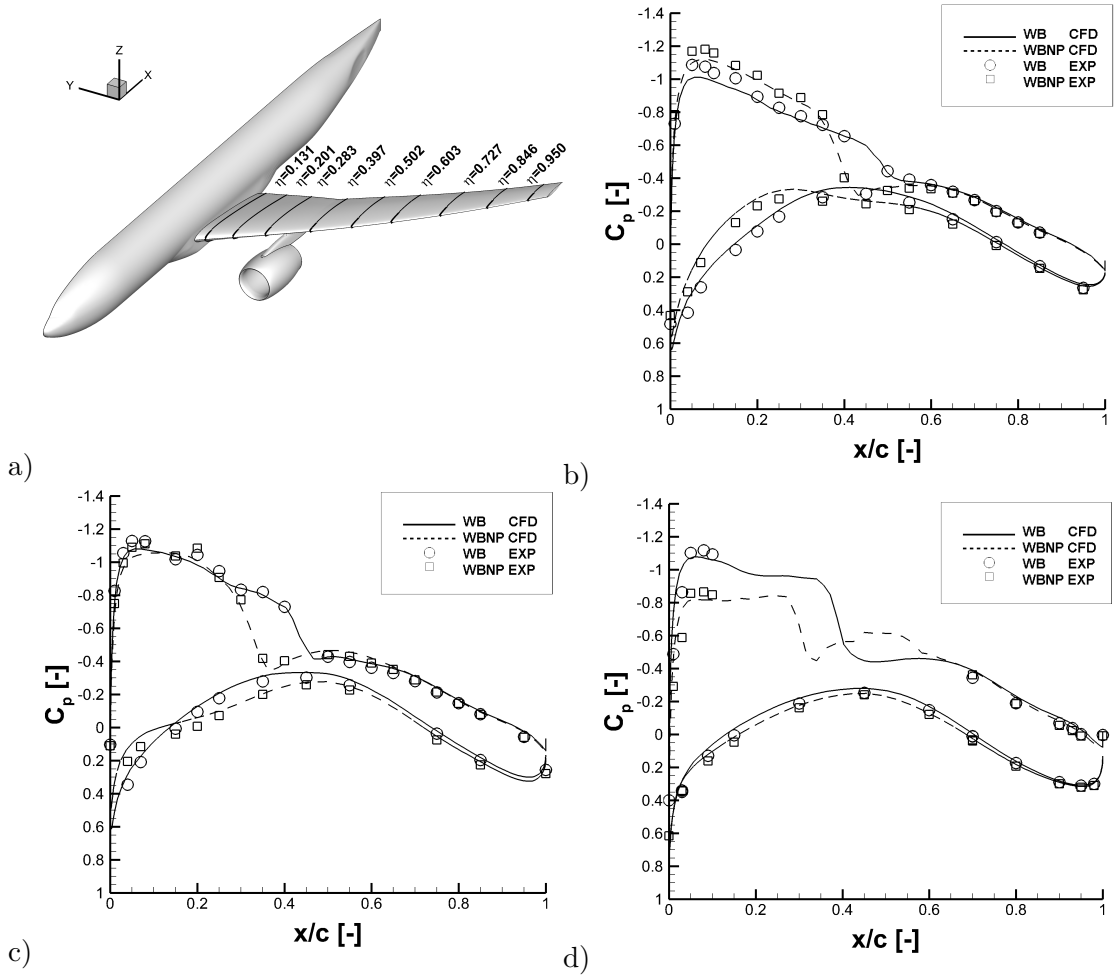


FIGURE 5.18: Comparison of pressure distribution C_p for CRM with (WBNP) and without the TFN (WB) with the experimental data measured at (a) chosen spanwise positions of the wing: b) $y/(b/2) = 0.201$ c) $y/(b/2) = 0.283$ d) $y/(b/2) = 0.397$; Computation with $k-\omega$ SST fully turbulent, “medium” mesh; clean wing (WB) $C_{L\ tot} = 0.5$ for both experimental and numerical; experimental¹⁵ comparison between WB and WBNP at constant angle of attack $AoA_{EXP} = 2.87^\circ$; numerical comparison between WB and WBNP at constant angle of attack $AoA_{EXP} = 2.5^\circ$; $Re = 5 \times 10^6$; $M_\infty = 0.83$;

5.4 Summary and conclusions

The ability of Computational Fluid Dynamics (CFD) to assess the thrust and drag of an engine-aircraft configuration was completed. The following conclusions arose from the work:

- A successful numerical strategy was developed for mesh generation and CFD computation of isolated engine, isolated airframe as well as engine-airframe configurations. The numerical strategy was a result of a range of sensitivity studies of the numerical methods to the domain size, mesh density, turbulence models
- Computations with the “fine” mesh resolution, with the use of fully turbulent $k - \omega$ *SST* model are required
- For the aircraft studies, the absolute accuracy for the total aircraft drag ($C_{D\ tot}$) was determined as 8.6dc for a WB and 8.4 dc for a WBNP configuration for the “fine” mesh at $M_\infty = 0.83$ at $C_{L\ tot}$
- The accuracy in the prediction of installation drag coefficient is 1dc (4% of the installation drag and 0.4% of total aircraft drag) over the range of aircraft lift coefficients from 0.4 to 0.6
- For the isolated engines, the accuracy in determination of thrust coefficient (C_V) is 0.01 (0.1%) over the range of fan nozzle pressure ratios (FNPR) from 1.4 to 2.8 that encompasses a full range of engine power settings considered
- Overall, the installation effects of the under-wing turbofan engines such as increase in drag, loss of lift and change in engine thrust due to the mutual interaction between an engine and an airframe can be assessed by means of Computational Fluid Dynamics

CHAPTER 6

Engine installation studies

“Realists do not fear the results of their study.”

F. Dostoevsky

This chapter concentrates on the analysis of the mutual aerodynamic interaction between the airframe and the turbofan engine. Initially, the drag decomposition for the airframe configuration with a through-flow nacelle is analysed to provide a context for the engine-airframe studies (Section 6.1). Subsequently, the design of the turbofan engines for the installation studies is presented (Section 6.2) and the dependency of the effects of engine installation on the aircraft incidence is analysed (Section 6.3). A test case with and without a pylon was analysed (Section 6.4) to assess the impact of the pylon on the overall installation effect. Furthermore, the sensitivity of the effect of engine installation to the engine position (Section 6.5), size (Section 6.6) To broaden the spectrum of considered operating conditions, a sensitivity of key installation effect to the flight Mach number was investigated (Section 6.7), and furthermore, a range of engine power setting (Section 6.8) was considered. In the overall context of the work, the computation of isolated engines and clean-wing airframe are treated as a supplementary topic required to present the effect of engine installation. Overall, the integrated fuel burn of the engine-airframe configurations is evaluated to determine the impact of different engine size over a long range mission (Section 6.9)

6.1 Drag decomposition for configuration with a through flow nacelle

Prior to the studies on the effects of engine installation, an assessment of the CRM configuration with a through-flow nacelle was considered to examine the component decomposition of the drag and its sensitivity to the angle of attack. The WBT0NP configuration was simulated with the use of the “fine” mesh at $M_\infty = 0.83$ for a range

of $AoA_{A/C}$. A drag breakdown into pressure and viscous forces was initially analysed. The interest in the specific pressure and viscous contributions is driven by the desire to develop reduced order methods for preliminary design assessments. The contribution of pressure and viscous forces to the drag was analysed as a proportion of total drag of an aircraft at each operating point (Figure 6.1). The total pressure contribution for all components varied from 40% at zero incidence up to 60% at incidence of $AoA_{A/C} = 3^\circ$. The significance of the nacelle as compared with other components was demonstrated by the contribution of a nacelle pressure drag from 20% at $AoA_{A/C} = 0^\circ$ to 5% at $AoA_{A/C} = 3^\circ$ (Figure 6.1a). This contribution stresses the fact that the detailed design of engine installation can significantly improve the overall aircraft drag. The absolute levels of the viscous contribution broadly remained constant with respect to an angle of attack for each of the individual components (Figure 6.1b). As a result, the relative viscous contribution dropped as the pressure field becomes dominant at higher angles of attack.

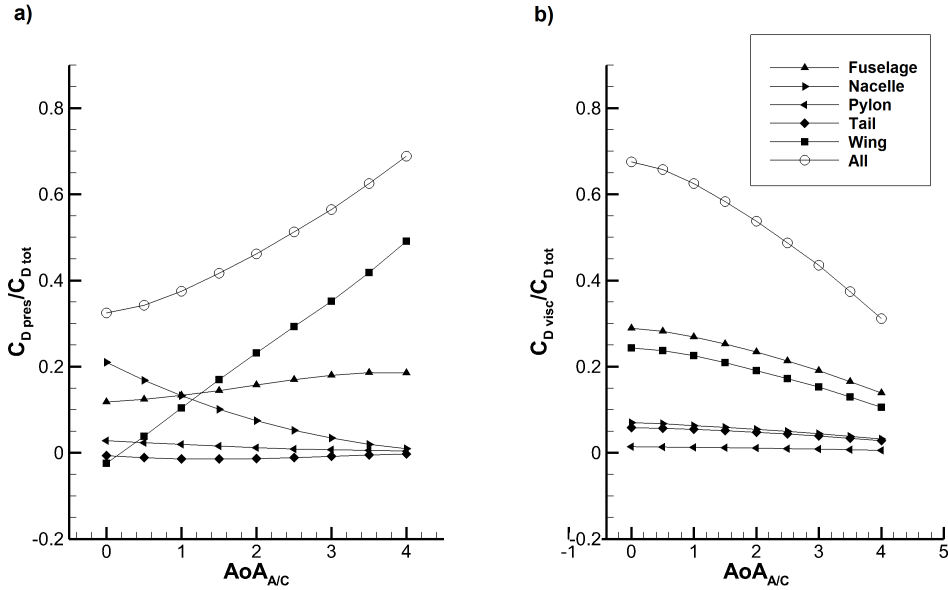


FIGURE 6.1: (a) pressure and (b) viscous contributions to total drag coefficient for CRM WBT0NP subscale as function of aircraft angle of attack; results for CFD fine mesh data for Mach number of 0.83

Furthermore, to observe the characteristics of the drag coefficient for the nacelles and pylons in greater detail a comparison is presented for selected aircraft components only (Figure 6.2). Of particular interest was the contribution of the pylon as well as the difference in the nacelle drag between an installed TFN and an isolated uninstalled TFN of the CRM. To allow a comparison with the isolated nacelle, additional 3D simulations of the isolated through-flow nacelle were carried out and the incidence of the nacelle was varied to cover the entire range of interest. The drag coefficient for two isolated nacelles was presented in aircraft drag counts to be comparable with other data computed in the presence of the airframe. Based on the WBT0NP configuration, the value of the drag

coefficient computed with the use of the standard near-field method at $AoA_{A/C} = 0^\circ$ for the nacelle on the airframe was $C_{D\ nac} = 60dc$ and for the pylon is $C_{D\ pyl} = 9dc$. As for the pylon, the contribution (Figure 6.2) was relatively insensitive to $AoA_{A/C}$ and reduced from 9dc to 5 dc for two pylons in the range of $AoA_{A/C}$ from 0° to 4° . Meanwhile, the nacelle drag component dropped down with an increase of aircraft angle of attack significantly from 60 aircraft drag counts (dc) at $AoA_{A/C} = 0^\circ$ down to 20dc at $AoA_{A/C} = 4^\circ$. This observable drop was explained by the effect of the pressure field from the underside of the wing which acts on the nacelle. As the wing loading increased with $AoA_{A/C}$, there was an increase in the pressure on the aft of the nacelle which reduced the pressure drag $C_{D\ pres}$ and counteracted the drag contribution from the rest of the nacelle and which dominated at zero incidence. To evaluate aerodynamic interference, the configuration without a pylon (WBT0N) was computed. The net result between configurations with (WBT0NP) and without the pylon (WBT0N) was computed and comprises of the pylon drag, the mutual interference drag between the pylon and the wing, and the mutual interference drag between the pylon and the nacelle. This net difference reduces monotonically from +9dc at $AoA_{A/C} = 0^\circ$ to +1dc at $AoA_{A/C} = 4^\circ$. Then the nacelle in the WBT0N configuration exhibited broadly the same trend as the nacelle in the WBT0NP configuration (Figure 6.2). This means that at $AoA_{A/C} = 0^\circ$ the drag for the nacelle in WBT0N is -6dc lower as compared with the WBT0NP configuration and the difference reduces to 0 drag counts at $AoA_{A/C} = 2.5^\circ$. Moreover, the value of $C_{D\ nac}$ at a cruise $AoA_{A/C} = 2.5^\circ$ was the same between the configurations with and without the pylon. The largest differences between the computations were observed for a range of incidence from $AoA_{A/C} = 0^\circ$ to $AoA_{A/C} = 1.5^\circ$, however the magnitude of difference was less than 6dc. It is considered to be a promising result, that the key effect on the nacelle drag is captured by the configuration without the pylon. The finding could be used to evaluate a large range of engine positions to determine the key sensitivities of the nacelle drag prior to the bespoke pylon design.

The values of drag coefficient for each component in the aircraft configurations depended on the other components present in the system. To isolate the effect of the nacelle from the effect of the pylon, the WBT0N configuration was chosen for analysis at $Re = 5 \times 10^6$ and $M_\infty = 0.83$ for a range of $AoA_{A/C}$ from 0° to 4° . The values of the drag coefficient for a component of the system changed in relation to the drag coefficient of the same component analysed in isolation (Figure 6.3a). The presence of the airframe has a mainly detrimental impact on the drag coefficient for the nacelle. Moreover, the observed trend for $C_{D\ nac}$ of a under-wing nacelle as function of $AoA_{A/C}$ was the opposite of that for the isolated nacelle and it decreased with rising incidence. The action on the nacelle was accompanied by a beneficial reaction on the airframe side. The presence of the nacelle increased local suction in the front of the pressure side of the wing. As a result, the drag coefficient of the airframe, in the presence of the nacelle, reduced for a given incidence in comparison to the clean wing computation (Figure 6.3a). Furthermore, by following the definition of the aerodynamic interference as a change of the flow due to the presence of

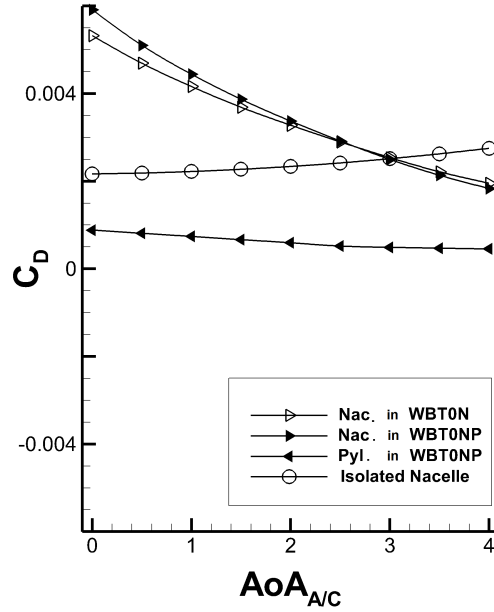


FIGURE 6.2: Drag breakdown for the configurations with (WBT0NP) and without (WBT0N) a pylon. The drag contribution for the nacelle (Nac.) and pylon (Pyl.) in chosen aircraft configurations compared against drag coefficient for isolated nacelles

other components¹¹¹, an interference drag coefficient can be defined as the change in drag coefficient due to aerodynamic interference for a component for a given aircraft angle of attack (Equation 6.1). The drag values have to be compared at appropriate angles of attack as the nacelle was installed with a pitch up angle of $AoA_{inst} = 1.5^\circ$ relative to the aircraft defined AoA . The angle of attack for the installed nacelle decomposed to Equation 6.2. For the sake of the comparison, the values of drag for an isolated nacelle were taken as the aircraft angle of attack increased by the geometrical angle of installation (AoA_{inst}), as the local wing upwash angle was determined to be negligible for the tested condition. In the tested configuration the average flow angle that entered the engine was only -0.2° from the nacelle angle of attack.

$$C_{D \text{ component interf}|AoA} = [C_{D \text{ (component in the system)}} - C_{D \text{ (component alone)}}]_{AoA_{A/C}=const} \quad (6.1)$$

$$AoA_{Nac} = AoA_{A/C} + AoA_{inst} + AoA_{upwash} \quad (6.2)$$

The values of interference drag were computed for a range of angles of attack for the nacelle and for the airframe, where the aircraft consists of wing, body and tail (Figure 6.3b). The detrimental effect of the aerodynamic interference on the nacelle was observed. The greatest penalty $C_{D \text{ nac interf}|AoA} = +32dc$ was observed at $AoA_{A/C} = 0^\circ$

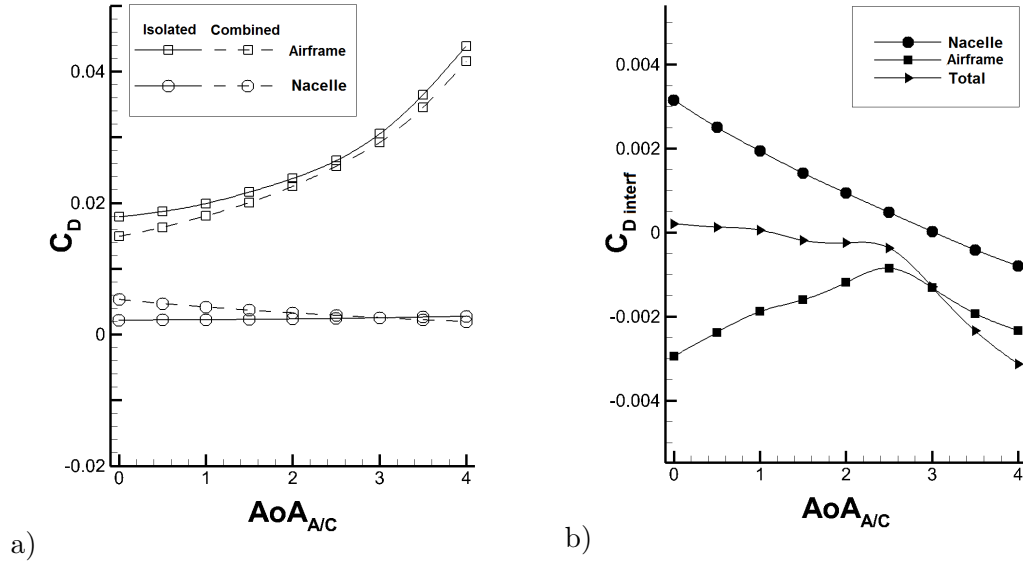


FIGURE 6.3: a) Comparison of drag coefficient as a function of aircraft $AoA_{A/C}$ for airframe (WBT0) and nacelle on their own and as component of WBT0N configuration; b) Aerodynamic interference ($C_{D\ interf|AoA}$) as a function of aircraft $AoA_{A/C}$ for airframe, nacelle and total aerodynamic interference; $Re = 5 \times 10^6$; $M_\infty = 0.83$

and the impact reduced down to zero at an $AoA_{A/C} = 3^\circ$. Although a small beneficial effect was observed at higher incidence, the magnitude of the effect was below 5dc. At $AoA_{A/C} = 0^\circ$, the presence of the wing reduced the suction on the forward facing portion of the nacelle forebody and thus increased the nacelle drag for the installed configuration. Meanwhile, the viscous drag of the installed nacelle was the same as the viscous drag on the isolated nacelle, and the nacelle MFCR had increased slightly due to installation by $\Delta MFCR = +0.01$. At higher incidences, the beneficial effect of the wing pressure field on the nacelle afterbody becomes dominant, thus the drag coefficient for the wing-installed nacelle reduces. While the nacelle was mostly negatively affected by the presence of the airframe, the airframe was subjected to beneficial aerodynamic interference. At zero incidence a benefit of $C_{D\ A/F\ interf|AoA} = -29\ dc$ was observed. The benefit decreased with incidence up to $C_{D\ A/F\ interf|AoA} = -9\ dc$ at $AoA_{A/C} = 2.5^\circ$ and increased again for greater incidence. Overall, the aerodynamic interference was the result of two competing effects such as the changes mainly in the pressure field on the nacelle and the airframe. For the current case, it was the benefit on the airframe and the penalty on the nacelle that resulted in an overall benefit $C_{D\ tot\ interf|AoA} = -4\ dc$ on the aircraft drag at $AoA_{A/C} = 2.5^\circ$. For clarity, this interference benefit ($C_{D\ tot\ interf|AoA}$) was different from the installation drag penalty ($\Delta C_{D\ inst}$), which was defined in the Section 5.3.1 and for this case was approximately +20 drag counts. The interference effects on the airframe and the nacelle are predominantly the result of the pressure field that acts on the surfaces. The effect on the nacelle was the most notable one.

6.2 Engine characteristics

To determine the effect of engine installation with respect to engine size, two relatively modern turbofan engines were considered for the mission profile and the thrust requirement of the CRM airframe ($M_\infty = 0.82$ and altitude of 10668 m [35000 ft]). The baseline engine (E1) was designed as to be representative of a typical turbofan (BPR=10, OPR=50) and the larger engine (E2) was designed to be representative of a prospectively larger engine (BPR=17, OPR=60). The engine thermodynamic cycles were designed with the use of the performance code (Section 4.2.1). While the engine by-pass ratio increased, the engine architecture, thrust requirement and component technology level were kept constant. The basic performance of the cycle can be characterised by the performance parameters (Table 6.1) such as engine overall pressure ratio (OPR), combustor outlet temperature (COT), specific fuel consumption (SFC). Moreover, the generated engine performance models were used as the main method in the determination of the engine fuel flow ($\dot{m}_{fuel}$) for a given thrust requirement. In this context, the relation of the fuel flow with regards to the engine thrust requirement was evaluated for both engines. For the same thrust requirement at the 'Start of Cruise' (100% of $F_{N\ ref}$), the fuel burn for the larger engine E2 reduced approximately by 5.8%. However, as the thrust requirement reduces, the advantage for the larger engine E2 deteriorates to 0% change at approximately 40% of $F_{N\ ref}$. Furthermore, the GEMINI design methodology (Section 4.2.2) was used to create the external engine aerolines for the two considered engines. As a result the two engine geometries were created (Figure 6.4) with the larger engine designed with a 23% larger fan diameter.

Config.	% $F_{N\ ref}$	OPR [-]	COT [K]	SFC [mg/s/N]	% $\dot{m}_{fuel}$
E1 PS1	100.0	46.3	1673	13.76	100.00
E1 PS2	78.8	39.4	1544	13.87	79.46
E1 PS3	60.0	33.5	1431	14.46	63.03
E1 PS4	42.1	27.6	1320	15.71	48.03
E2 PS1	100.0	56.1	1697	12.96	94.18
E2 PS2	71.0	45.3	1530	13.50	69.60
E2 PS3	42.8	34.9	1374	15.61	48.59
E2 PS4	15.4	23.0	1209	25.18	28.14

TABLE 6.1: Key performance parameters for E1 and E2 engines

The engine models were the main method for the generation of the engine boundary conditions to provide a consistent link between the engine intake and the engine nozzles. Based on the engine performance model at the start of cruise 'SoC', the power setting for the baseline engine (E1) (Table 6.2) is characterised by the massflow capture ratio (MFCR) for the cruise condition of 0.75, the fan nozzle pressure ratio (FNPR) of 2.71 and the core nozzle pressure ratio (CNPR) of 1.37. Similarly for the study of engine size, the larger engine (E2) 'SoC' power setting is represented by MFCR of 0.75, FNPR of 2.11

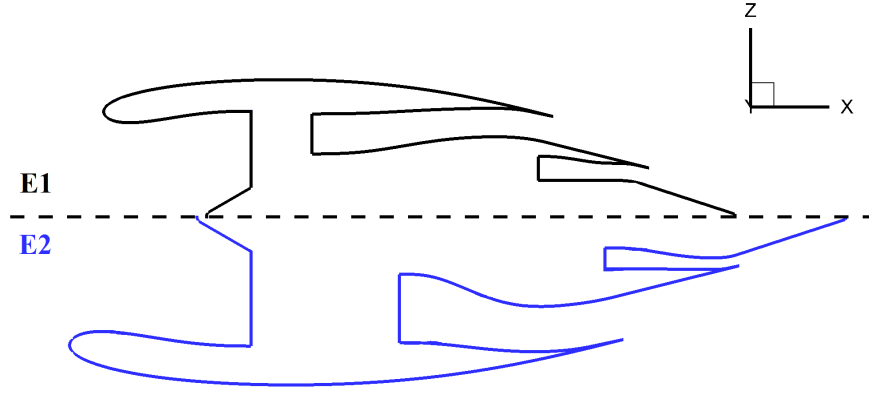


FIGURE 6.4: An example of the engine geometry created by GEMINI

and CNPR of 1.68. To quantify the sensitivity of the overall installation effect (F_{inst}) to the engine power setting, a broad range of airframe thrust requirements were considered (Table 6.2). The airframe thrust requirement was set to encompass the engine operating points from the start of cruise ('SoC') until the start of descent. Moreover, an initial check of the key aerodynamic parameters such as the intake massflow capture ratio (MFCR) and the nozzle pressure ratio for the fan (FNPR) and core (CNPR) nozzles was completed. For example, a typical intake MFCR at cruise condition¹¹² should be around 0.7. The nacelles for both engines were designed based on typical design guidelines and the computations indicated a plausible design of the nacelle with peak Mach numbers of 1.06 and 1.08 at the nacelle forebody at zero incidence, for the E1 and E2 engines respectively. The intake and nacelle performance is considered acceptable for this study, as a fully attached flow in the intake was observed and conventional pressure distributions over the nacelle were noted. The inlet pressure recovery of 0.998 was determined and a nominal fan face Mach number of 0.54 at $AoA_{eng} = 4.25^\circ$ was achieved (Figure 6.5a and b).

Engine	Power Setting	$\%F_{N \text{ ref}}$	MFCR [-]	FNPR [-]	CNPR [-]
E1	PS1	100	0.75	2.71	1.37
	PS2	78.8	0.7	2.51	1.27
	PS3	60.0	0.65	2.32	1.2
	PS4	42.1	0.6	2.12	1.14
E2	PS1	100	0.75	2.11	1.68
	PS2	71.0	0.7	1.98	1.44
	PS3	42.8	0.65	1.83	1.25
	PS4	15.4	0.6	1.7	1.11

TABLE 6.2: Engine power settings for the baseline engine (E1) and the large engine (E2)

For the baseline engine E1 (Figure 6.5a), the modified nacelle drag coefficient ($C_{D^* \text{ nac}}$) for two E1 engines was approximately 35 dc at $AoA_{eng} = 0^\circ$. The $C_{D^* \text{ nac}}$ for two

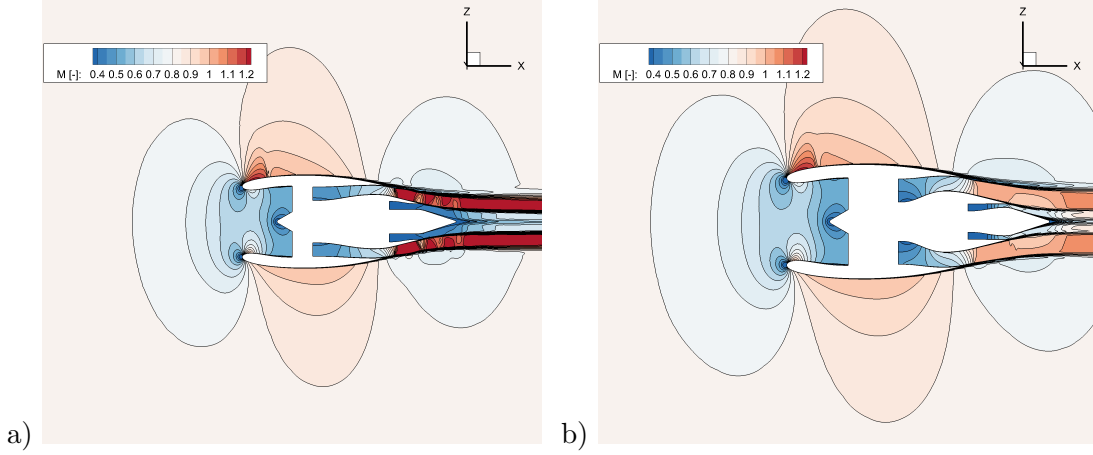


FIGURE 6.5: Contours of Mach number for a) E1 engine and b) E2 engine at $Re = 45 \times 10^6$, $M_\infty = 0.82$, $AoA_{eng} = 4.25^\circ$

E1 engines was considered to enable easier comparison with the typical effect of engine installation on the airframe of typically 30-50dc. The drag coefficient for the isolated engines ($C_{D^* nac}$) was a quadratic function with respect to engine angle of attack (AoA_{eng}) across the incidence range of incidence from 0° to 8° with 41dc at 8° . The cruise condition of an isolated engine is considered based on the mission specification of the airframe. When installed on the wing, the engine presents an effective incidence of approximately 4.25° at cruise. At this condition the pair of isolated engines has a $C_{D^* nac}$ of 36dc which was approximately 15% of the clean aircraft total drag. Compared with the baseline engine E1, the larger engine ‘E2’ (Figure 6.5b) had a $C_{D^* nac}$ of 50 dc at $AoA_{eng} = 0^\circ$ with this increase in $C_{D^* nac}$ approximately proportional to the increase in the wetted area. Moreover, other secondary effects on the drag value were present such as a modest effect of the Reynolds number and the effect of lower specific thrust on the jet entrainment drag. At a typical cruise condition $AoA_{eng} = 4.25^\circ$, the pair of isolated engines E2 has a $C_{D^* nac}$ of 52dc which is approximately 22% of the clean aircraft total drag. Across the computed range of incidence the lift coefficient ($C_{L eng}$) for the baseline engine E1 was relatively small and increased from 0 to 0.008. For the same range of incidence, the $C_{L eng}$ for the E2 engine increased from 0 to 0.012. For the effective incidence onto the engine of approximately 4.25° at the cruise condition, the lift force is negligible at approximately 0.004% and 0.006% of the lift force for the clean airframe of the cruise lift of $C_{L tot} = 0.5$, for the E1 and E2 engines respectively.

The key differences between the engines were observed in the exhaust region due to the different specific thrusts, which result in different Mach number distributions for the jet of the by-pass nozzle (Figure 6.5a and b). As a result of the relatively high FNPR of 2.71 for the E1 engine, a set of expansion and compression waves arise in the jet, which are reflected in the pressure distribution by the negative relative pressure that occurs between 20% and 60% of the core-cowl length ($L_{core cowl}$). Nevertheless, the overall force on the exhaust $\theta_{exhaust}$ is beneficial and corresponds to 11.1% of the

calculated standard net thrust. For the low specific thrust engine E2, the FNPR of 2.11 results in a different flow topology and a relatively constant distribution of positive pressure coefficient $C_p = +0.3$ that acts on the surface of the core-cowl. This provides a beneficial forward oriented force ($\theta_{exhaust}$) which corresponds to approximately 18% of $F_{N\ ref}$. For the larger engine with the lower specific thrust and reduced FNPR, these beneficial changes in $\theta_{exhaust}$ on the core-cowl make it relatively easier to achieve the same values of modified thrust (F_N^*) to compensate for the increase in modified nacelle drag (D_{nac}^*)

6.3 The effect of aircraft incidence on the engine installation

The presence of the engine on the airframe (Figure 6.6a) considerably affected the aerodynamics of the engine-airframe configuration. At a typical cruise condition with $AoA_{A/C} = 2.5^\circ$, the clean-wing (Figure 6.6b) and the isolated engine (Figure 6.6c) were compared with the engine-airframe configuration, in which the E1 engine was under the wing in position C3 (Figure 3.3) without a pylon (Figure 6.6d). As compared with the clean-wing, the engine-airframe configuration exhibited a weaker shock on the upper section of the wing due to the local reduction of incidence, and a modest acceleration on the lower side of the wing due to the presence of the nacelle and the interaction with the engine. As compared with the isolated engine, the engine under the wing was exposed to the effects from the wing pressure field on the nacelle afterbody and the engine core-cowl.

To study the effect of incidence on the effects of engine installation to the lift, drag and thrust, a range of aircraft incidence were considered from $AoA_{A/C} = 0^\circ$ to $AoA_{A/C} = 5^\circ$. The engine installation under the wing notably affected the performance of the aircraft in both the drag and thrust domains. To determine the effect in the drag domain, the total drag ($C_{D\ tot}$) and overall vertical force ($C_{Z'}$) polar is created for clean-wing configuration and engine-airframe configuration. For the engine-airframe configuration the vertical component of thrust was taken into account into the overall vertical force ($C_{Z'}$). As compared with the clean-wing configuration, the presence of the engine in position C3 caused an increase of aircraft total drag (Figure 6.7a) across almost the entire range of angles of attack from 0° to 5° . The installation drag ($\Delta C_{D\ inst}$) was extracted for the configuration at constant value of overall vertical force (Figure 6.7b). Over the range of overall vertical force coefficients ($C_{Z'}$), the drag penalty for engine E1 in position C3 increased from 23dc at $C_{Z'}=0.2$ to 32dc at $C_{Z'}=0.5$. The installation penalty ($\Delta C_{D\ inst}$) is lower than the modified drag coefficient of an isolated engines ($C_{D^*\ nac}$) at a respective overall vertical force coefficient from the 'no-interference' configuration. At $C_{Z'}=0.5$, the installation penalty in the drag domain ($\Delta C_{D\ inst}$) is -4 dc lower than the modified drag of two isolated engines ($C_{D^*\ nac}$). The reduction of the installation penalty

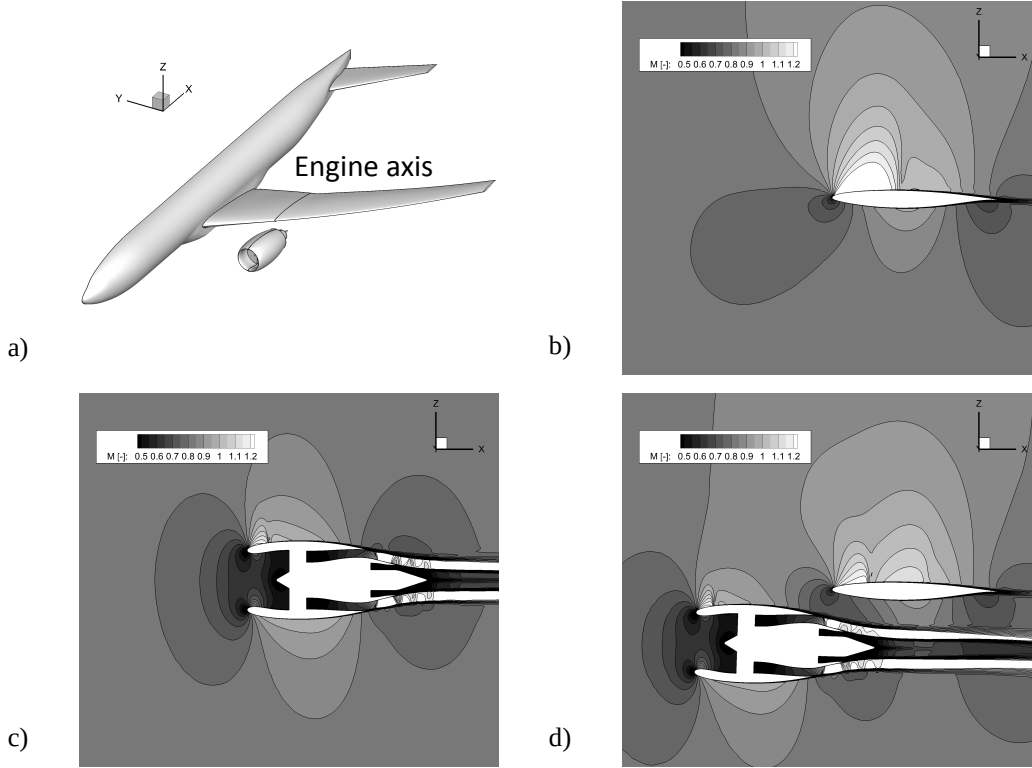


FIGURE 6.6: Contours of Mach number (M) at a) engine axis for a) CRM clean-wing configuration, b) isolated E1 engine and c) CRM with E1 engine in position C3;
 $Re = 45 \times 10^6; M_\infty = 0.82; AoA_{A/C} = 2.5^\circ$

($\Delta C_{D \text{ inst}}$) as compared with the isolated engines is a result of beneficial aerodynamic interference. The interference effect in the drag domain is present for both the airframe and the nacelle. As for the effect on the nacelle (Figure 6.7b), the installation of the engines under the wing increased the modified drag ($C_{D^* \text{ nac}}$) of the engines notably for a range of $C_{Z'}$ from 0.2 to 0.35. However, a further increase of the $C_{Z'}$ results in a reduction of $C_{D^* \text{ nac}}$ for the wing-installed nacelles. As a result, as the aircraft angle of attack increased the modified drag ($C_{D^* \text{ nac}}$) of the installed engine (D_{nac}^*) reduced from 45dc at $C_{Z'}=0.2$ to 25dc at $C_{Z'}=0.7$. At a typical cruise condition $C_{Z'}=0.5$, the modified drag of the installed engines ($C_{D^* \text{ nac}}$) is -6dc less than for the isolated nacelles and -1dc lower than the resultant installation penalty ($\Delta C_{D \text{ inst}}$).

For a given angle of attack, the configuration with an engine had a reduced overall vertical force in comparison with the clean-wing configuration (Figure 6.8a). The loss of vertical force ($\Delta C_{Z'}$) is approximately constant with aircraft incidence from $AoA_{A/C} = 0^\circ$ to 4° with a value of -0.018. To compensate for the loss of $C_{Z'}$ by -0.018, the clean wing configuration is required to increase the aircraft incidence by $\Delta AoA = +0.13^\circ$. The loss of $C_{Z'}$ is caused by the reduction of local angle of attack and mainly by a reduction in suction on the wing upper surface. However, as the aircraft incidence ($AoA_{A/C}$) increased, the maximum wing loading for the clean-wing configuration was achieved, whereas the nacelle-airframe configuration could continue increasing the incidence to

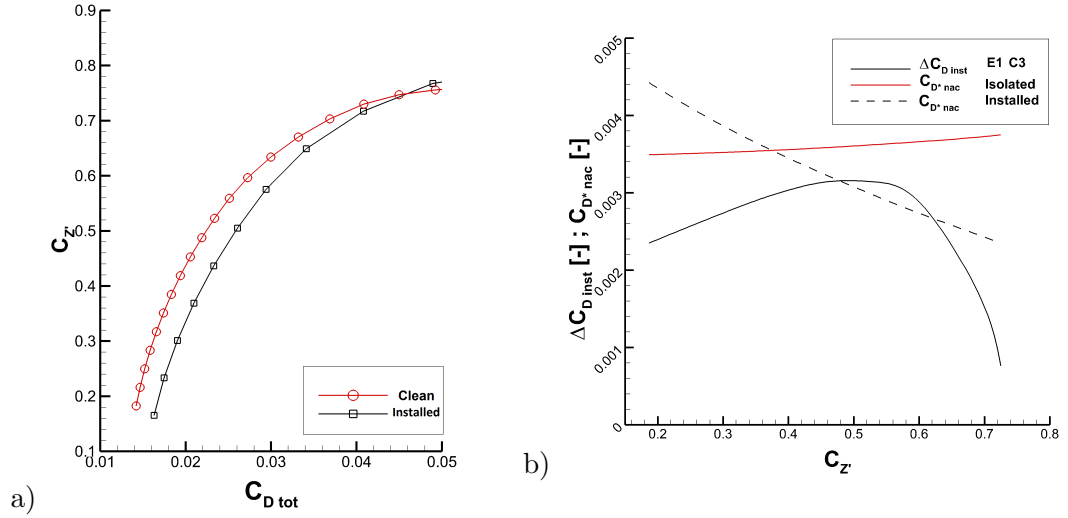


FIGURE 6.7: a) The total drag ($C_{D\ tot}$) and overall vertical force ($C_{Z'}$) polar for the CRM without (Clean) and with (E1 in pos C3) the E1 engine in position C3; b) the installation drag coefficient ($\Delta C_{D\ inst}$; solid line) compared with the modified nacelle drag ($C_{D^*\ nac}$; dashed line) of isolated and wing installed engines; $Re = 45 \times 10^6$; $M = 0.82$;

compensate for the local reduction of angle of attack due to the installation. Due to the compensation for the reduction of the local incidence, it was possible to obtain the maximum wing loading at a higher aircraft incidence ($AoA_{A/C}$) as compared with the clean-wing. At $AoA_{A/C} = 4.5^\circ$, the loss of lift equaled zero. An even greater increase of aircraft incidence resulted in a beneficial change in lift as a result of obtaining the maximum wing loading for the nacelle-airframe configuration as well as a modest contribution of the lift from the nacelle. To obtain the span-wise lift distribution (Figure 6.8b), an integration of pressure and viscous forces in individual cross-sections of the wing was done. The installation of the nacelle predominantly affects the inboard side of the nacelle ($y/(b/2) < 0.3$), where the local loss of lift coefficient was roughly -0.05 for a typical aircraft cruise incidence $AoA_{A/C} = 2.5^\circ$. Moreover, outboard of the nacelle the effect of installation on the local lift coefficient diminishes and beyond $y/(b/2) > 0.6$ it drops below -0.01 .

It was assessed that the changes to the nacelle and wing viscous forces due to incidence were broadly negligible and that the pressure field effects were the dominant terms. Furthermore, as expected, the tail and fuselage component were not affected by the installation of the engine. As a result, the changes to the lift and drag were predominantly the effect of changes in pressure distributions (Figure 6.9). On the upper side of the wing, the increase of incidence from $AoA_{A/C} = 1.0^\circ$ to 3.0° for a clean-wing configuration resulted in an expected increase of peak suction at the wing leading edge. For the $AoA_{A/C} = 1.0^\circ$, the peak suction was $C_p = -0.9$, $C_p = -1.1$ and $C_p = -1.3$ at $x/c = 0.05$ for angles of attack of 1.0° , 2.0° and 3.0° , respectively. Relatively weak shocks were observed on the upper side of the wing at $x/c = 0.2$ for the incidence of 1.0° , at $x/c = 0.35$ for the $AoA_{A/C} = 2.0^\circ$, and at $x/c = 0.5$ for the 3.0° . However

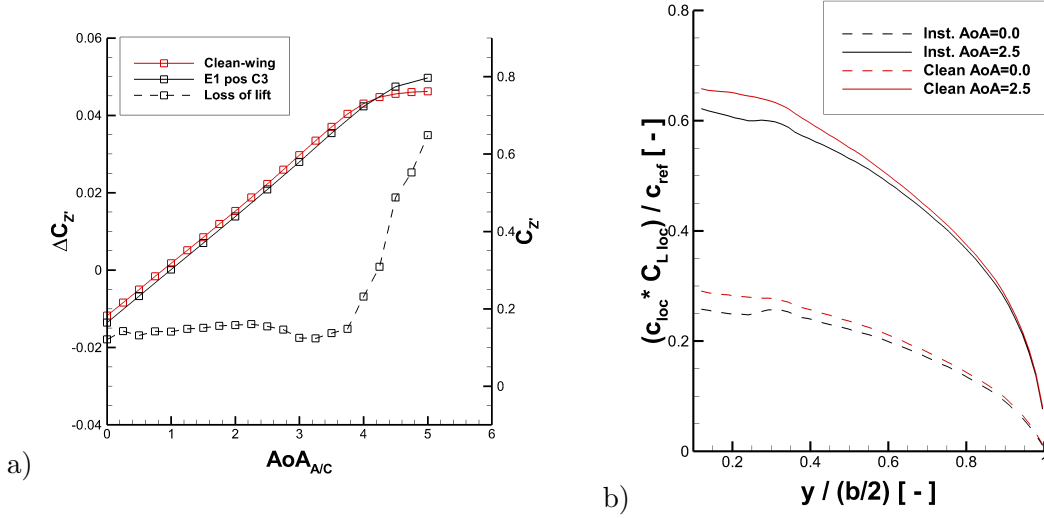


FIGURE 6.8: a) The loss of the overall lift coefficient ($\Delta C_{Z'}$) as a function of aircraft incidence ($AoA_{A/C}$), based on the values of overall vertical force coefficient ($C_{Z'}$) for the clean-wing and E1 engine configurations b) the spanwise distribution of non-dimensional lift coefficient ($C_{L_{loc}} \cdot \frac{C_{loc}}{C_{ref}}$) for CRM with (installed) and without (clean) engine at $AoA_{A/C} = 0.0^\circ$ and $AoA_{A/C} = 2.5^\circ$; $Re = 45 \times 10^6$; $M = 0.82$;

the shock for the case at $AoA_{A/C} = 2.0^\circ$ appeared particularly weak due to the locally favourable distribution of the curvature of the wing profile at the shock location. The presence of the installation reduced the peak suction at the wing leading edge by $\Delta C_p = +0.4$, $\Delta C_p = +0.35$ and $\Delta C_p = +0.3$ for angles of attack of 1.0° , 2.0° and 3.0° , respectively. The local reduction of incidence led to the disappearance of the weak shock at $AoA_{A/C} = 1.0^\circ$ due to the presence of the engine. For the incidence of $AoA_{A/C} = 2.0^\circ$ and 3.0° , a strong shock appeared to move by $\Delta x/c = -0.15$ upstream as compared with the clean-wing. On the lower side of the wing, the increase of aircraft incidence from $AoA_{A/C} = 1.0^\circ$ to 3.0° for a clean-wing configuration led to an increase of pressure coefficient at the leading edge ($x/c = 0.05$) on the lower side of the wing from $C_p = +0.4$ to $C_p = +0.6$. Due to the presence of the installation, an observable flow acceleration in the range of $\Delta C_p = -0.25$ was present close to the wing leading edge on the lower side of the wing at $x/c = 0.05$, as compared with the clean-wing over the considered range of incidence. At the maximum profile thickness on the lower side of the wing, the presence of installation caused a reduction of peak suction by approximately $\Delta C_p = +0.1$ at $x/c = 0.45$ for all the incidences considered.

The pressure distributions at the top-line of the fan-cowl of the wing-installed engine were compared with those of the isolated engine (Figure 6.9). To provide a fair comparison between these configurations, it is necessary to make the evaluations at a similar incidence. For the installed configuration the local engine incidence ($AoA_{eng,ideal}$) is taken as the sum (Equation 6.3) of the aircraft angle of attack and the engine installation angle (AoA_{inst}). For the CRM configurations $AoA_{inst} = +1.75^\circ$. Therefore at each aircraft angle of attack ($AoA_{A/C}$) the comparisons were made with the equivalent

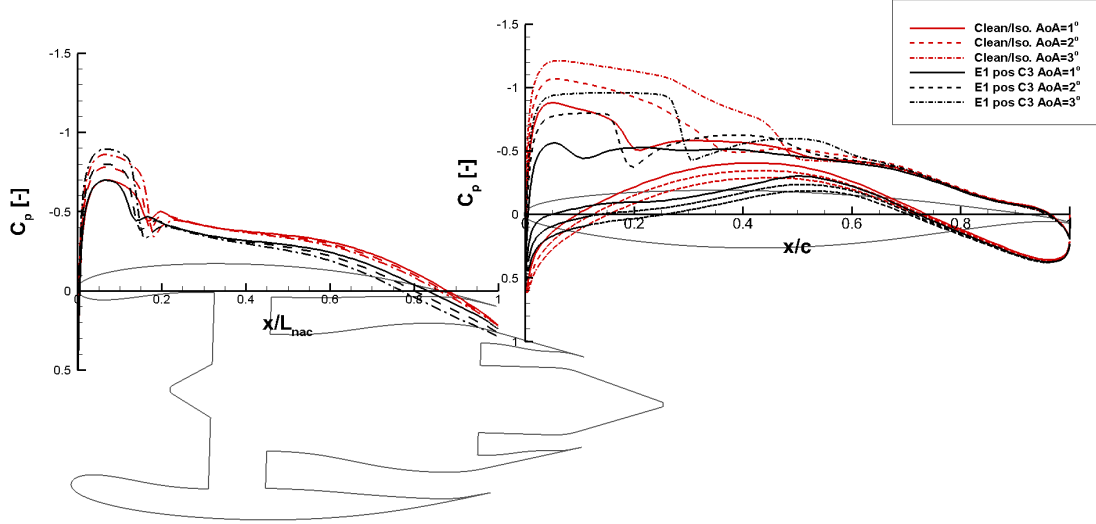


FIGURE 6.9: Pressure coefficient distribution at topline of the fan-cowl for installed engine at given aircraft angles of attack ($AoA_{A/C}$), compared with isolated engines at its angle of attack (AoA_{eng}); $Re = 45 \times 10^6$; $M = 0.82$

isolated engine configuration taking the additional AoA_{inst} into account. Moreover, in reality the engine is present in the flow-field distorted by the presence of the wing and as a result the local angle of attack deviates from the nacelle incidence (AoA_{eng} ; Equation 6.4). The deviation between local angle of attack and the aircraft global angle of attack due to the wing pressure field is the up-wash angle (AoA_{upwash}) and it determines the true engine incidence (AoA_{eng} ; Equation 6.4). For the current comparison, the up-wash angle was determined based on the clean-wing configuration at the point where the centre of the engine highlight would be in the engine-aircraft configuration. For the configuration with engine at position C3, the up-wash angle increases relatively linearly from -0.1° at $AoA_{A/C} = 0^\circ$ to roughly 0.6° at $AoA_{A/C} = 2.5^\circ$. It is determined that the effect of the up-wash is localised mostly at the forebody of the fan-cowl (Figure 6.9).

$$AoA_{eng,ideal} = AoA_{A/C} + AoA_{inst} \quad (6.3)$$

$$AoA_{eng} = AoA_{A/C} + AoA_{inst} + AoA_{upwash} \quad (6.4)$$

At a local engine incidence of approximately $AoA_{A/C} = 1.0^\circ$ the installed engine presents a neutral effect on the peak suction at the nacelle forebody as compared with the isolated engine, where the up-wash was evaluated to be $AoA_{upwash} = 0^\circ$. An increase in aircraft incidence ($AoA_{A/C}$) resulted in an increase in the up-wash effect and as a result the peak suction at the fan-cowl forebody for the wing-installed engine increased by $\Delta C_p = -0.025$ as compared with the isolated engine with $C_p = +0.8$ at $\xi/L_{nac} = 0.1$ at $AoA_{A/C} = 2.0^\circ$ (Figure 6.9). The effect of up-wash increased linearly with the aircraft incidence, and therefore a $\Delta C_p = +0.05$ was noted for the wing-installed engine at $AoA_{A/C} = 3.0^\circ$ as compared with the isolated engine. It has to be noted that the effect of the up-wash was localised in the forebody and the pressure distributions exhibit no

difference at $\xi/L_{nac} = 0.3$ (Figure 6.9) for both isolated engine with $C_p = -0.4$ and for the wing-installed with $C_p = -0.35$. Beyond the point of $\xi/L_{nac} > 0.3$ (Figure 6.9) the isolated engine was exposed to an additional pressure force generated as a result of the change of the jet direction to align with the ambient flow. The magnitude of this increase is $\Delta C_p = +0.03$ at $\xi/L_{nac} = 0.7$ (Figure 6.9) across the entire investigated range of aircraft incidence from $AoA_{A/C} = 1.0^\circ$ to $AoA_{A/C} = 3.0^\circ$ and corresponding isolated engine incidence from $AoA_{eng,ideal} = 2.75^\circ$ to $AoA_{eng,ideal} = 4.75^\circ$.

Furthermore, the increase of positive relative pressure at the aft of the fan-cowl was also observed for the wing-installed engine (Figure 6.9). However, the magnitude of the increase of this beneficial pressure force was greater than for the isolated engine and it equals $\Delta C_p = +0.03$ for every degree of increased incidence ($AoA_{A/C}$). The significant increase of the forward oriented force is explained by the pressure field perturbation and the force provides a beneficial reduction of the nacelle drag (D_{nac}^*) (Figure 6.7b). The effect of the buoyancy force on the aft of the fan-cowl, and the contribution of the suction on the nacelle leading edge sum up to a single forward oriented force (Figure 6.9).

The installation of the engine under the wing also affected the engine performance in the thrust domain as considered within the efflux streamtube. To quantify the effect, a gross propulsive force in the direction of the engine axis (GPF_ξ) (Equation (6)) was considered for both the isolated and wing-installed engine (Figure 6.10) and referenced to $GPF_{ref} = 160462N$ of an isolated engine at power setting 'SoC' at $AoA_{eng} = 0^\circ$. The gross propulsive force of an isolated engine was relatively insensitive to incidence with less than -0.1% reduction of GPF over the engine range of incidence considered. The installation of the engine under the wing resulted in a modest reduction of the gross propulsive force by approximately 0.1% at a typical cruise condition of $AoA_{A/C} = 2.5^\circ$. Moreover, the effect of engine installation on GPF_ξ was a function of aircraft angle of attack and GPF_ξ reduced by approximately 0.25% at $AoA_{A/C} = 4^\circ$ as compared with the isolated engine. This was explained by a reduction of mass flow in the core nozzle and it was manifested in the similar trend of discharge coefficient (C_d^{Core}) with incidence (Figure 6.11). The reduction in discharge coefficient was a result of the higher base pressure at the core nozzle and it was the consequence of the wing pressure-field. Meanwhile, the by-pass nozzle operated choked and there was little sensitivity of discharge coefficient to incidence and little effect to the installation (Figure 6.11). Furthermore, the modified gross propulsive force (GPF_ξ^*) was considered to take into account the forces which act on the external parts of the exhaust such as the core-cowl and the plug. Even though, the external surfaces produce friction drag, the effect of the pressure forces dominates the balance and a beneficial forward oriented force was generated that contributed to an increase in GPF_ξ^* (Figure 6.10). For the isolated engine, the GPF_ξ^* is greater than GPF_ξ by 3.8% at $AoA_{A/C} = 0^\circ$ and by 3.6% at $AoA_{A/C} = 4^\circ$. The beneficial pressure force exerted on the external surfaces of the exhaust was the result of a higher than ambient local base pressure in the jet. Furthermore, due to the installation, the local

base pressure of the engine interacted with the wing pressure field. At low aircraft incidence ($AoA_{A/C} = 0^\circ$) the engine exhaust was exposed to the local flow acceleration that occurred below the lower side of the wing. As a result, at zero incidence, relative to the isolated engine there was a reduction of GPF_ξ^*/GPF_{ref} from 1.038 to 1.021. For the wing-installed engine, the GPF_ξ^* was sensitive to $AoA_{A/C}$ and rose monotonically as a result of the increase in wing loading. The positive relative pressure-field was generated on the lower side of the wing and it acted on the engine and affected GPF_ξ^* (Figure 6.10). At the cruise point of $AoA_{A/C} = 2.5^\circ$, the installed and isolated GPF_ξ^* were almost the same while at $AoA_{A/C} = 4^\circ$ the GPF_ξ^* for the wing-installed engine had increased by 0.3% as compared with the isolated engine (Figure 6.6c and d).

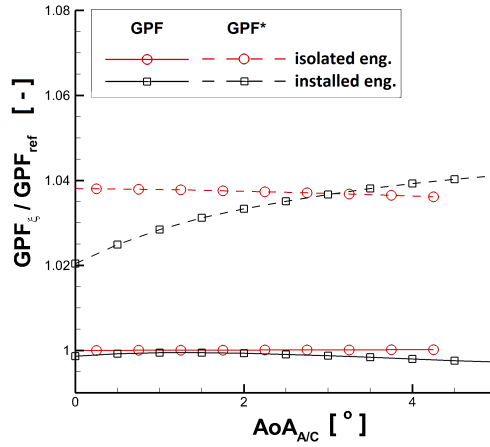


FIGURE 6.10: The overall gross propulsive force (GPF_ξ) for the Baseline engine in isolation and under the wing (installed) compared with the modified gross propulsive force (GPF_ξ^*); non-dimensionalised by GPF_ξ of an isolated engine at $AoA_{A/C} = 0^\circ$ ($GPF_{ref} = 160462N$);

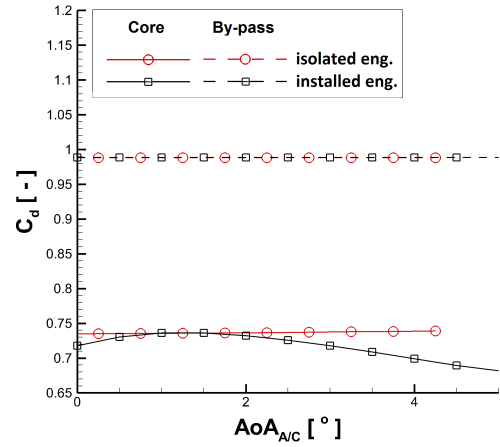


FIGURE 6.11: The discharge coefficient (C_d) for core nozzle and by-pass nozzle of the Baseline engine in isolation compared with the Baseline engine under the wing (installed); $Re = 45 \times 10^6$; $M = 0.82$;

To determine the overall impact of the installation of the engine on the airframe, an assessment of the overall installation effect (F_{inst}) of the baseline E1 engine in position C3 is considered (Figure 6.12). A re-trimmed aircraft configuration was considered at the cruise condition of $M_\infty = 0.82$ for an overall vertical force coefficient $C_{Z'} = 0.5$ that includes the vertical component of thrust. Furthermore, to determine the sensitivity of the overall installation effect (F_{inst}) to the coefficient of overall vertical force $C_{Z'}$, a range of $C_{Z'}$ from 0.2 to 0.5 was assessed. The F_{inst} was decomposed into the drag (D_{nac}^*) and thrust elements ($\theta_{exhaust}$) for the isolated engine as well as the interference terms for the engine (ΔD_{nac}^* ; ΔF_N^*) and the airframe ($\Delta D_{A/F}$). The results were referenced to the value of standard net thrust of the baseline engine E1 at cruise condition ($F_{N ref}$).

For the bare engine at $C_{Z'} = 0.5$ (left side of Figure 6.12), the benefit from $\theta_{exhaust}$ of +11% $F_{N ref}$ was opposed by the -14% $F_{N ref}$ drag penalty associated with the nacelle (D_{nac}^*). As a result, the isolated Net Propulsive Force (NPF) equaled -3.2% relative

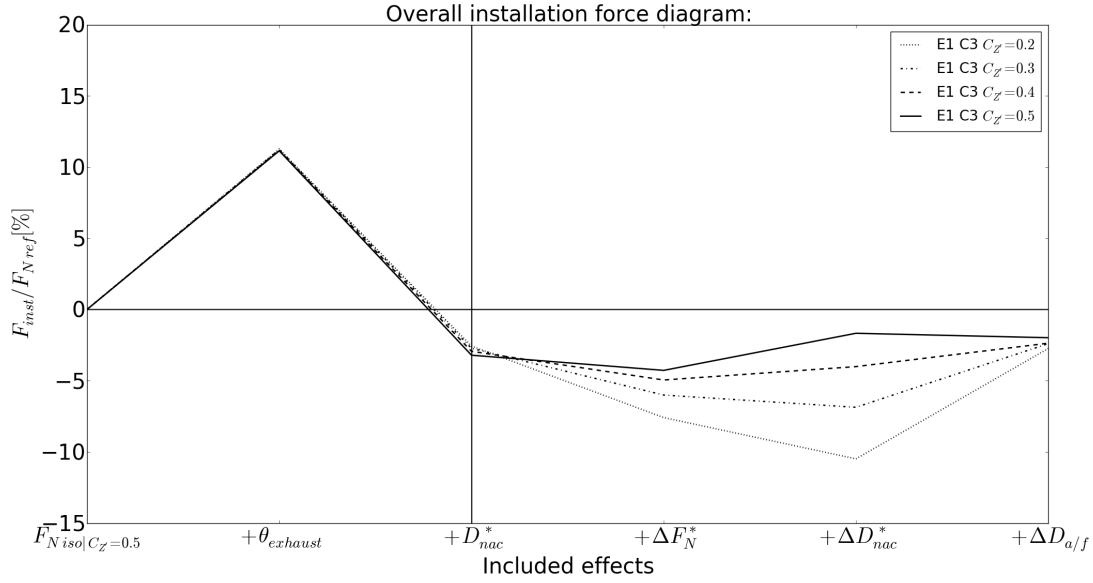


FIGURE 6.12: Overall installation force (F_{inst}) diagram for the CRM with E1 engine in position C3 at a range of overall vertical force coefficient (C_Z'); $Re = 45 \times 10^6$; $M_\infty = 0.82$

to the bare engine standard net thrust ($F_{N ref}$). With the reduction of the C_Z' , the modified drag of the nacelle decreased and the isolated net propulsive force rose to $-2.5\% F_{N ref}$ for the $C_Z' = 0.2$

To evaluate the interference terms due to the installation onto the aircraft, the interference for thrust (ΔF_N^*), modified nacelle drag (ΔD_{nac}^*) and airframe drag ($\Delta D_{A/F}$) were considered in the installation force diagram (right side of Figure 6.12). The interference forces were sensitive to the overall vertical force coefficient which ultimately affect the final resultant force (F_{inst}). The overall sensitivity of F_{inst} to the C_Z' is $0.8\% F_{N ref}$, where the most favourable interference effect was observed at $C_Z' = 0.5$. The decomposition of the overall effect on to the individual interference terms was assessed (Figure 6.13). For all installation positions the interference effect on the thrust (ΔF_N^*) was detrimental with penalty in the range from $-5\% F_{N ref}$ at $C_Z' = 0.2$ to -1.1% at $C_Z' = 0.5$

The interference effect on the airframe ($\Delta D_{A/F}$) had a different sensitivity to C_Z' . At low values of the overall lift coefficient, a beneficial effect of $+7.8\%$ and $+4.4\% F_{N ref}$ was observed at $C_Z' = 0.2$ and $C_Z' = 0.3$, respectively. An increase of aircraft incidence and C_Z' resulted in a deterioration of the observed benefit to $+1.8\% F_{N ref}$ at $C_Z' = 0.4$. Eventually, at a cruise condition ($C_Z' = 0.5$), a slightly detrimental effect of $-0.3\% F_{N ref}$ was observed. The effect on the airframe is closely coupled with the counteraction on the nacelle (ΔD_{nac}^*). The ΔD_{nac}^* is an opposing effect to the $\Delta D_{A/F}$ and at low values of overall lift (C_Z') a detrimental effect of $-3\% F_{N ref}$ at $C_Z' = 0.2$ and $-0.8\% F_{N ref}$ at $C_Z' = 0.3$ was observed that counteracts the benefits from the airframe interference. A further increase of aircraft incidence and C_Z' resulted in a beneficial buoyancy force

exerted on the fan-cowl afterbody from the wing-pressure field. The ΔD_{nac}^* became beneficial with $+1\%F_{N\ ref}$ at $C_{Z'} = 0.4$ and further increased to $+2.7\%$ at $C_{Z'} = 0.5$. The presence of the positive relative pressure field at the back of the nacelle generated the beneficial force on the nacelle, however, simultaneously is the source of the pressure drag on the lower surface of the wing.

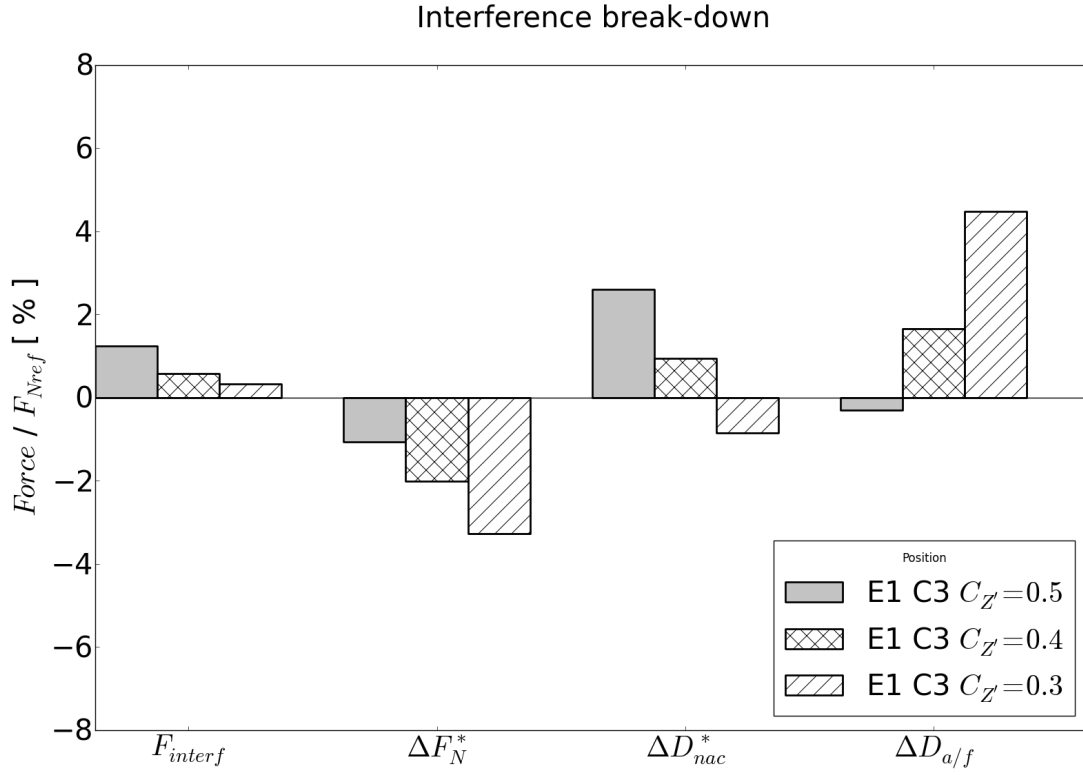


FIGURE 6.13: The breakdown of the interference force (F_{interf}) on to interference on thrust (ΔF_N^*), interference on the nacelle (ΔD_{nac}^*), and interference on the airframe ($\Delta D_{A/F}$) for the CRM with E1 engine in position C3; $Re = 45 \times 10^6$; $M_\infty = 0.82$

6.4 The influence of a engine pylon on the engine installation

The pylon is a vital element of engine installation. In general, the design of a pylon is a difficult and bespoke task and heavily dependent on engine position. In the context of a preliminary design tool, the detailed design of a pylon for every position considered in the study was not possible. Therefore, to determine the impact of a pylon on the overall aircraft performance, it was decided to compare the baseline engine E1 in a single position without (Figure 6.14a) and with a pylon (Figure 6.14b) at a cruise condition with $M_\infty = 0.82$ and $Re = 45 \times 10^6$. The case study on the pylon is necessary to

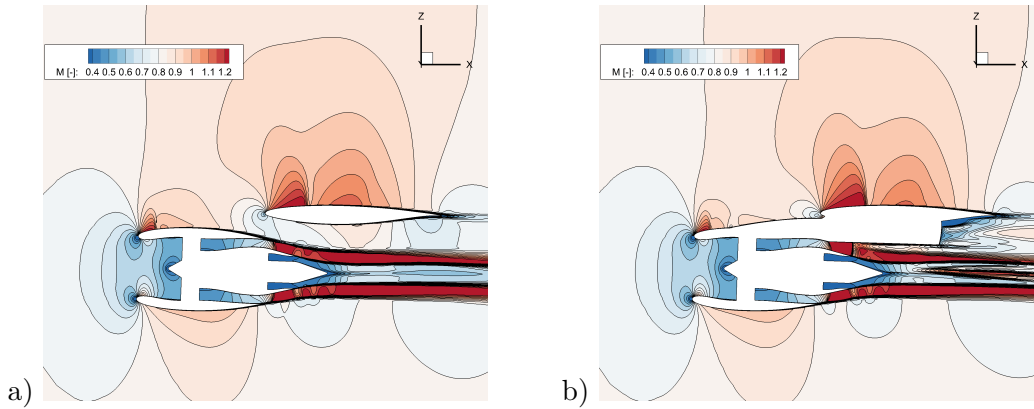


FIGURE 6.14: Contours of Mach number for the CRM with E1 engine in position D3 a) without the pylon and b) with the pylon; $Re = 45 \times 10^6$; $M = 0.82$; $AoA_{A/C} = 2.5^\circ$

assess the impact of the pylon on the component decomposition of the drag, as well as, the impact of the pylon on the breakdown of the interference effects. As a result, the baseline engine E1 was considered in the position D3, which is the same as the position of the original CRM nacelle. It was also decided to use the original CRM pylon with approximately similar shape of the gully between the nacelle and the wing. The pylon is a drag-domain pylon that extends from the wing to the trailing edge of the nacelle. The use of axisymmetric engines in the study limited the use of the pylon in the thrust domain, as the exhaust of the engines was sized without the pylon. The consideration of non-axisymmetric exhaust systems with a bifurcation in the by-pass duct remained outside of the project scope.

The presence of the pylon had a notable effect on the aircraft total drag ($C_{D\ tot}$) and at a typical cruise condition with the overall vertical force of $C_{Z'} = 0.5$ the drag increased by +5.2dc as compared with the configuration without the pylon (Figure 6.15). The result aligned with the expectation, as the effect of a pylon on the total drag of +6dc was quoted in the literature⁴² at flight Mach 0.8. Moreover, the net effect of the pylon for the E1 D3 configuration was dependent on the overall vertical force ($C_{Z'}$; Table 6.3). With a decrease in $C_{Z'}$, the net pylon effect increased from +5.2dc at $C_{Z'} = 0.5$ to 10.6dc at $C_{Z'} = 0.2$

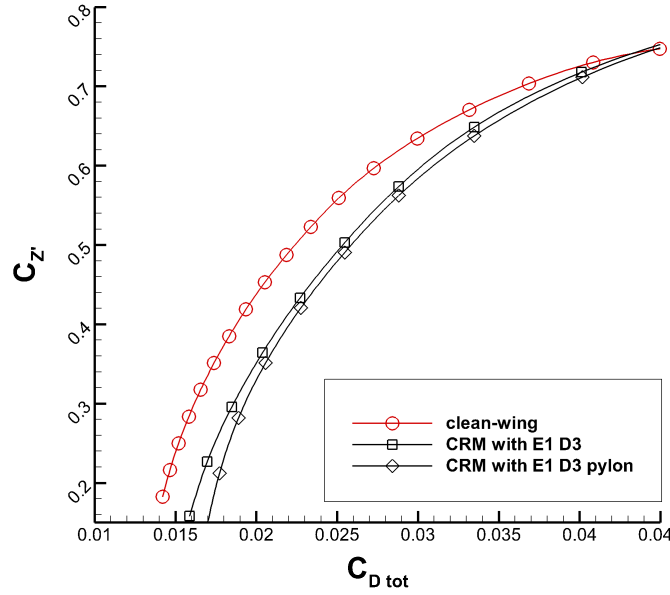


FIGURE 6.15: The overall vertical force ($C_{Z'}$) and aircraft total drag ($C_{D \text{ tot}}$) polar for the CRM without an engine, with the E1 engine in D3 position and with the E1 engine in D3 position with the pylon; $Re = 45 \times 10^6$; $M_\infty = 0.82$

	$C_{D \text{ tot}}$ [dc]		
$C_{Z'}$	CRM	CRM E1 D3	CRM E1 D3 pyl
0.2	144.3	165.0	175.6
0.3	161.7	185.6	192.9
0.4	187.7	215.4	220.6
0.5	233.9	253.5	258.7

TABLE 6.3: Aircraft total drag ($C_{D \text{ tot}}$ [dc]) at a range of overall vertical forces ($C_{Z'}$) from 0.2 to 0.5 for CRM clean-wing, CRM with E1 engine in D3 position and CRM with E1 engine in D3 position and a pylon

To determine the origin of the net effect of the pylon on the aircraft total drag ($C_{D \text{ tot}}$), the drag of single components of the aircraft such as airframe ($C_{D \text{ A/F}}$), nacelle ($C_{D^* \text{ nac}}$) and pylon ($C_{D \text{ pyl}}$) were analysed. As considered at a typical cruise incidence $AoA_{A/C} = 2.5^\circ$ ($C_{Z'} \approx 0.5$), the $C_{D \text{ A/F}}$ of 233.8dc for the clean-wing CRM (Figure 6.16a) reduced to 213.1dc for the CRM configuration with the E1 engine in position D3. The presence of the pylon reduced the airframe drag to 209.9dc. As a result, the presence of the pylon improved the beneficial interference effect on the airframe ($\Delta C_{D \text{ A/F} | AoA = \text{const}}$) from -20.7dc to -23.9dc. However, the beneficial effect on the airframe was counteracted by a detrimental effect of the pylon on the modified drag of the nacelle ($C_{D^* \text{ nac}}$; Figure 6.16b). At an incidence of $AoA_{A/C} = 2.5^\circ$, $C_{D^* \text{ nac}}$ increased from 36.2dc for the isolated engine to 41.2dc and 38.5dc for the E1 under-wing configurations with and without the pylon, respectively. The beneficial effect on the airframe and the detrimental effect on the nacelle broadly neutralised each other. Nevertheless, the pressure and viscous forces integrated over the surfaces of the pylon ($C_{D \text{ pyl}}$) contributed 3.5dc to the

net effect of the pylon on $C_{D\ tot}$. Overall, the net effect of the pylon can be explained as the result of the force integration on the pylon ($C_{D\ pyl}$), the mutual interference drag between the pylon and the wing, the mutual interference drag between the pylon and the nacelle, and the lift-induced drag, which was generated as an effect of compensation for the loss of lift on the wing.

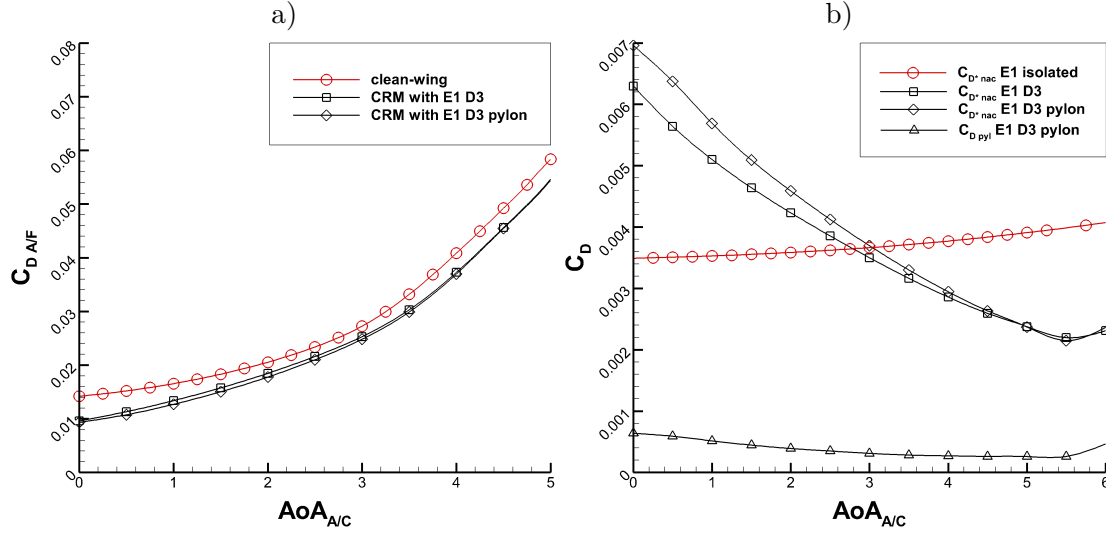


FIGURE 6.16: The a) airframe drag ($C_{D\ A/F}$) for the configuration without engine, with E1 engine in D3 position and with E1 engine in D3 position with the pylon b) the modified nacelle drag ($C_{D*_{nac}}$) and pylon drag ($C_{D\ pyl}$) for the isolated E1 engine, E1 engine under the wing in D3 position and the E1 engine under the wing in D3 position with the pylon; $Re = 45 \times 10^6$; $M = 0.82$; $AoA_{AC} = 2.5^\circ$

To determine the impact of the pylon on the overall installation effect (F_{inst}), the installation force diagram (Figure 6.17) for the baseline E1 engine in position D3 is considered for configurations with and without the pylon. A re-trimmed aircraft configuration was considered at cruise condition $M_\infty = 0.82$ for an overall vertical force coefficient $C_{Z'} = 0.5$ that includes the vertical component of thrust. The F_{inst} was decomposed into the drag (D_{nac}^*) and thrust elements ($\theta_{exhaust}$) for the isolated engine as well as the interference terms for the engine (ΔD_{nac}^* ; ΔF_N^*) and the airframe ($\Delta D_{A/F}$). For the pylon configuration, the additional penalty in terms of the drag of the pylon (D_{pyl}) was included. The results were referenced to the value of standard net thrust of the baseline engine E1 at cruise condition ($F_{N\ ref}$). For the bare engine at $C_{Z'} = 0.5$, the isolated Net Propulsive Force (NPF) equals -3.2% relative to the bare engine standard net thrust ($F_{N\ ref}$). To evaluate the interference terms due to the installation onto the aircraft, the interference for thrust (ΔF_N^*), modified nacelle drag (ΔD_{nac}^*) and airframe drag ($\Delta D_{A/F}$) were considered (Figure 6.17). The interference forces were sensitive to the overall vertical force coefficient which ultimately affect the final resultant force (F_{inst}). The overall sensitivity of F_{inst} to the presence of the pylon was -6.6% $F_{N\ ref}$. However, the breakdown of F_{inst} indicates that the -6.6% is a pessimistic evaluation. The F_{inst} was dominated by the detrimental effect of the pylon on the interference in thrust (ΔF_N^*). The detrimental $\Delta F_N^* = -2.7\%$ for the E1 engine in position D3 without

the pylon increased to -7.3% $F_{N \text{ ref}}$, which corresponds to approximately 18dc. The use of a drag pylon caused an acceleration of the flow at the exit from the by-pass duct and created a strong shock between the core-cowl and the bottom of the pylon. The presence of the suction and strong shock increased the thrust losses for the E1 under-wing configuration with the pylon. It is anticipated that the adequate non-axisymmetric design of the exhaust system with the presence of bifurcation in the by-pass duct can mitigate a significant portion of the incurred losses. As for the drag of the nacelle, ΔD_{nac}^* increased from -0.5% to $+0.1\%$ of $F_{N \text{ ref}}$ for the configurations without and with the pylon, respectively. The presence of the pylon negligibly affected the interference on the airframe ($\Delta D_{A/F}$). The inclusion of the pylon increased the $\Delta D_{A/F}$ from $+3.65\%$ to $+3.70\%$ of $F_{N \text{ ref}}$. As a result, the beneficial pylon effect in the drag domain (ΔD_{nac}^* and $\Delta D_{A/F}$) summarised to $+0.65\%$ of $F_{N \text{ ref}}$. The rather modest effect in the drag domain suggests that the general trends of the aerodynamic interference in the drag domain can be captured with computations without the pylon. In addition to the interference effects, the drag of the pylon (D_{pyl}), which was a sum of pressure and viscous forces on the surface of the pylon, was -2.8% of $F_{N \text{ ref}}$. The D_{pyl} of -2.8% of $F_{N \text{ ref}}$ corresponds to approximately 7dc and it is an expected value based on the literature (Sections 2.2 and 2.3)

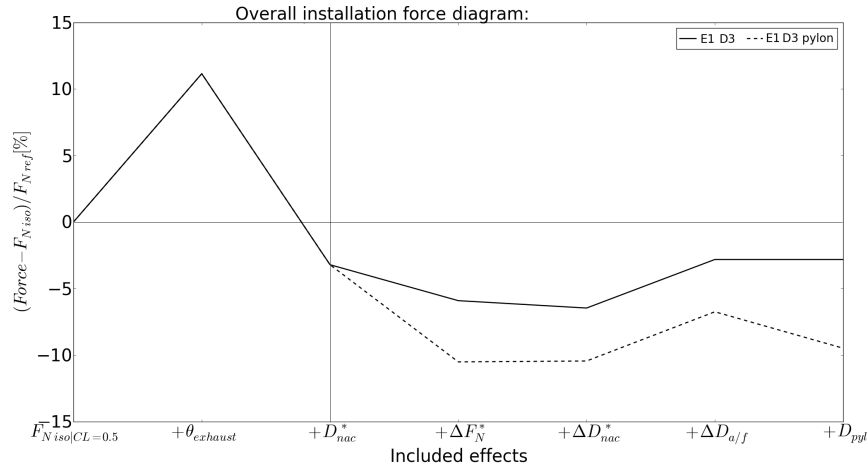


FIGURE 6.17: Overall installation force (F_{inst}) diagram for the CRM with E1 engine in position D3 at a overall vertical force coefficient ($C_{Z'} = 0.5$); $Re = 45 \times 10^6$; $M_\infty = 0.82$

Overall, the pylon had little effect on the interference in the drag domain (ΔD_{nac}^* and $\Delta D_{A/F}$) with a beneficial effect of $+0.65\%$ of $F_{N \text{ ref}}$. The overall pressure and viscous force drag of the pylon reached the values anticipated from the literature, which corresponds to -2.8% of $F_{N \text{ ref}}$. The overall sensitivity of F_{inst} to the presence of the pylon is -6.6% $F_{N \text{ ref}}$, where the significant portion of the detrimental effect was attributed to the interference on thrust (ΔF_N^*) with a -4.6% of $F_{N \text{ ref}}$ decrease when pylon was present. The detrimental effect on the thrust should be alleviated by the use of non-axisymmetric bespoke design of the exhaust system. Therefore, within the scope of the

current study it was accepted that design of the pylon for every engine position can be avoided and configurations without a pylon were used throughout the study.

6.5 The effect of engine position on the installation

To assess the impact of the engine position on the effects of engine installation, a range of engine positions was computed for the baseline engine E1 at $M = 0.82$ and $Re = 45 \times 10^6$ for a range of $AoA_{A/C}$ from 0° to 5° . Based on the flowfield at $AoA_{A/C} = 2.5^\circ$ (Figure 6.18), it was noted that the presence of the engine notably affected the shock location on the upper side of the wing; the afterbody of the fan-cowl was exposed to the wing pressure field; and the jet interacted with the lower side of the wing. For example, the engine in the close-coupled C1 position was clearly located in the positive relative pressure field from the wing (Figure 6.18). The effect on the nacelle was a buoyancy force that acted on the afterbody, whereas the wing experienced an acceleration in the gully on the lower side of the wing from $x/c = 0.05$ to $x/c = 0.2$ as compared with clean-wing (Figure 6.18). Even though the positions further away from the wing (A positions - Figure 6.18) exhibited less interaction between the nacelle and the wing compared with the C positions (Figure 6.18), there was an observable change in the flow field in relation to the isolated engine (Figure 6.18).

6.5.1 The effect in the drag domain

To determine the effect of the installation in the drag domain, the sensitivity of aircraft total drag ($C_{D\ tot}$) to the engine relative position was investigated (Figure 6.19). The difference between chosen horizontal positions was analysed (A1, B1, C1 in Figure 6.19a), as well as chosen vertical positions were compared (C1, C2, C3 in Figure 6.19b). Finally, a range of horizontal positions with and without overlap (A3, B3, C3 and the overlap position D3 in Figure 6.19c) was considered. The change of the axial position had a significant effect on the aircraft total modified drag ($C_{D\ tot}$) (Figure 6.19a). At typical cruise condition of $C_{Z'}=0.5$, the installation drag ($C_{D\ inst}$) is 41dc, 41dc and 35dc for A1, B1 and C1 respectively as compared with the clean-wing aircraft. The difference in $C_{D\ inst}$ between the configuration with different engine axial positions was approximately 6dc at cruise lift coefficient of $C_{Z'}=0.5$ and it corresponds to 2% of total aircraft drag. Moreover, the difference in $C_{D\ inst}$ across the range of axial positions at a lift coefficient of $C_{Z'}=0.2$ was 15 dc. Meanwhile, the sensitivity to engine vertical position (Column C; Figure 6.19b)) was tested and a sensitivity of 6dc was obtained at typical cruise condition of $C_{Z'}=0.5$. The installation of the engine in positions from C1 to C3 (Figure 6.19b) resulted in the installation drag of 35dc, 37dc and 31dc for C1, C2 and C3 respectively. The reduction of aircraft incidence and overall vertical force coefficients to $C_{Z'}=0.2$ led to a reduction of installation drag ($C_{D\ inst}$) to approximately 24dc, 24 dc and 28dc

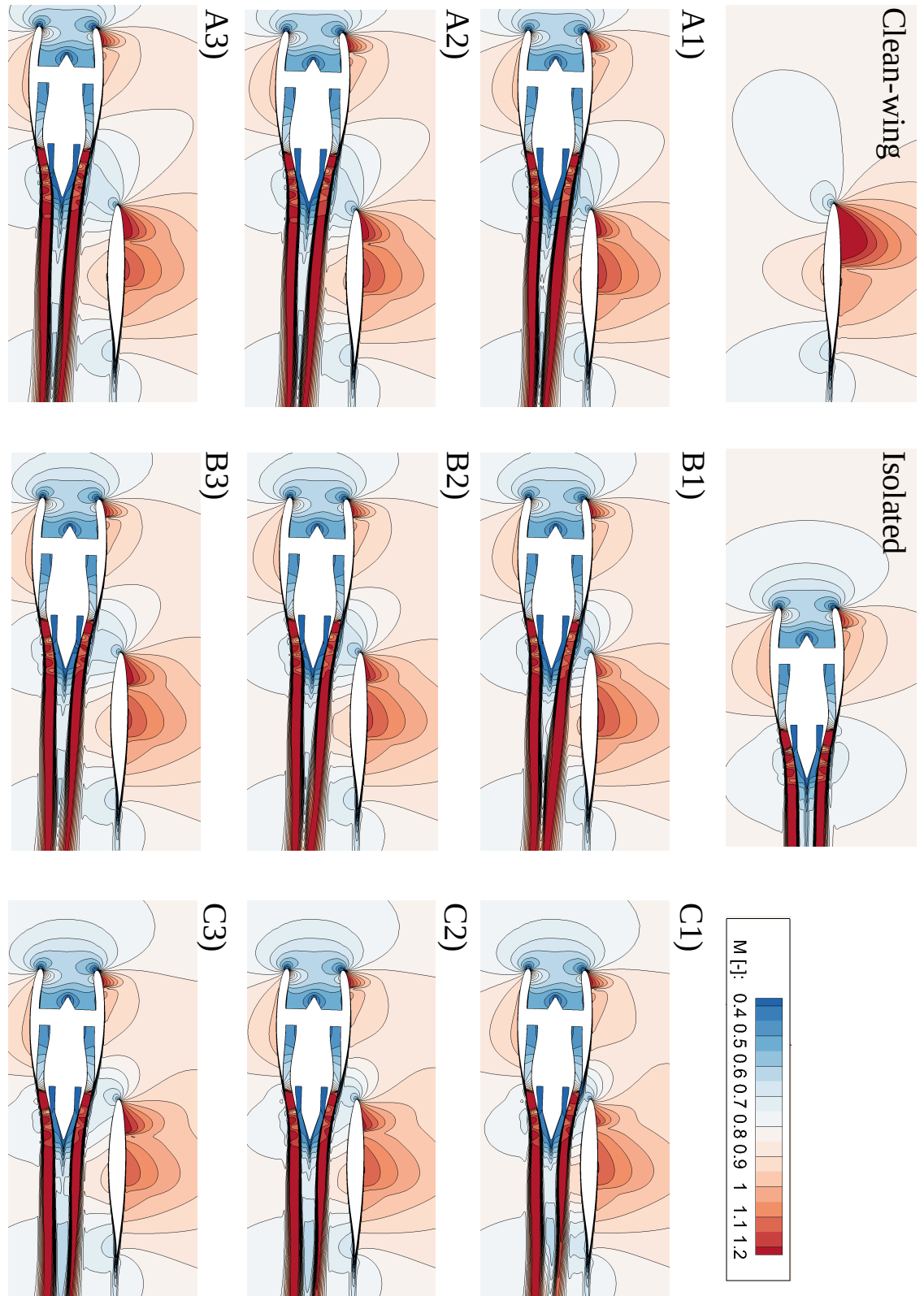


FIGURE 6.18: Contours of Mach Number at span-wise cross-section through the engine axis for the clean-wing isolated engine and a range of considered positions (A1, A2, A3, B1, B2, B3, C1, C2, C3); $Re = 45 \times 10^6$; $M = 0.82$; $AoA_{A/C} = 2.5^\circ$

(Figure 6.19b) as compared with the clean-wing aircraft. Overall, a sensitivity of the $C_{D\ inst}$ to the engine position is determined to be within 10dc, with the greatest drag penalty for the upstream shifted, vertically closely located engines.

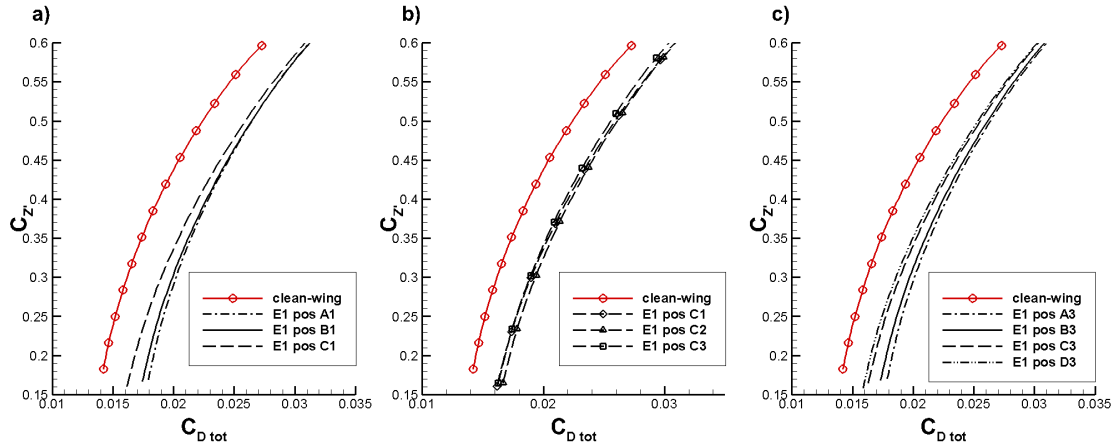


FIGURE 6.19: Overall vertical force ($C_{Z'}$) and total drag ($C_{D\ tot}$) polar for CRM without (Clean) and with engine in a) three horizontal positions A1, B1 and C1; b) three vertical positions C1, C2, C3; c); four horizontal positions A3, B3, C3, D3; $Re = 45 \times 10^6$; $M = 0.82$

Furthermore, the pressure (Figure 6.20a) and viscous (Figure 6.20b) drag coefficients for the wing and the nacelle surfaces were computed for chosen positions A1, B1 and C1. The interaction between the wing and the nacelle resulted in the reduction of pressure drag coefficient ($C_{D\ pres}$) computed at the wing for the C1 position (Figure 6.20a) and an increase of pressure drag computed at the wing for the B1 and A1 positions in relation to the clean-wing configuration. The increase in $C_{D\ pres}$ for the wing was 35dc for a movement of the engine from C1 to A1 at low incidence $AoA_{A/C} = 0^\circ$ and roughly 20 dc at a typical cruise incidence of $AoA_{A/C} = 2.5^\circ$. Modest changes to the viscous drag coefficient ($C_{D\ visc}$) (Figure 6.20b) were noted. No significant difference in the $C_{D\ visc}$ on the wing was noted between the engine-aircraft configuration and approximately +2dc constant offset as compared with the clean-wing configuration was observed over the entire range of aircraft incidence ($AoA_{A/C}$). The $C_{D\ visc}$ of the wing reduces with the aircraft incidence from about +38dc at $AoA_{A/C} = 0^\circ$ to a value of 35 dc at $AoA_{A/C} = 5^\circ$. Furthermore, a negligible difference between the isolated engine configuration and wing-installed engine was noted for the nacelle with a constant value of $C_{D\ visc} = 10dc$ over the entire range of aircraft incidence. The reduction of wing drag for close-coupled position C1 was explained as an increased suction on the pressure side of the wing near the wing leading edge.

To compare the isolated engine with the wing-installed engines, the engine idealised incidence was considered for the comparison (Equation 6.3). For the investigated range of horizontal engine positions the up-wash angle changed linearly from approximately $AoA_{upwash} \approx 0.0^\circ$ at zero aircraft incidence to $AoA_{upwash} = +0.5^\circ$ at the typical cruise condition $AoA_{A/C} = 2.5^\circ$. An up-wash angle of 0.5° corresponds to approximately

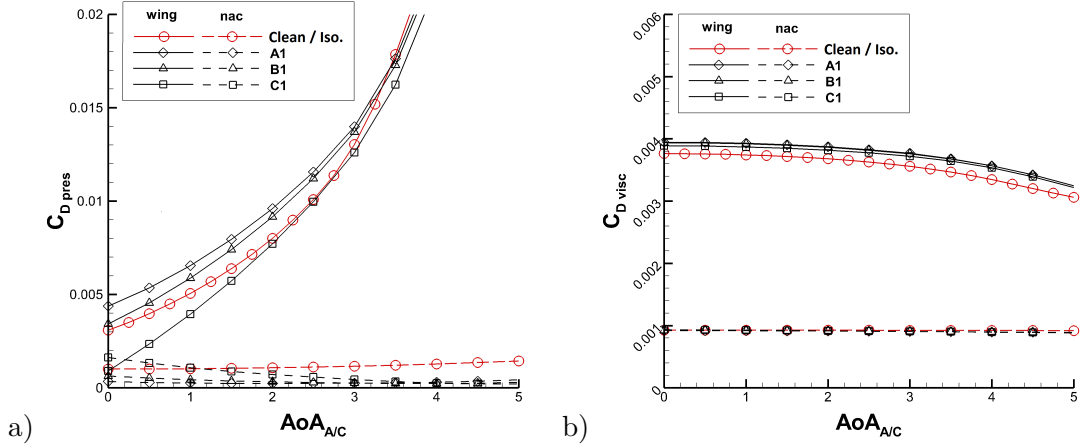


FIGURE 6.20: a) Pressure drag coefficient and b) viscous drag coefficient as a function of aircraft angle of attack ($AoA_{A/C}$) for clean-wing aircraft, isolated nacelle and three engine-aircraft configurations (A1/B1/C1); $Re = 45 \times 10^6$; $M = 0.82$; $Re = 45 \times 10^6$; $M = 0.82$;

$\Delta C_p = +0.04$ and an increase in $\Delta C_{D^* nac} = 0.2dc$. For the current comparison, the idealised engine incidence ($AoA_{eng,ideal}$ as in Equation 6.3) is used for simplicity. An idealised effective engine incidence ($AoA_{eng,ideal}$) equals $AoA_{eng,ideal} = 1.75^\circ$ for the case at $AoA_{A/C} = 0^\circ$ and $AoA_{eng,ideal} = 4.25^\circ$ at typical cruise aircraft incidence of $AoA_{A/C} = 2.5^\circ$.

Based on pressure distributions (Figure 6.21a), the suction peaks at the nacelle top-line for the wing-installed engines at $AoA_{eng,ideal} = 1.75^\circ$ ($AoA_{A/C} = 0.0^\circ$) presented the same peak suction $C_p = -0.6$ at the forebody as the isolated engine. As for the afterbody of the fan-cowl, the B1 and A1 position were affected by an increase of pressure coefficient $\Delta C_p = +0.05$ in relation to the isolated engine, and the engine in the close-coupled position C1 was affected by $\Delta C_p = +0.1$. As the incidence increased to $AoA_{eng} = 4.25^\circ$ ($AoA_{A/C} = 2.5^\circ$, Figure 6.21 b) there was an increase of peak suction of $\Delta C_p = -0.07$, which was the result of an up-wash of $AoA_{upwash} = +0.5^\circ$ for all three engine positions. Moreover, the effect of the wing on all three positions on the central section ($\Delta\xi/L_{nac} = 0.5$) of the fan-cowl was approximately $\Delta C_p = +0.05$ and there were only minor differences between the positions up to $\Delta\xi/L_{nac} = 0.7$. Only a slight movement of the shock downstream occurred as the engine was repositioned forward from the wing. In the afterbody section, the close-coupled position C1 observed an additional effect of $\Delta C_p = +0.05$ as compared with other positions A1 and B1 at $\Delta\xi/L_{nac} = 0.9$. Furthermore, due to the engine installation at $AoA_{A/C} = 0^\circ$ (Figure 6.22a), the suction on the upper side of the wing reduced by $\Delta C_p = +0.35$ for all three engine positions at $y/((b/2)) = 0.283$. The major difference between the three horizontal engine positions was observed on the wing pressure side. The positions A1 and B1 positions shifted downstream the point of maximum suction as compared with the clean-wing by $\Delta x/c = +0.1$ $\Delta x/c = +0.15$, respectively. Meanwhile, the C1 position experienced a completely different behaviour. The suction region for the C1 position had three local maxima

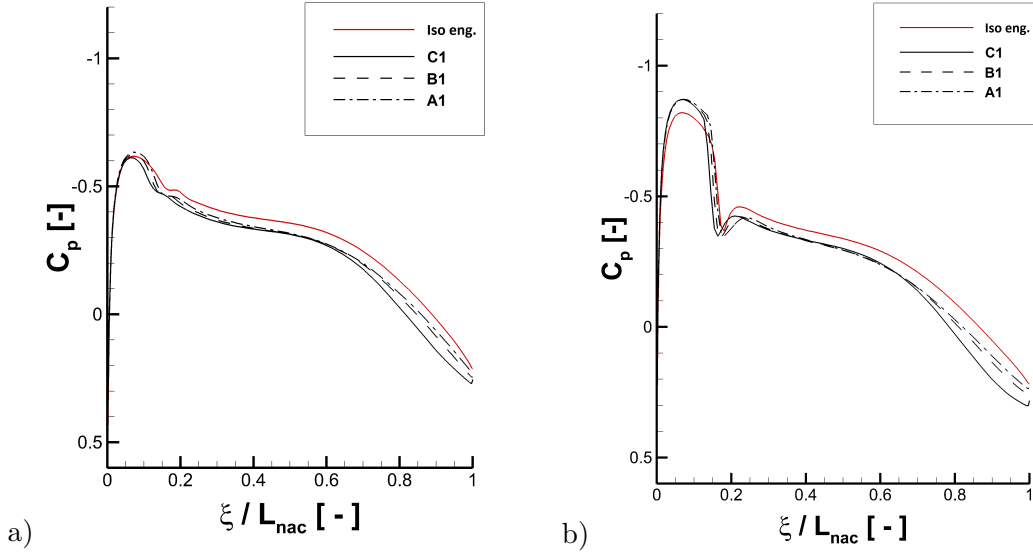


FIGURE 6.21: a) Pressure distribution at nacelle top-line ($\Theta = 0^\circ$) for engine E1 in three engine positions (A1, B1 and C1) compared with an isolated engine for idealised engine incidence $AoA_{eng,ideal} = 1.75^\circ$ ($AoA_{A/C} = 0.0^\circ$) and b) $AoA_{eng,ideal} = 4.25^\circ$ ($AoA_{A/C} = 2.5^\circ$); $Re = 45 \times 10^6$; $M = 0.82$;

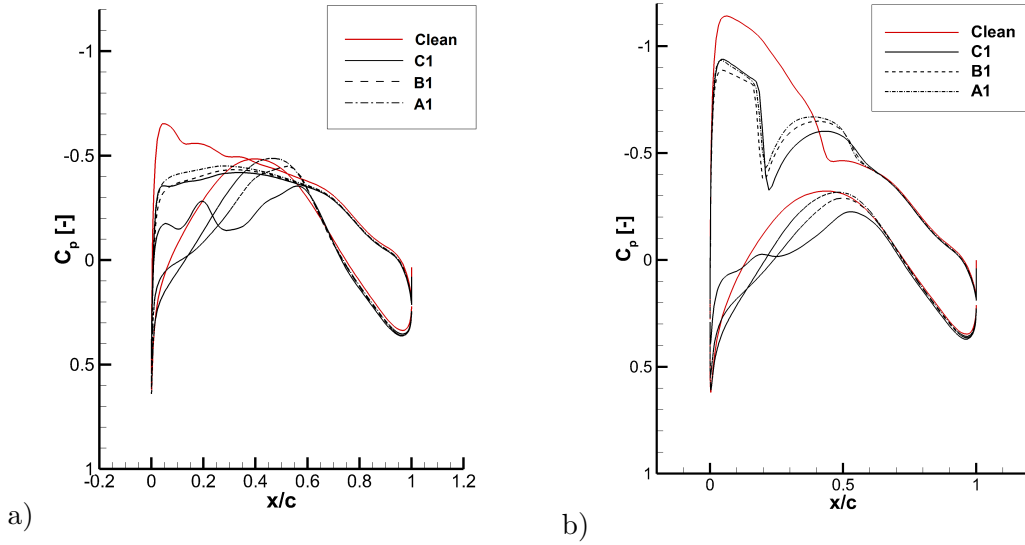


FIGURE 6.22: a) Pressure distribution at wing cross-section at engine installation plane ($y/((b/2)) = 0.283$) at $AoA_{A/C} = 0^\circ$ for engine E1 in three engine positions (A1, B1 and C1) compared with clean-wing configuration at $AoA_{A/C} = 0.0^\circ$ and b) compared at $AoA_{A/C} = 2.5^\circ$; $Re = 45 \times 10^6$; $M = 0.82$;

of suction (Figure 6.22a). The change of the flow mechanism was caused by a partial overlap of the close-coupled configuration and it explained the significant differences between the close-coupled and other engine positions. As the aircraft incidence increased to $AoA_{A/C} = 2.5^\circ$, the same mechanism for the closed-coupled position C1 was still observed, but the magnitude of the three local maxima reduced (Figure 6.22b). As for the suction side, all three positions present broadly similar pressure distributions with a reduction in peak suction by $\Delta C_p = +0.3$.

To appreciate the 3D nature of the mechanism, the lower surface of the wing (Figure 6.23a) and the surfaces of the fan-cowl, core-cowl and plug were unwrapped to be presented in 2D form (Figure 6.23b), Figure 6.23c), Figure 6.23d). Even though the close-coupled configuration (C1) (Figure 6.23b) had an increased pressure at the top of the nacelle afterbody ($\Theta = 0^\circ$) in relation to two other configurations (A1, B1) (Figure 6.23c), Figure 6.23d), the inboard (Θ from 0° to -180°) of the nacelle afterbody was subject to increased acceleration that resulted in a detrimental increase of pressure drag at zero aircraft incidence ($AoA_{A/C} = 0^\circ$).

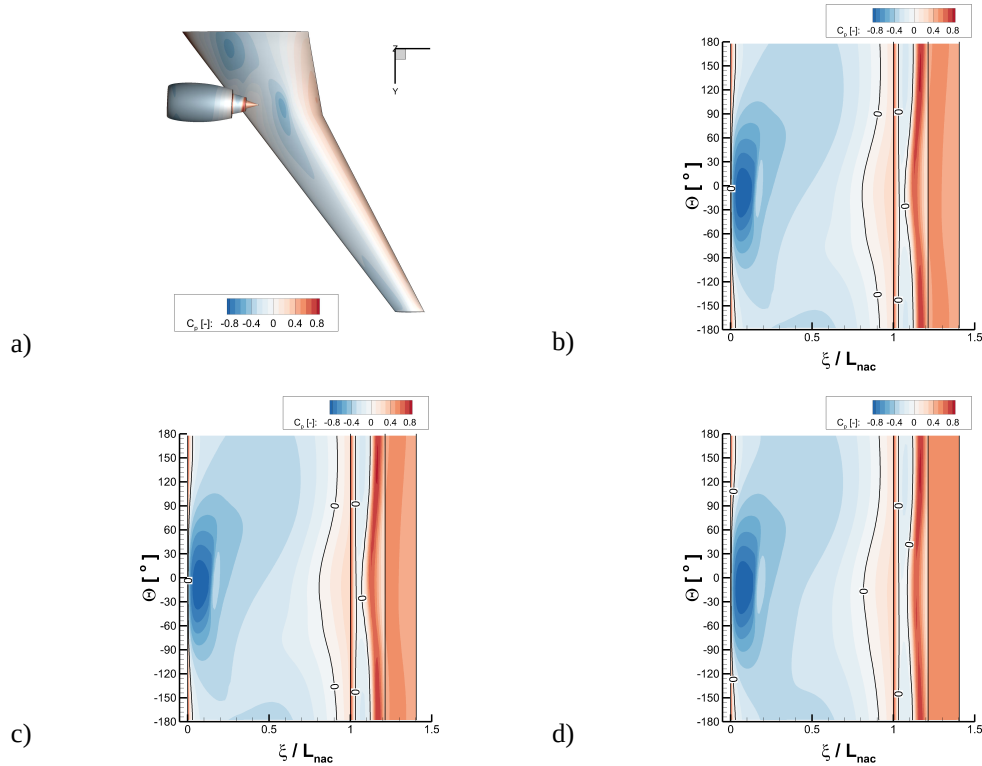


FIGURE 6.23: Contours of pressure distribution C_p on a) the wing in presence of the engine in position C1 viewed from the underside. Contours of pressure distribution on unwrapped surfaces of fan-cowl, core-cowl and plug for engines in position: b) C1, c) B1 and d) A1; $AoA_{A/C} = 2.5^\circ$; $Re = 45 \times 10^6$; $M = 0.82$; nacelle inboard for $\Theta < 0^\circ$

6.5.2 The effect in the thrust domain

To determine the impact of the engine position on the forces in the thrust domain, the effect of engine installation on the gross propulsive force in the engine axis (GPF_ξ) and the modified gross propulsive force in the engine axis (GPF_ξ^*) was assessed. The change in the base pressure at the nozzle exit reduces the gross propulsive force (GPF_ξ) of the engine (Figure 6.24a) by roughly -0.5% at typical cruise condition $AoA_{A/C} = 2.5^\circ$ in relation to the GPF of the isolated engine at $AoA_{A/C} = 0^\circ$. The effect depended on the engine position and wing loading and for the range of horizontal positions from A1 to C1 the effect was less severe for C1 position by -0.25%. Moreover, the positive relative

pressure field of the wing acted on the surfaces of the engine core-cowl and engine plug and increased the modified gross propulsive force (GPF_ξ^*) for the position A1 by 0.6% at $AoA_{A/C} = 2.5^\circ$ in relation to the isolated engine. The close-coupled position C1 exhibited a loss of GPF_ξ^* by 0.1% at $AoA_{A/C} = 2.5^\circ$ in relation to the isolated engine as a result of close coupling and flow accelerations between the engine and the wing. The greatest impact on the GPF_ξ^* of the engine in the position C1 reached -2% and was at an aircraft incidence of $AoA_{A/C} = 0^\circ$. Furthermore, the range of vertical positions was considered. The GPF_ξ and GPF_ξ^* for positions C1, C2 and C3 were calculated over the range of aircraft incidence (Figure 6.24b). As for the GPF_ξ , at a typical cruise incidence $AoA_{A/C} = 2.5^\circ$ a reduction of 0.2% is observed for all three positions in comparison with the isolated engine and it is explained by the effect of the pressure-field on the core nozzle. At the same $AoA_{A/C} = 2.5^\circ$, the GPF_ξ^* reduced as compared with isolated engine by 0.5% for all positions C1, C2 and C3. Little difference between the engine positions was observed and it was determined that GPF_ξ and GPF_ξ^* were predominantly sensitive to the horizontal position rather than vertical position.

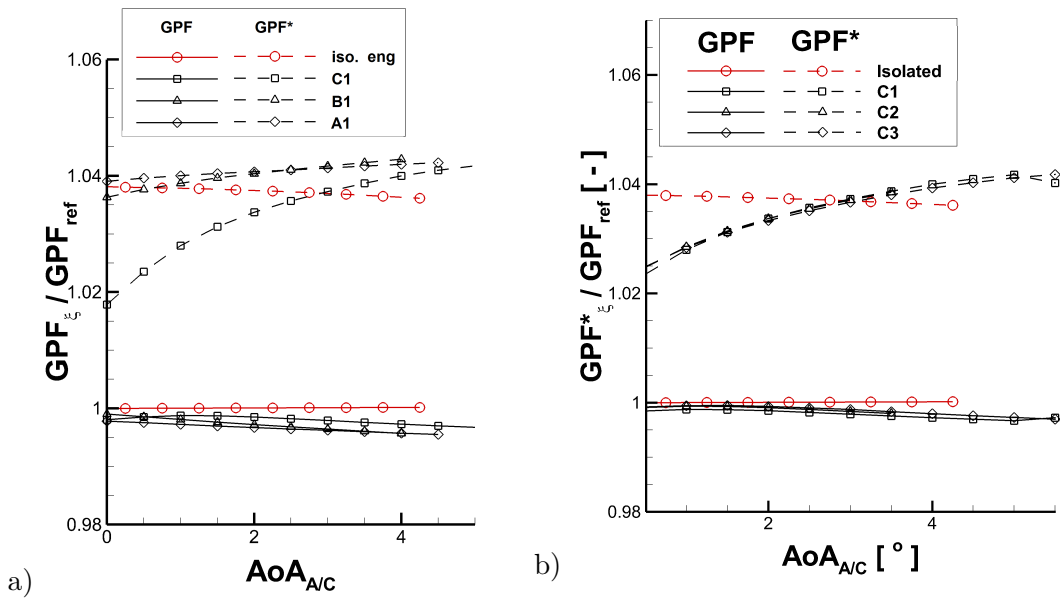


FIGURE 6.24: Specific gross propulsive force (GPF_ξ) and modified gross propulsive force (GPF_ξ^*) for isolated nacelle compared with a) three engine horizontal positions (C1, B1 and A1) and b) three engine vertical positions (C1, C2, C3); $Re = 45 \times 10^6$; $M = 0.82$

The effect on the gross propulsive force (GPF_ξ) was found to be linked with the base pressure and its effect on the discharge coefficient. The effect on the modified gross propulsive force (GPF_ξ^*) was predominantly a result of the buoyancy force from the wing pressure field that acted on the surfaces of the core-cowl and the plug. However, the interaction was not a simple superposition due to the presence of the jet. The increase of base pressure at the nozzle exit due to the wing pressure-field resulted in the reduction of the expansion at $\xi/L_{core} = 0.2$ by $\Delta C_p = +0.075$ and $\Delta C_p = +0.125$ for A1 and B1 respectively (Figure 6.25a). Similarly the strength of the compression at $\xi/L_{core} = 0.7$

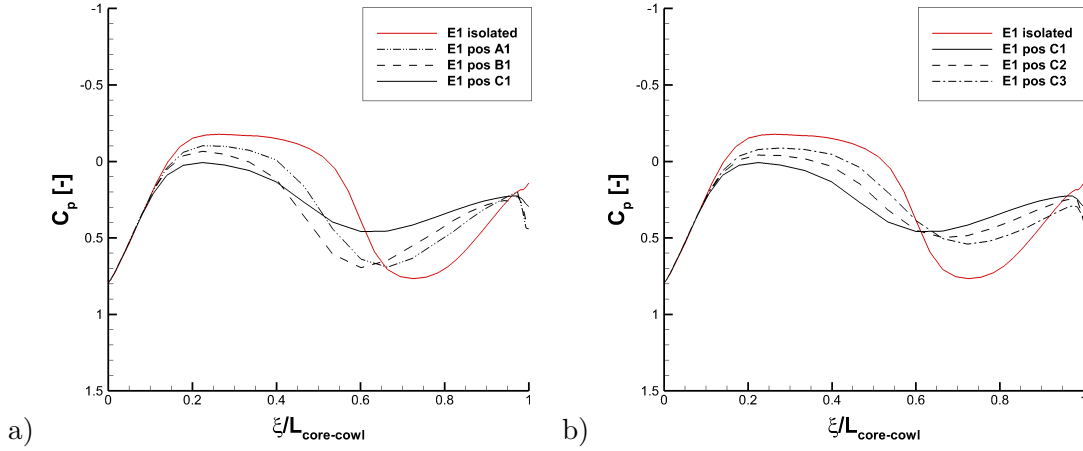


FIGURE 6.25: Pressure distribution at top-line ($\Theta = 0^\circ$) of engine core-cowl of an isolated engine compared with a) three wing installed engines in horizontal positions (C1, B1 and A1) b) three wing-installed engines in vertical positions (C1, C2, C3) at a respective engine incidence ($AoA_{eng,ideal}$); Engine installation angle $AoA_{inst} = 1.75^\circ$; $Re = 45 \times 10^6$; $M = 0.82$

was reduced by $\Delta C_p = -0.1$ and $\Delta C_p = -0.15$ for A1 and B1 respectively (Figure 6.25a). The C1 position of the engine exhibited a different characteristic, where the typical jet pattern was dominated by the pressure interaction from the wing (Figure 6.23b). The inboard section of the core-cowl between $\Theta = 0^\circ$ and $\Theta = -60^\circ$ was dominated by a negative relative pressure distribution as a result of exposure of the inboard core-cowl to a local acceleration (Figure 6.23b) on the wing. The B1 and A1 position were located further upstream and were less sensitive to the effect of partial overlap (Figure 6.23c) and Figure 6.23d). The movement of the engine downward away from the wing reduced the interaction between the wing pressure field and the jet of the engines (Figure 6.25b). As compared with the peak suction of $C_p = -0.2$ for the isolated engine, the positions C2 and C3 have a peak suction of $C_p = -0.05$ and $C_p = -0.1$ at $\xi/L_{core} = 0.3$, respectively. Meanwhile, the position C1 exhibited no suction region on the core-cowl. Furthermore, the installation of the engine minimised the compression peak at $\xi/L_{core} = 0.7$ by $\Delta C_p = -0.4$ and $\Delta C_p = -0.3$ and $\Delta C_p = -0.2$ for positions C1, C2 and C3 respectively.

6.5.3 The overall installation effect

To investigate the impact of engine position on the aerodynamic effects of engine installation, a total of nine positions were investigated with the E1 engine. Re-trimmed aircraft configurations were considered at cruise condition $M_\infty = 0.82$ for an overall vertical force coefficient $C_{Z'} = 0.5$ that includes the vertical component of thrust. An assessment of the overall installation effect (F_{inst}) of the baseline E1 engine is considered, as well as, its sensitivity to engine position (Figure 6.26). The F_{inst} was decomposed into the drag (D_{nac}^*) and thrust elements ($\theta_{exhaust}$) for the isolated engine as well as the interference terms for the engine (ΔD_{nac}^* ; ΔF_N^*) and the airframe ($\Delta D_{A/F}$). The results

were referenced to the value of standard net thrust of the baseline engine E1 at cruise condition ($F_{N \text{ ref}}$). For the bare engine, the added +11% $F_{N \text{ ref}}$ benefit from θ_{exhaust} is offset by the -14% $F_{N \text{ ref}}$ drag penalty associated with the nacelle (D_{nac}^*) to provide an isolated Net Propulsive Force (NPF) -3.2% relative to the bare engine standard net thrust (F_N).

To evaluate the interference terms due to the installation onto the aircraft, the interference for thrust (ΔF_N^*), modified nacelle drag (ΔD_{nac}^*) and airframe drag ($\Delta D_{A/F}$) were considered in the installation force diagram (Figure 6.26). For the different installation positions there were different sensitivities of the interference forces which ultimately affected the final resultant force (F_{inst}). The overall sensitivity of F_{inst} to the engine position was 1.7% $F_{N \text{ ref}}$ depending on the engine location. The decomposition of the overall effect on to the individual interference terms was assessed. The interference on thrust (ΔF_N^*) was predominantly sensitive to the engine axial position and the overlap of the exhaust with the wing ($\Delta X/L_{nac|E1} = -0.06$, positions C). For these cases there was a penalty where $\Delta F_N^* < 0$ which reduced the obtained engine thrust (F_N^*). For the group of positions A and B ($\Delta X/L_{nac|E1} = -0.45$ and $\Delta X/L_{nac|E1} = -0.26$), the changes in ΔF_N^* were relatively minor with a slight beneficial effect of $\Delta F_N^* < +1\%$. For all installation positions the interference effect on the nacelle (ΔD_{nac}^*) was advantageous with benefits in the order of +2.6% to +5.1% $F_{N \text{ ref}}$. These benefits were slightly more pronounced for the group of engines in positions A and B which were further forward of the wing leading edge. The largest change in ΔD_{nac}^* was +5.2% $F_{N \text{ ref}}$ which arose for position A2 and partially offsets the -14% associated with the isolated nacelle drag D_{nac}^* . In general, for all engine positions, there was an adverse effect on the airframe with a penalising $\Delta D_{A/F}$ which ranges from -0.3% to -6.6% $F_{N \text{ ref}}$. The engines in the relatively close axial position (positions C, $\Delta X/L_{nac|E1} = -0.06$) have less of an impact on $\Delta D_{A/F}$. However, the engines positioned further forward (positions B, $\Delta X/L_{nac|E1} = -0.26$ and positions A, $\Delta X/L_{nac} = -0.45$) have a more notable impact with $\Delta D_{A/F}$ in the range of -4.7% to -6.6% $F_{N \text{ ref}}$. This typically offset the previous benefits in ΔD_{nac}^* for this family of installation positions.

To indicate the sensitivity of the interference effects, a map of the overall interference (F_{interf}) effects was created based on the nine engine positions for the E1 engine (Figure 6.27) with the use of a natural neighbour interpolation based on Delaunay triangulation¹¹³. Although nine points were relatively few to create detailed contour maps, the visualization was considered helpful to understand the key sensitivity at a preliminary design stage. In the considered range, the position C3 was the most beneficial in terms of F_{inst} and other results were presented as relative to this position. Based on the interference map (Figure 6.27), a trend along the C3-A1 diagonal was observed. Movement of the engine closer to the wing vertically or upstream axially was detrimental as compared with position C3. The worst position was A2 where it was approximately -1.7% of reference thrust relative to C3. It has to be noted that the pylon was neglected in

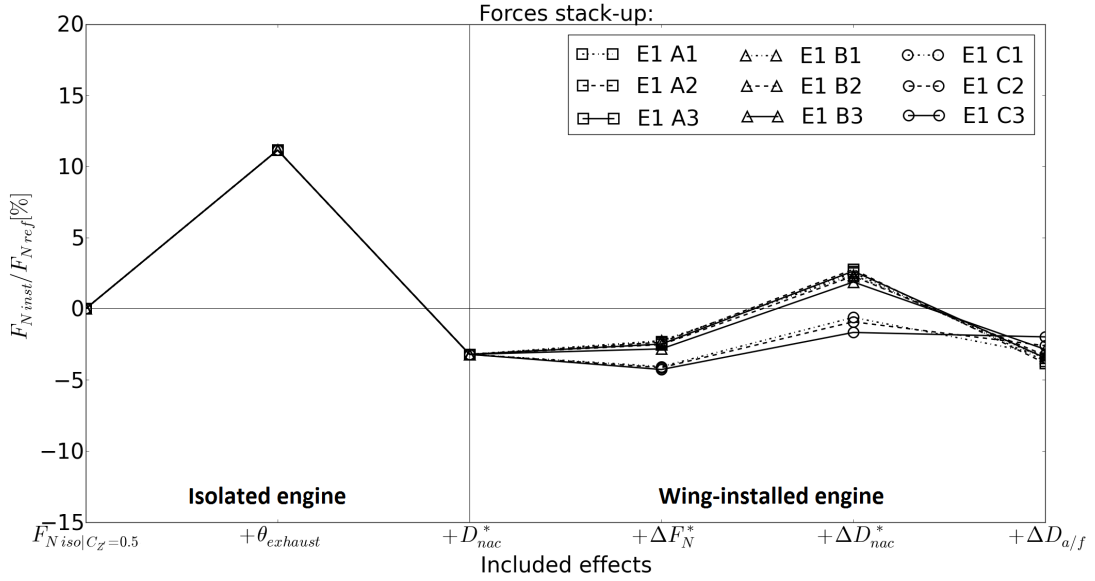


FIGURE 6.26: Overall installation force diagram for engine E1 in different engine positions; standard net thrust of isolated engine at cruise ($F_{N\ iso|C_z'=0.5}$) plus the effect of exhaust forces ($+\theta_{exhaust}$), nacelle drag ($+D_{nac}^*$), and detailed interference effects ($\Delta F_N^* / \Delta D_{nac}^* / \Delta D_{A/F}$); $Re = 45 \times 10^6$; $M = 0.82$; $C_{Z'} = 0.5$

the study of engine positions and it a bespoke and optimised shape of a pylon for each configuration would be required.

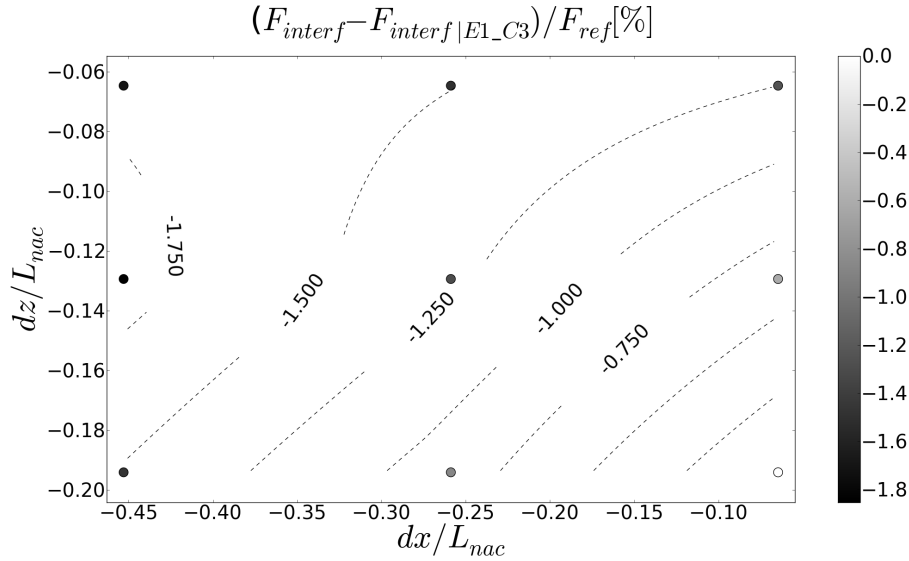


FIGURE 6.27: Map of the overall interference (F_{interf}) for baseline engine (E1), where the axial (dx) and vertical (dz) displacements were referenced to respective nacelle lengths (L_{nac}); $Re = 45 \times 10^6$; $M = 0.82$; $C_{Z'} = 0.5$

The impact of engine position on the overall installation effect for the larger engine E2 was also considered. The E2 engine was tested for the same range of positions as engine E1 and between the two engine sizes the location of the nacelle trailing edge ($\Delta X/L_{nac|E1}$; $\Delta Z/L_{nac|E1}$) was kept constant. Similarly, the installation force diagram

(Fig. 11) for the E2 engine was created to decompose the overall installation effect (F_{inst}) in to the drag (D_{nac}^*) and thrust ($\theta_{exhaust}$) terms for isolated engine as well as the interference terms (ΔF_N^* , ΔD_{nac}^* , $\Delta D_{A/F}$). For this larger E2 engine, the effect of the $\theta_{exhaust}$ was $+18\%F_{N\ ref}$ which was offset by $-21\%F_{N\ ref}$ for the D_{nac}^* to give an overall NPF for the isolated engine which was -2.5% lower than the bare engine standard net thrust (Figure 6.28). As expected, for aerodynamically similar nacelle designs for E1 and E2, the engine modified drag (D_{nac}^*) approximately scales with the engine fan area. However, as for the $\theta_{exhaust}$, the benefits were approximately $+2\%$ greater than an evaluation based on the geometrical relation between the engines. It was explained by the reduced and more favourable FNPR for the E2 engine with 2.11 as compared with 2.71 for the E1 engine. Overall, the isolated engine E2 performed $+0.7\%$ better than E1 engine in terms of the isolated net propulsive force.

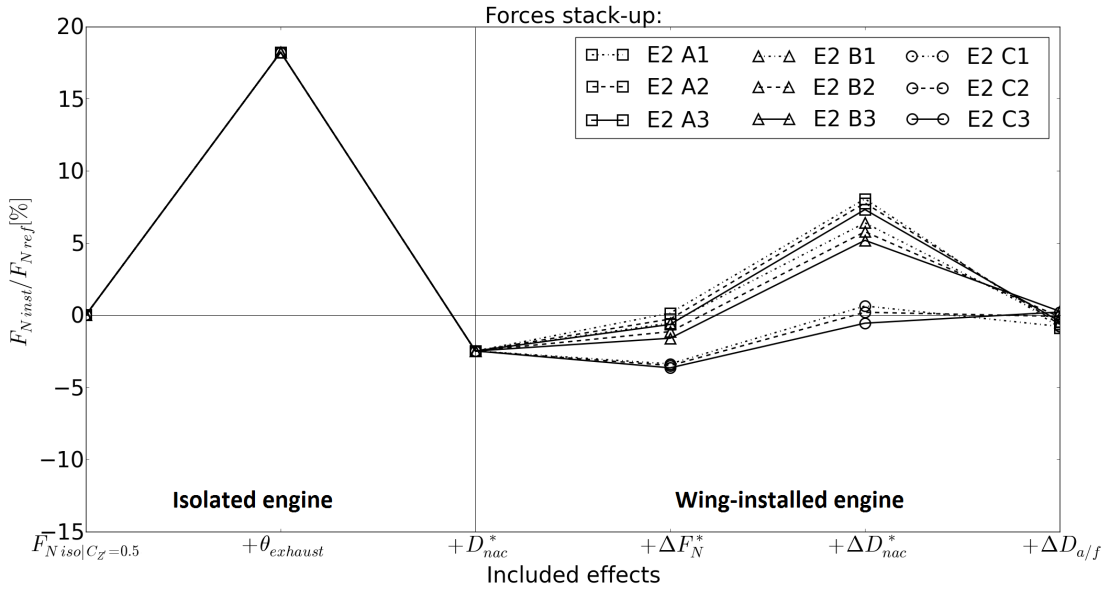


FIGURE 6.28: The overall installation force diagram for engine E2 in different engine positions; standard net thrust of isolated engine at cruise ($F_{N\ iso|C_z'=0.5}$) plus the effect of exhaust forces ($+\theta_{exhaust}$), nacelle drag ($+D_{nac}^*$), and detailed interference effects (ΔF_N^* / ΔD_{nac}^* / $\Delta D_{A/F}$); $Re = 45 \times 10^6$; $M = 0.82$; $C_{z'} = 0.5$

For the E2 engine, there were some notable differences in the interference terms which depend on the engine position and overall the final installed NPF varies by $1.3\% F_{N\ ref}$ across the full range of positions (Figure 6.28). This was less than the range of 1.7% which was calculated for the smaller E1 engine across the same range. For the E2 engine, the effects of the interference reflect some of the aspects observed for the smaller E1 engine. For example, the group of engines positioned closest to the wing leading edge ($\Delta X/L_{nac|E1}=-0.06$, C1, C2, C3) exhibited a negative effect of the exhaust interference (ΔF_N^*) of approximately $-1\% F_{N\ ref}$. The other engines positioned further upstream show a beneficial impact from about $+0.9\%$ to $+2.6\% F_{N\ ref}$. All of the engines showed a beneficial impact of the interference on the nacelle across the range of $+3\%$ to $+8\% F_{N\ ref}$ with, in general, the largest benefits observed for the engines furthest upstream

of the wing with $\Delta X/L_{nac|E1}=-0.45$ (group A). The effect of interference on the aircraft ($\Delta D_{A/F}$) ranged from slightly positive (0.8%) for the close coupled position C3 ($\Delta X/L_{nac|E1}=-0.06$, $\Delta Z/L_{nac|E1}=-0.26$) to significantly negative of up to -9% for the other configurations further upstream (groups A and B).

The sensitivity of the overall impact on the resultant force (F_{inst}) depended on the balance of these competing interference elements and was visualised based on the Delaunay triangulation as previously for the range of non-dimensional positions $\Delta X/L_{nac|E1}$ and $\Delta Z/L_{nac|E1}$ (Figure 6.29). The spatial sensitivity of all configurations was presented as related to the configuration C3 (Figure 6.29). For the larger E2 engine, the best engine locations with the greatest resultant force were B3 and C3. This was further upstream than the best position for the smaller E1 engine which was at C3. The distribution highlighted the reduced sensitivity of the resultant force (F_{inst}) due to the installation position for the E2 engine (Figure 6.29) as compared with the E1 engine (Figure 6.29). The greatest penalty for the E2 engine was observed at the A1 position, which was similar to the highest penalty for E1 at the positions A1 and A2. However, the detrimental effect for the engine at position A1 for E1 engine was approximately 1.5 times greater than for the E2 engine.

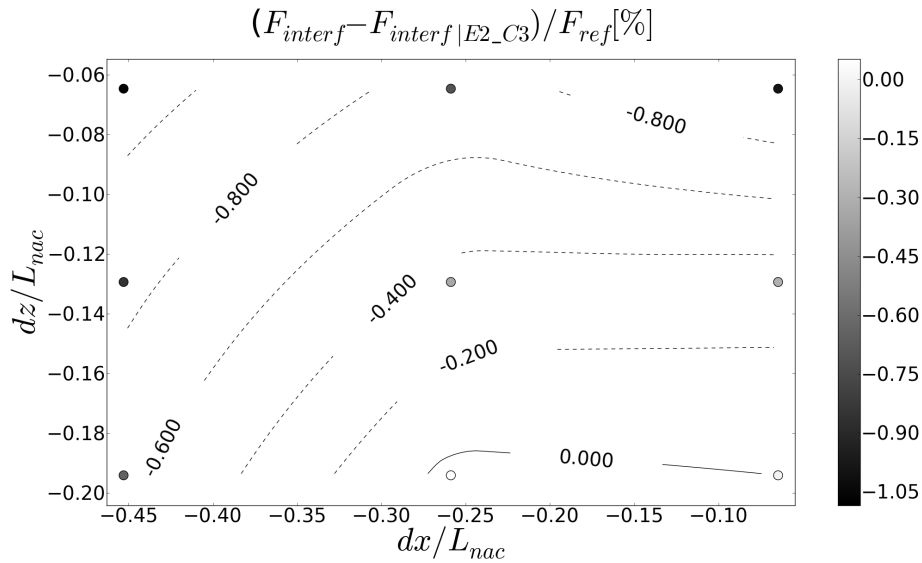


FIGURE 6.29: Map of the overall interference (F_{interf}) for the larger engine (E2), where the axial (dx) and vertical (dz) displacements were referenced to respective nacelle lengths (L_{nac}); $Re = 45 \times 10^6$; $M = 0.82$; $C_{Z'} = 0.5$

6.6 The effect of engine size

The effect of engine size on the effects of engine installation was addressed in two ways. Based on the mission requirement the thrust requirement of the engine was constant. Therefore, the core work on the effect of engine size is completed with the larger engine, which was designed to keep the same thrust (Section 6.6.1). To isolate the effect of engine size at a constant massflow capture ratio and constant efflux, the geometrically enlarged engines were considered. However, the analysis was considered auxiliary due to the unrealistic values of thrust generated by the geometrically enlarged engines and the analysis was located in the Appendix E.3. To determine the impact of the large size engine on the drag and lift of the airframe, three engine sizes were considered E1, 1.23xE1 and 1.45xE1 (Appendix E.3.1). Furthermore, to decouple the effect of the engine size from the change in the specific thrust a direct comparison between the E2 engine and a geometrically enlarged engine 1.23xE1 was completed (Appendix E.3.2). The assessment of geometrically enlarged engines provides valuable lessons for the prospective generation of a prediction model for engine installation (Chapter 7).

6.6.1 The effect of size with constant thrust

To evaluate the impact of engine size on the installation effects, two size engines were considered over a range of positions. The baseline engine E1 ($D_{FF\ Base}$, Figure 6.30a) was compared with the very-high by-pass ratio engine E2 ($1.23D_{FF\ Base}$, Figure 6.30b) and over the range of engine positions considered, the position of the trailing edge point of the nacelle top-line was kept constant between the engines.

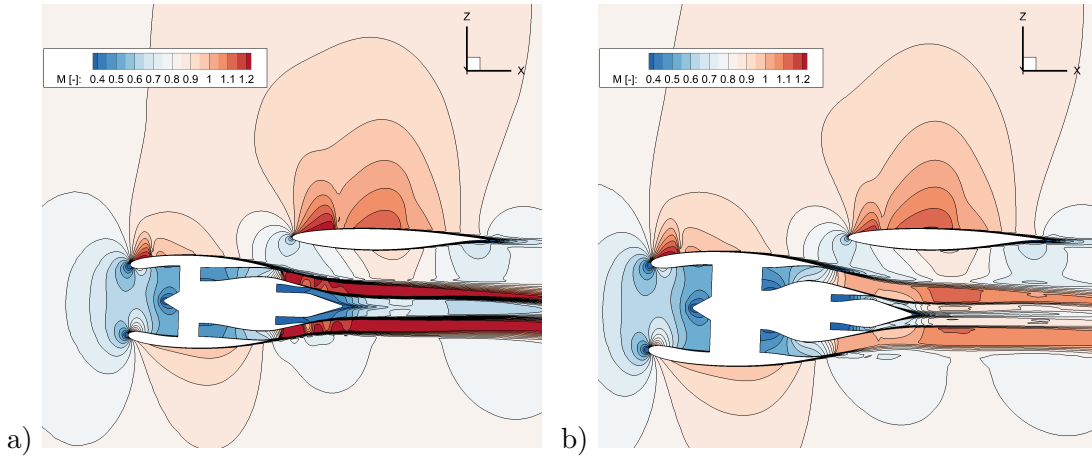


FIGURE 6.30: Contours of Mach number for engine-aircraft configurations a) with E1 engine and b) with E2 engine in engine position C3 at $AoA_{A/C} = 2.5^\circ$; $Re = 45 \times 10^6$; $M_\infty = 0.82$

The replacement of the E1 engine with E2 engine resulted in notable changes to the flowfield (Figure 6.30) and wing aerodynamics. For example, at a typical cruise incidence

of $AoA_{A/C} = 2.5^\circ$, a reduction of the peak suction on the upper side of the wing and the upstream movement of the shock were observed. The changes in the local aerodynamics led to the overall effect in the aerodynamic forces and the installation of the engine resulted in the increase of total drag ($C_{D\ tot}$) (Figure 6.31a) which was the sum of airframe drag ($C_{D\ A/F}$) and the nacelle drag ($C_{D^* \ nac}$). At a constant overall vertical force ($C_{Z'}$) that included the aerodynamic lift ($C_{L\ tot}$) and the vertical component of thrust (GPF_{vert}), the $C_{D\ tot}$ increased by 27 dc for the E1 configuration and by 36 dc for E2 configuration, as compared with the clean-wing airframe. The installation penalty in the drag domain was approximately proportional to the increase of engine size as the drag of the nacelle was the dominant component of the installation drag. The changes in the aerodynamics also affected the lift force of the aircraft ($C_{L\ tot}$, Figure 6.31b). For the baseline engine E1 at a typical cruise incidence of $AoA_{A/C} = 2.5^\circ$, the effect of loss of lift as compared with the clean-wing configuration was $\Delta C_{L\ tot} = -0.015$. The loss of lift was predominantly an effect of changes to local wing aerodynamics, as the engine contribution to the lift force was $C_{L\ nac} = +0.001$ and vertical component of thrust $+0.005$. As compared with the baseline engine E1, the E2 engine resulted in a further reduction of lift coefficient of $\Delta C_{L\ tot} = -0.020$ with a negligible beneficial contribution of the nacelle $C_{L\ nac} = +0.002$ and vertical component of thrust $+0.007$.

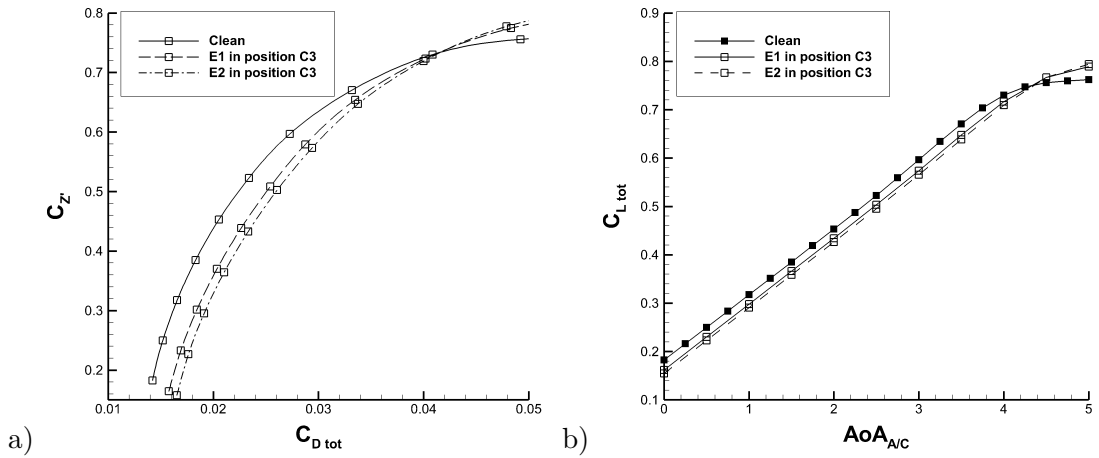


FIGURE 6.31: a) Overall vertical force ($C_{Z'}$) and total drag ($C_{D\ tot}$) polar for CRM aircraft without (Clean) and with engines (E1 and E2), compared with drag polar of the clean-wing; b) total lift ($C_{L\ tot}$) as function of aircraft incidence ($AoA_{A/C}$)
 $Re = 45 \times 10^6$; $M_\infty = 0.82$

The effect on the aircraft aerodynamic performance was predominantly the result of changes to the pressure distribution. It was determined that the changes of the viscous forces due to the increase of engine size were triggered mainly by the upstream movement of the shock location on the upper side of the wing. The magnitude of the contribution to overall drag force was broadly two orders of magnitude less significant than the changes to the pressure distributions. The impact of the larger engine as compared with the baseline engine on the pressure distribution was of interest for the wing cross-section closest to the installation at $y/(b/2) = 0.3$ (Figure 6.32). For both engine-airframe

configurations E1 and E2 (Figure 6.32), the reduction of the suction on the upper side of the wing leads to a reduction of local lift coefficient and as a consequence a local reduction of lift induced drag on the wing. Compared with the clean-wing (Figure 6.32) the installation of baseline engine E1 also changed the pressure distribution on the lower side of the wing. The increased acceleration $\Delta C_p = -0.25$ occurs towards the wing leading edge $x/c = 0.05$. Beyond $x/c = 0.1$ a beneficial effect of increased pressure was observed with a maximum increase of $\Delta C_p = +0.3$ at approximately $x/c = 0.4$. The pressure distribution for the aft section of the wing ($x/c > 0.7$) of the E1 configuration was the same as the distribution for the clean-wing. Compared with the E1 engine, the installation of larger E2 engine resulted in broadly the same peak acceleration at the wing leading edge and a relatively modest increase of pressure by $\Delta C_p = +0.05$ at $x/c = 0.1$ and at $x/c = 0.5$. The E2 engine has a significantly lower fan nozzle pressure ratio (FNPR) of 2.11 as compared with 2.71 for E1 engine. The increased suction on the lower side of the wing for the E1 configuration was a result of stronger influence on the wing surface from the expansion waves in the jet.

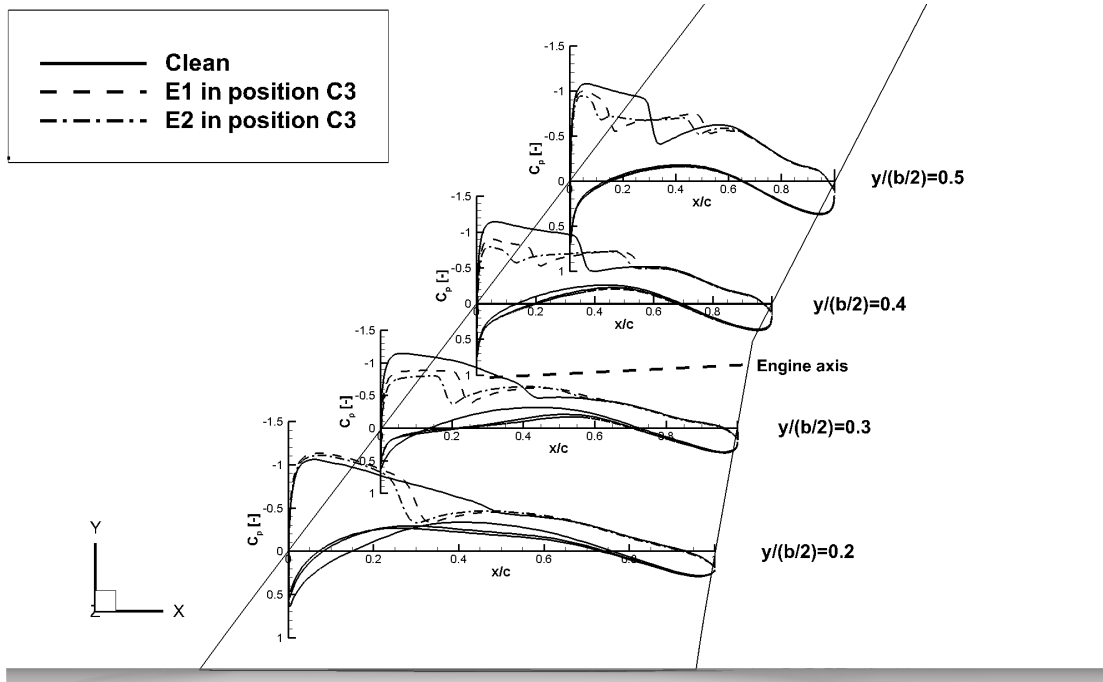


FIGURE 6.32: Pressure distribution for aircraft without (Clean) and with engines (E1 and E2); $Re = 45 \times 10^6$; $M_\infty = 0.82$; $AoA_{A/C} = 2.5^\circ$; $y/(b/2)$ from 0.2 to 0.5

On the inboard side of the engines ($y/(b/2) = 0.2$, Figure 6.32), the presence of the installation caused an increase of peak suction by $\Delta C_p = -0.1$ on the upper side of the wing for both engine configurations and it moves the shock location forward by $\Delta x/c = -0.2$ and $\Delta x/c = -0.25$ for E1 and E2 configurations, respectively. On the lower side of the wing ($y/(b/2) = 0.2$), a significant flow acceleration $\Delta C_p = -0.25$ was observed between $x/c = 0.05$ and $x/c = 0.2$. The changes to the pressure distribution on the wing affect the spanwise lift distribution. On the inboard side of the engine between

$y/(b/2) = 0.1$ and 0.3 , the magnitude of the reduction of local lift coefficient was below -0.04 and -0.05 for E1 and E2 configuration, respectively.

On the outboard side of the wing ($y/(b/2) = 0.4$, Figure 6.32), the presence of the installation reduced the peak suction on the upper side of the wing by $\Delta C_p = +0.2$ and $\Delta C_p = +0.3$ for E1 and E2 configuration respectively. Two rather weak shocks in the pressure distribution were noted at approximately $x/c = 0.2$ and $x/c = 0.6$ for both engine configurations as compared with a single shock at $x/c = 0.4$ for the clean-wing. The impact of the installation on the lower side of the wing was modest for the outboard side of the wing ($y/(b/2) = 0.4$). As a result of pressure distribution, the local loss of lift reduces to less than -0.01 on the outboards side of the engine ($y/(b/2) > 0.4$) and no difference between the two engine configuration was observed at approximately $y/(b/2) = 0.6$ and further to the tip of the wing.

To localise the changes in pressure and viscous forces and to highlight the importance of the changes to the overall drag, the difference of local drag force per unit area between the E1 configuration and the idealised ‘no-interference’ configuration is calculated (Figure 6.33a). As for the upper side of the wing (Figure 6.33a), the installation of the E1 engine reduced the local drag due to the movement of the shock location upstream on the inboard side of the installation. The changes to the drag occur predominantly for the root and centre sections of the wing with negligible difference to the wing tip. The wing pressure-field affects the force distributions on the engine. The fan-cowl, core-cowl and plug surfaces were presented in cylindrical coordinate system (ξ, Θ) (Figure 6.34a). A notable beneficial effect on the afterbody of the fan-cowl was present due to the forward pressure force exerted by the wing pressure field. Moreover, the shock pattern in the jet was affected by the interaction with the pressure field and the force distribution on the core-cowl was changed but no overall beneficial trend was observed. At the surface of the plug, a beneficial pressure force was also exerted due to the pressure-field of the wing. Furthermore, the difference in drag distribution between the E2 configuration compared with the clean-wing and isolated engine was analysed (Figure 6.33b). The installation of E2 engine caused a greater reduction of drag at the root of the wing as compared with the E1 engine. For the engine surfaces, the beneficial pressure force was exerted by the wing pressure-field. A modest interaction of the wing pressure-field with the jet was observed in the forces distribution, due to the relatively low FNPR for the E2 configuration (Figure 6.34b) as compared with the E1 engine. A modest beneficial pressure force was exerted by the wing pressure field on the surface of the engine core-cowl.

To quantify the overall installation effect, an installation force diagram (Figure 6.35) for two engines E1 and E2 was developed. The cruise operating point was chosen with the coefficient for the overall vertical force of $(C_{Z'}) = 0.5$ at $M_\infty = 0.82$. The installation effects were considered for a re-trimmed aircraft and were referenced to the value of standard net thrust of the baseline engine E1 at cruise condition $(F_{N \text{ ref}})$ (Figure 6.35). The isolated net propulsive force (NPF) was deconstructed into beneficial thrust terms

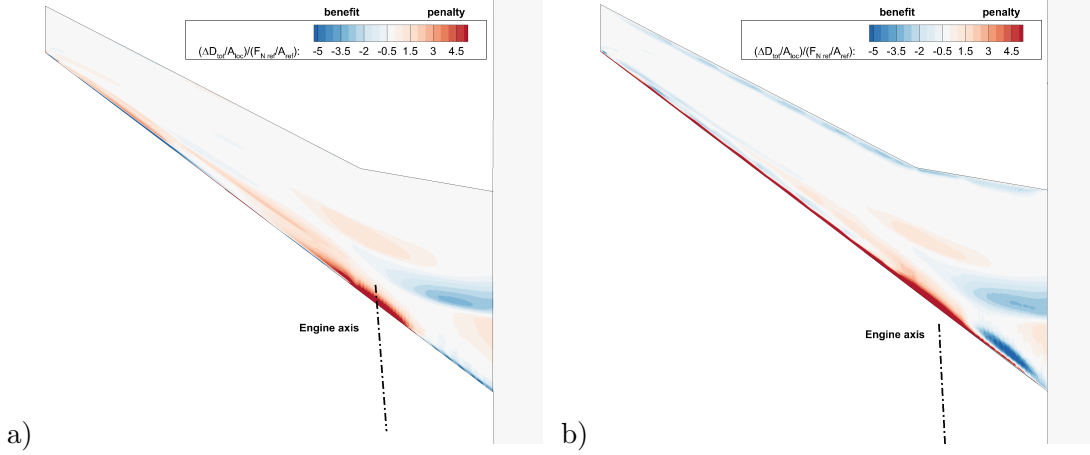


FIGURE 6.33: Distribution of difference drag force per area (ΔD_{tot}) between a) configuration E1 and clean-wing configuration; between b) configuration E2 and clean-wing configuration; $Re = 45 \times 10^6$; $M_\infty = 0.82$; $AoA_{A/C} = 2.5^\circ$

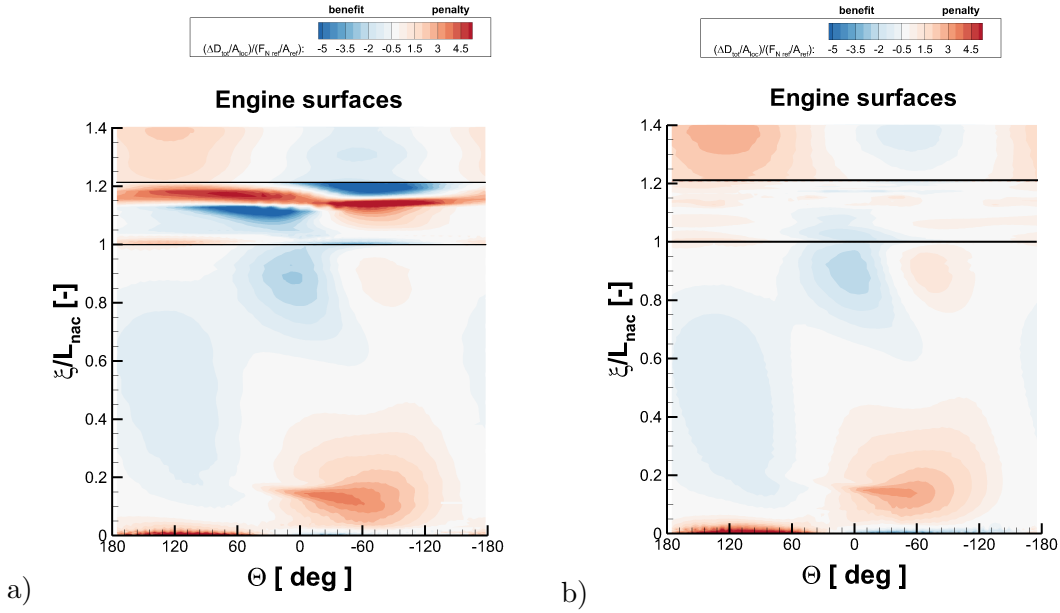


FIGURE 6.34: Distribution of difference drag and thrust force per area ($\Delta \phi_{tot}$) between a) configuration E1 and isolated engine; between b) configuration E2 and isolated engine; $M_\infty = 0.82$; $AoA_{A/C} = 2.5^\circ$

(F_N and $\theta_{exhaust}$) and detrimental modified nacelle drag (D_{nac}^*) for both engines. As a result, the difference in isolated NPF between the two engines was determined to be a benefit of $+0.7 F_{N \text{ ref}}$ for the larger engine E2. It has to be noted that the D_{nac}^* increased approximately proportionally to the engine size. The benefit for the larger engine comes purely from the advantageous design of the exhaust system. The relatively high values of the boat-tail angles 14° for the core-cowl and 16° for the plug result in the divergent channel created for the by-pass and core flows. The decelerating flow causes the presence of relatively positive pressure field that acts on the large projected area of the nozzle in the thrust direction. As a result, a significant beneficial pressure force increases the

$\theta_{exhaust}$.

To include the interference terms due to the installation onto the aircraft, the interference for thrust (ΔF_N^*), modified nacelle drag (ΔD_{nac}^*) and airframe drag ($\Delta D_{A/F}$) were considered (Figure 6.35a). Considering the overall interference effect (F_{interf}) at baseline position C3, a benefit of +1.2% of $F_{N \text{ ref}}$ and +2.7% of $F_{N \text{ ref}}$ was observed for the engines E1 and E2, respectively. The overall difference in the installation force (F_{inst}) between the two engines was +2.2% for the larger engine as a result of benefits in isolated NPF and advantageous interference (F_{interf}) as compared with the baseline engine. The benefit for E2 as compared with E1 was explained by an improvement in ΔD_{nac}^* by +0.4% $F_{N \text{ ref}}$ and in $\Delta D_{A/F}$ by +1% $F_{N \text{ ref}}$ for the E2 engine. Meanwhile, a very modest difference between the engines was observed in ΔF_N^* by +0.1% $F_{N \text{ ref}}$ in favour of the E2 engine. Moreover, the sensitivity of overall interference and the sensitivity of the individual terms was dependent on the engine position. By locating the engines vertically closer to the wing (position C1, Figure 6.35b), the overall installation force (F_{inst}) increased by +2.45% $F_{N \text{ ref}}$ for the E2 engine as compared with E1 at corresponding position. The major advantage for the larger engine comes from the improvement in $\Delta D_{A/F}$ by +1.2% $F_{N \text{ ref}}$, as the improvement in ΔD_{nac}^* was +0.5% $F_{N \text{ ref}}$ and the effect on ΔF_N^* was neutral as compared with E1. The movement of the engines upstream from the wing affected the interference terms significantly (position A1, Figure 6.35c). A notable benefit in ΔF_N^* of +1.7% $F_{N \text{ ref}}$ and improvement in ΔD_{nac}^* by +3.1% $F_{N \text{ ref}}$ were observed for the E2 engine as compared with E1 at a corresponding position. However, the interference on airframe $\Delta D_{A/F}$ became more penalising for the larger engine by -2.7% $F_{N \text{ ref}}$ and it offsets the gains from the thrust and nacelle interferences. As a result, the difference in the overall installation force between E2 and E1 engines at position A1 results in +2.8% $F_{N \text{ ref}}$ for E2 engine. To assess the spatial sensitivity of the overall installation effect, a map of difference in F_{inst} was considered (Figure 6.36). Apart from the benefit for E2 as compared with E1 in terms of isolated NPF (+0.7% $F_{N \text{ ref}}$), the additional difference in F_{interf} was considered. The difference in overall installation effect between the two configurations was a result of considerably different sensitivities of interference effect (F_{interf}) for each engine. Relative to the E1, the +2.2% $F_{N \text{ ref}}$ benefit for E2 engine at position C3 increased up to +3.2% $F_{N \text{ ref}}$ benefit for E2 at position A2. Even though, the position A2 was less favourable for both engines as compared with C3, it was relatively advantageous to locate a larger engine in position A2, if other design constraints allow it.

The aerodynamic benefit for the larger engine E2 was not expected. Therefore, the same results as in Figure 6.35 were re-assessed with a different philosophy in mind (Figure 6.37). The forces that acted on the engine and the installation were divided into thrust group and thrust requirement group. The thrust group consisted of the standard net thrust (F_N) and the exhaust surface forces ($\theta_{exhaust}$), which summed up to the modified net thrust (F_N^*). The thrust requirement group represented the drag of the nacelle

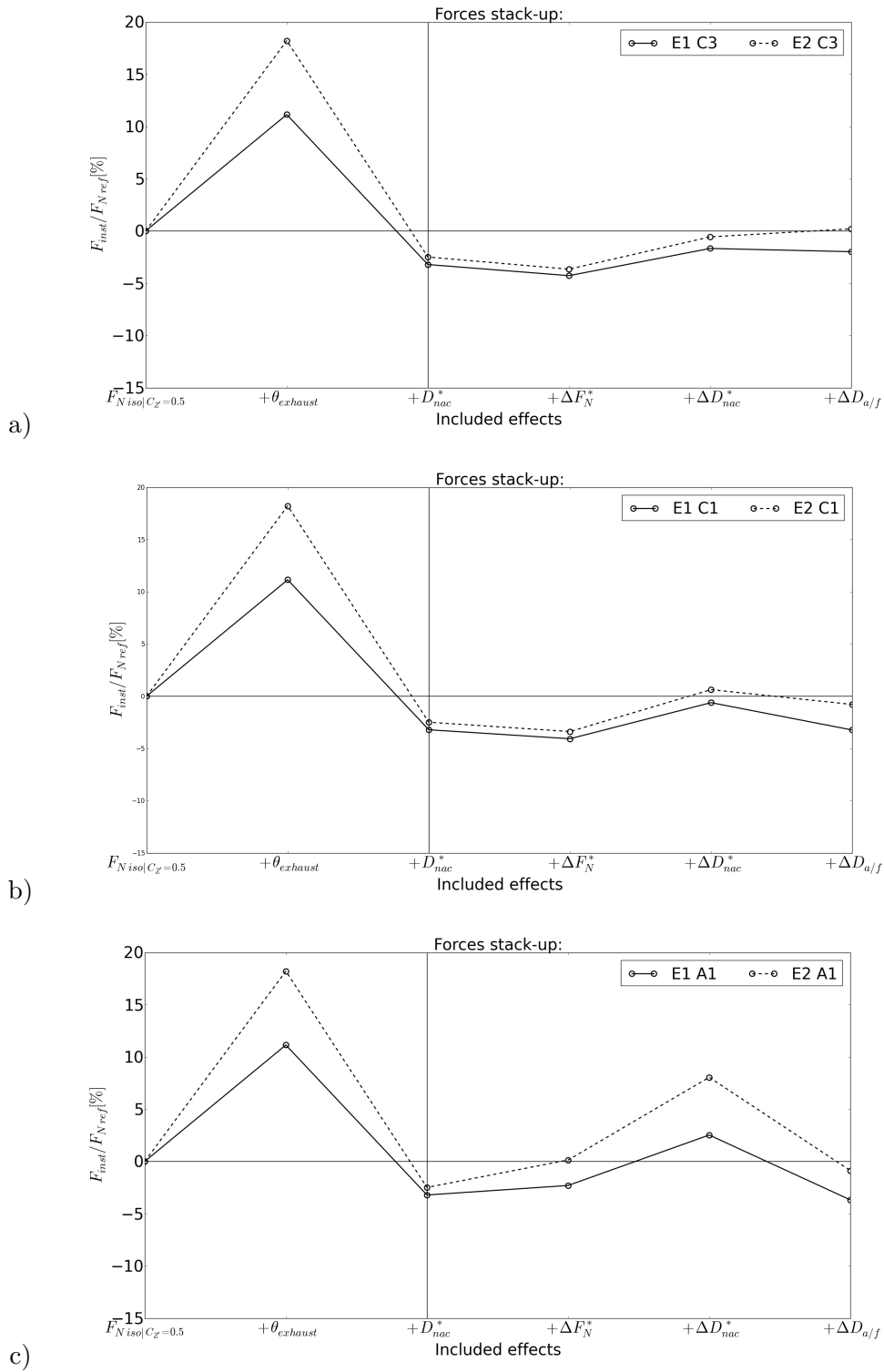


FIGURE 6.35: Installation force diagram for two engine sizes E1 and E2 in a) baseline position C3 b) close coupled position C1 and c) furthest upstream position A1; standard net thrust of isolated engine at cruise ($F_{N \text{ iso}}|_{Cz'=0.5}$) plus the effect of exhaust forces ($+\theta_{\text{exhaust}}$), nacelle drag (D_{nac}^*), and detailed interference effects ($\Delta F_N^* / \Delta D_{nac}^* / \Delta D_{A/F}$); $Re = 45 \times 10^6$; $M = 0.82$; $C_{Z'} = 0.5$

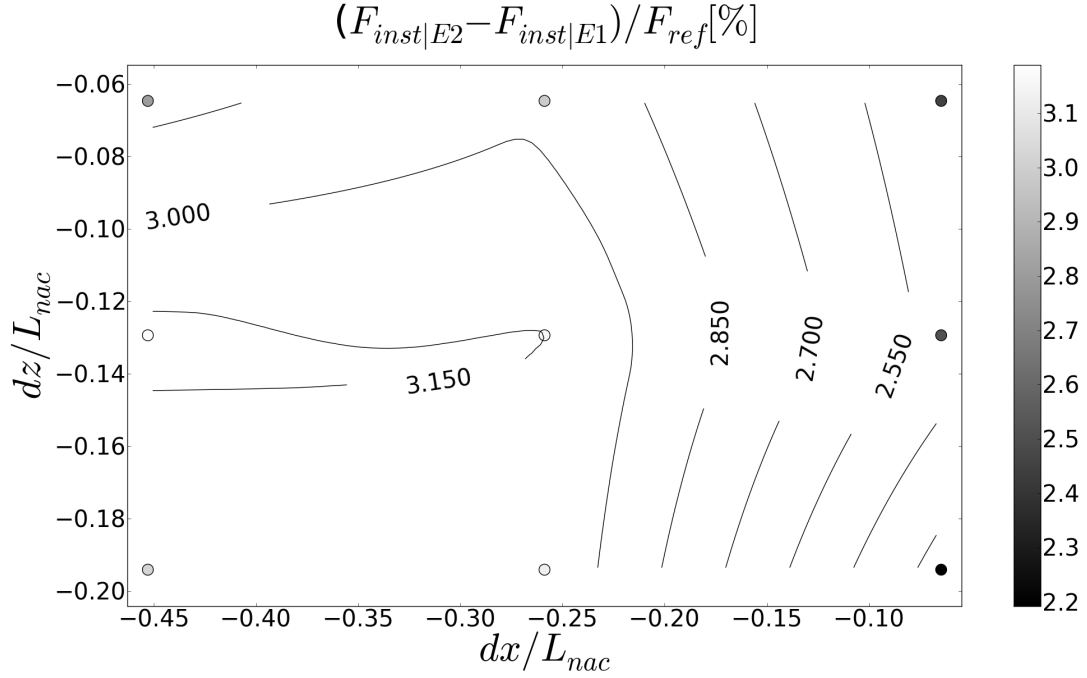


FIGURE 6.36: Map of the difference in overall installation effect ($F_{inst|E2} - F_{inst|E1}$) between E2 and E1 engines, where the axial (dx) and vertical (dz) displacements were referenced to nacelle length of E1 engine ($L_{nac|E1}$); $Re = 45 \times 10^6$; $M = 0.82$; $C_{Z'} = 0.5$

(D_{nac}^*) and the interference forces (ΔF_N^* , ΔD_{nac}^* , $\Delta D_{A/F}$). The thrust requirement group could be represented by the installation effect in the sense of ALVAST studies^{8–10} (X_{inst} ; Appendix A.2). Indeed, the thrust requirement and the installation drag X_{inst} increased for the larger engine E2. As compared with E1, the E2 engine required 4.3%, 4.6% and 4.8% more thrust (F_N^*) for positions A1, C1 and C3, respectively. However, it has to be noted that the decomposition of the thrust (F_N^*) changed with the engine size and 18% of thrust for the large engine was generated as the force on the core-cowl and plug, as compared with 11% for the E1 engine. The result is not contradictory with respect of the benefit in F_{inst} for the E2 engine. Simply, it was relatively easier for the larger engine to obtain the same level of thrust as the E1 engine. Overall, the installation effect (X_{inst}) increased with engine size, but the study indicated that a significant portion of this penalty can be balanced by favourable force on the exhaust. From the mission analysis perspective, either F_{inst} or X_{inst} should be used for compatible definition with the engine performance code. In this research (Section 6.9), the F_{inst} definition was followed as required by the performance code, which used the standard net thrust (F_N) as thrust. To use the X_{inst} , a significant improvement to the quantification methods of the $\theta_{exhaust}$ at a preliminary design stage should be considered, to enhance engine performance tools and optimise engines for the constant value of F_N^* rather than F_N .

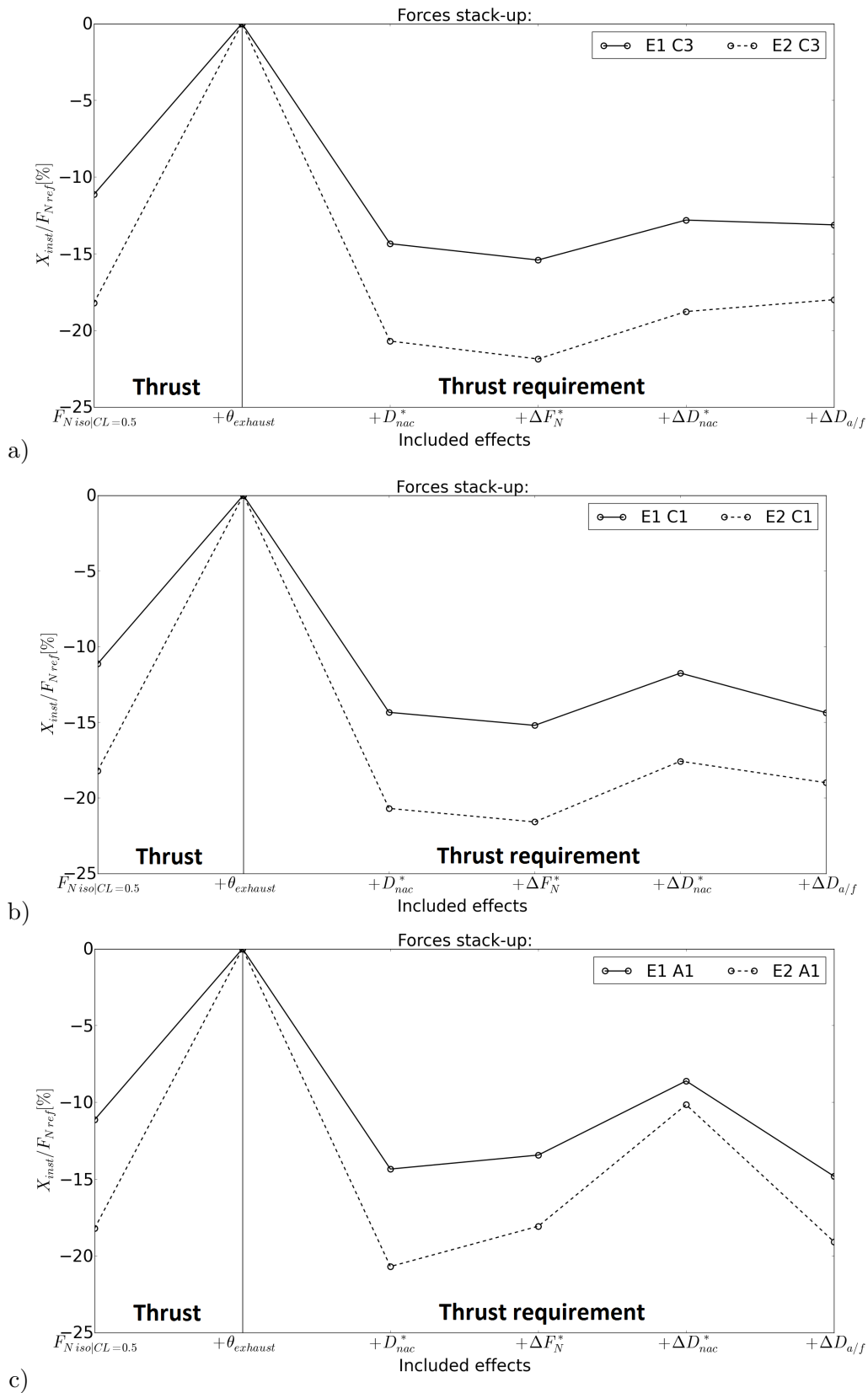


FIGURE 6.37: Installation force diagram (X_{inst}) for two engine sizes E1 and E2 in a) baseline position C3 b) close coupled position C1 and c) furthest upstream position A1; standard net thrust of isolated engine at cruise ($F_{N iso|Cz'=0.5}$) plus the effect of exhaust forces ($+\theta_{exhaust}$), nacelle drag (D_{nac}^*), and detailed interference effects (ΔF_N^* / ΔD_{nac}^* / $\Delta D_{A/F}$); $Re = 45 \times 10^6$; $M = 0.82$; $C_{Z'} = 0.5$

6.6.2 The comparison of results with previous studies

It was useful to compare these results for the impact of engine size with previous experimental studies on ALVAST TPS⁸⁻¹⁰. The details of the thrust and drag book-keeping system used in the ALVAST studies⁸⁻¹⁰ was carefully analysed (Appendix A.2) These studies using TPS systems for a range of engine sizes indicated that the installation penalty ranged from about 35dc to 42dc for a range of engine sizes from D_{FF}/c_{ref} of 0.42 to 0.51. These experimental studies were also under nominal cruise conditions although the Mach number was notably lower⁸ with 0.75. The current computational studies have been conducted at a nominal cruise condition of $M=0.82$ and the initial nominal reference condition for the engines was the standard net thrust ($F_{N\ ref}$). To ensure consistency with the experimental work, the current analysis was modified to present installation drag within the same definition⁸. Taking this into account the current computational results indicate a typical installation penalty ($C_{X\ inst}$) of 33dc for the E1 engine ($D_{FF}/c_{ref} = 0.36$) and 45dc for the E2 engine ($D_{FF}/c_{ref} = 0.45$). For each of these engines, the $C_{X\ inst}$ varies with position by 5dc and 3dc for the E1 and E2 engines, respectively. Although the configurations and operating conditions were different, the general trend of an increase of the $C_{X\ inst}$ with the engine size in agreement with these previous experimental studies. The penalty increases more rapidly with respect to size than expected based on previous study, which was explained by a greater value of Mach number. As a result, approximately 10dc difference is observed for the engine E2 at $D_{FF}/c_{ref} = 0.45$.

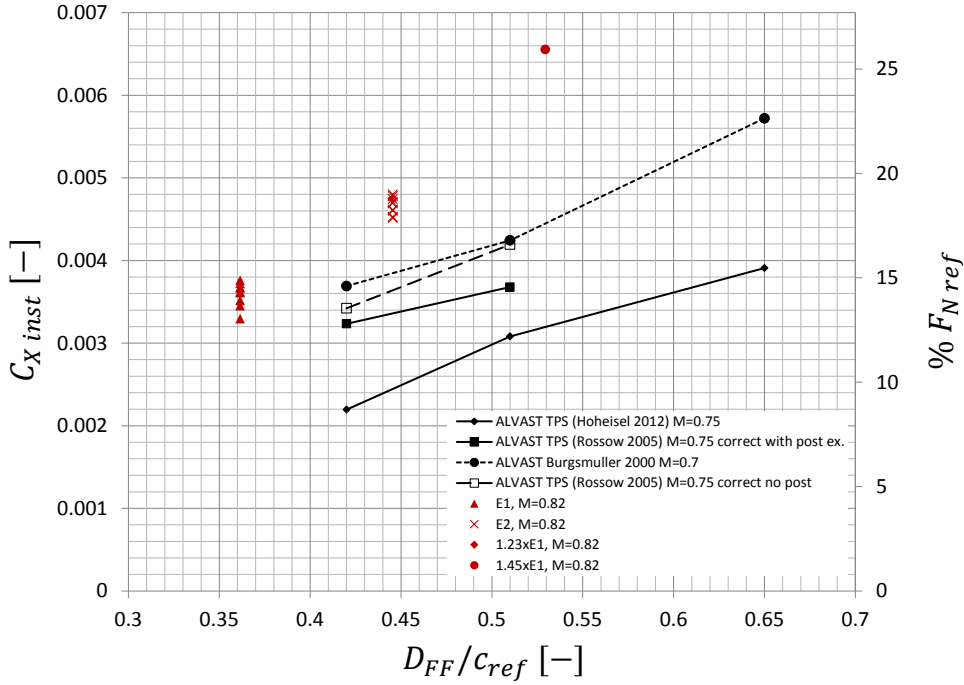


FIGURE 6.38: The installation force in “ALVAST definition” (X_{inst}) at $Re = 45 \times 10^6$; $M = 0.82$; $C_{Z'} = 0.5$; compared with the experimental results from ALVAST studies^{11,52,54}

6.7 The effect of flight Mach number

To determine the effects on drag, lift and thrust due to the presence of engine installation and the sensitivity of the effects to the flight Mach number (M_∞), a series of computations at a constant angle of attack ($AoA_{A/C} = 2.5^\circ$) was completed for a range of M_∞ from 0.7 to 0.86 for the clean-wing CRM and the CRM with E2 engine in position C3. In addition, the isolated engine was computed for the same range of M_∞ at a corresponding isolated engine incidence ($AoA_{eng} = 4.25^\circ$). With the variation of M_∞ , it is impossible to maintain constant relation for all the other operating conditions of the engine-airframe configurations. Therefore, it was decided that the engines at all Mach numbers were operated with the intake mass-flow capture ratio (MFCR) of 0.75 to minimise the spillage and maintain the conditions in the drag domain to be relatively constant. Meanwhile, the exhaust condition was set accordingly based on the engine performance model (Section D.1). The relation between the thrust (F_N^*) and the M_∞ was linear for the constant MFCR. The airframe drag has a non-linear relationship with regards to the M_∞ , and therefore the aircraft remained in force imbalance. To determine the impact of Mach number on the forces in the drag domain, the total aircraft drag ($C_{D\ tot}$) for the clean-wing airframe and the engine-airframe configuration was computed (Figure 6.39).

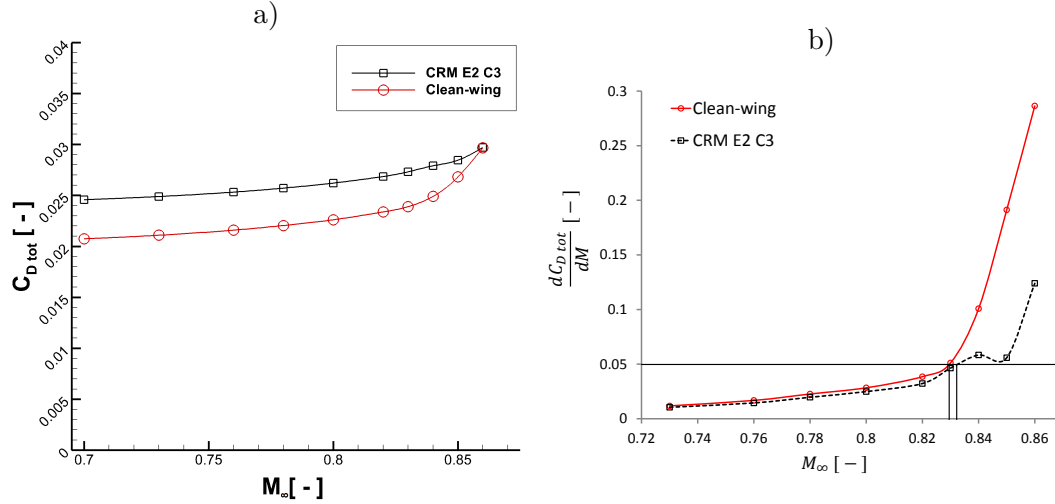


FIGURE 6.39: a) Aircraft total drag ($C_{D\ tot}$) ; b) drag gradient ($\frac{dC_{D\ tot}}{dM}$) as a function of flight Mach number (M_∞) for the clean-wing CRM and the CRM with E2 engine in position C3; $AoA_{A/C} = 2.5^\circ$

At $M_\infty = 0.7$ (Figure 6.39a), the aircraft total drag ($C_{D\ tot}$) was 246 dc and 207dc for the configuration with and without E2 engine, respectively. As anticipated, the increase of Mach number for the aircraft configuration with and without the engine to $M_\infty = 0.82$ increased the $C_{D\ tot}$ to 234dc and 269dc for the configuration with and without E2 engine, respectively. A further increase of the flight Mach number resulted in a significant increase of the $C_{D\ tot}$ for the clean-wing configuration and at $M_\infty = 0.86$ both configuration exhibited 297dc. Based on the slope of the drag curves (Figure 6.39b) the drag rise Mach number (M_{DR}) was computed with the use of the gradient method¹⁰⁵ ($\frac{dC_{D\ tot}}{dM_\infty} = 0.05$). The drag rise Mach number was determined to be at $M_{DR} = 0.83$ for the clean-wing configuration and at $M_{DR} = 0.833$ for the configuration with E2 engine in position C3. It is postulated that the presence of installation reduced the local angle of attack of the wing and in return it decreased the wing-loading. As compared with the clean-wing configuration at a constant $AoA_{A/C}$, the flight Mach number for the engine-airframe configuration could be increased without achieving the maximum wing loading and increasing the wave drag.

To confirm the hypothesis, the decomposition of the total aircraft drag ($C_{D\ tot}$) into the airframe component ($C_{D\ A/F}$) and the modified nacelle drag ($C_{D^* \ nac}$) was analysed. The $C_{D\ tot}$ of the clean-wing configuration is naturally equal to the $C_{D\ A/F}$ (Figure 6.40). As compared with the clean-wing, the $C_{D\ A/F}$ of the E2 configuration is less sensitive to the flight Mach number and the $C_{D\ A/F}$ of 213dc at $M_\infty = 0.7$ increased mildly to 235dc at $M_\infty = 0.85$. The increase by +22dc is considered fairly mild as compared with the increase by +61dc for the clean-wing over the same range of M_∞ . It can be clearly stated that the presence of installation delayed the drag rise on the airframe.

Furthermore, the sensitivity of the nacelle modified drag ($C_{D^* \ nac}$, Figure 6.41) was analysed for the isolated and wing-installed engines to determine the contribution of

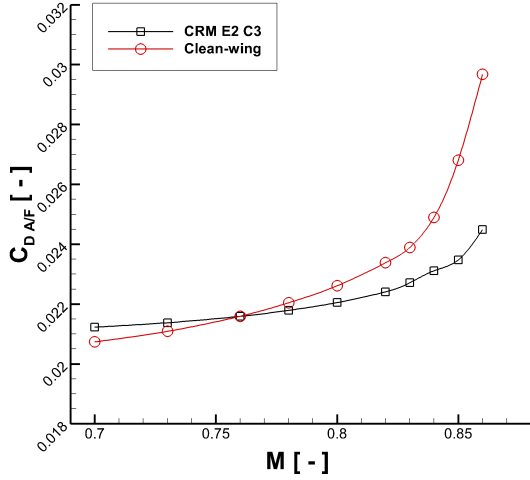


FIGURE 6.40: Airframe drag ($C_{D A/F}$) for the clean-wing and the airframe in presence of E2 engine as a function of flight Mach number (M_∞); $AoA_{A/C} = 2.5^\circ$

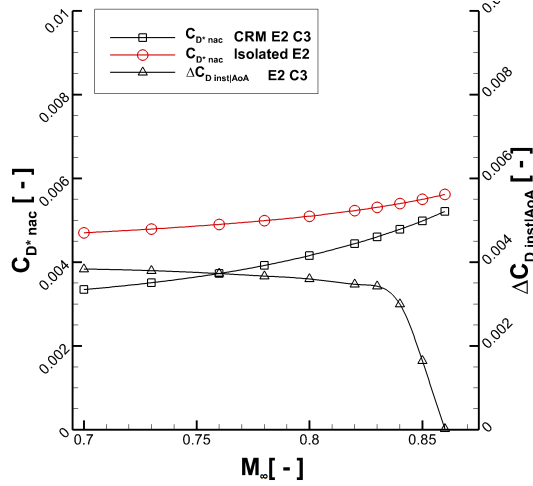


FIGURE 6.41: Modified nacelle drag ($C_{D^* nac}$) for the isolated E2 engine and E2 under the wing, both compared with the installation drag ($\Delta C_{D inst}$) for the presence of E2 engine under the wing of CRM in position C3 as a function of flight Mach number (M_∞); $AoA_{A/C} = 2.5^\circ$

the nacelle to the overall drag rise result. As anticipated, the $C_{D^* nac}$ for the isolated engines rose from 47dc for two engines at $M_\infty = 0.7$ to approximately 56dc at $M_\infty = 0.86$. No sudden drag rise was noted for the tested engine, due to the plausible nacelle aerodynamics and a relatively long fan-cowl. The nacelle modified drag ($C_{D^* nac}$) for a wing-installed engine increased significantly from 33dc for two engines at $M_\infty = 0.7$ to approximately 52dc at $M_\infty = 0.86$. Even though the wing-installed nacelle is more sensitive to the M_∞ , there is a beneficial difference in $C_{D^* nac}$ between the isolated and wing-installed engines at each M_∞ , as a result of a beneficial interaction between the engine and the wing pressure-field. A forward oriented force acted on the afterbody of the fan-cowl and reduced the modified drag of the nacelle ($C_{D^* nac}$) at all flight Mach numbers considered. It can be noted that the slope of the $C_{D^* nac}$ for the wing-installed engine is steeper, which counteracts the delayed drag rise on the airframe. The different sensitivities of the drag rise for the nacelle and airframe significantly affect the installation effect. Based on the aircraft total drag ($C_{D tot}$; Figure 6.39a), the installation drag at a constant angle of attack at each Mach number ($\Delta C_{D inst|AoA}$) was computed (Figure 6.41) to compare with the nacelle modified drag. The $\Delta C_{D inst|AoA}$ of 38 dc at $M_\infty = 0.7$ reduced gradually to 35dc at $M_\infty = 0.83$ and a sudden reduction of $C_{D inst|AoA}$ above $M_\infty = 0.83$ was a result of an early drag rise for the clean-wing configuration. As a result, the $\Delta C_{D inst|AoA}$ reached 0dc at $M_\infty = 0.86$. The fairly constant value of the installation drag ($\Delta C_{D inst|AoA}$) with the flight Mach number facilitates a potential use of computed databases for mission analysis with a wider range of cruise Mach numbers.

To consider the full extent of installation effects, a loss of lift due to the presence of the engine was considered. Clearly, the presence of the engine had an impact on the lift force of the aircraft ($C_{L\ tot}$; Figure 6.42). As compared with the clean-wing airframe the presence of the engine caused a loss of lift over the entire range of Mach numbers. A fairly constant loss of lift ($\Delta C_{L\ tot}$) of -0.025 was observed for a range of Mach numbers from 0.7 to 0.8. A further increase of Mach number caused an increase in the detrimental effect on lift to a $\Delta C_{L\ tot} = -0.045$ at $M_\infty = 0.85$. However, the loss of lift reduced for $M_\infty = 0.86$ to $\Delta C_{L\ tot} = -0.04$. The compensation for the loss of lift during the flight requires an increase of the aircraft incidence and an associated increase of lift induced drag. The rather constant effect of loss of lift due to the presence of the engine reassures that a broadly similar compensation of $AoA_{A/C}$ will be required over the entire range of M_∞ .

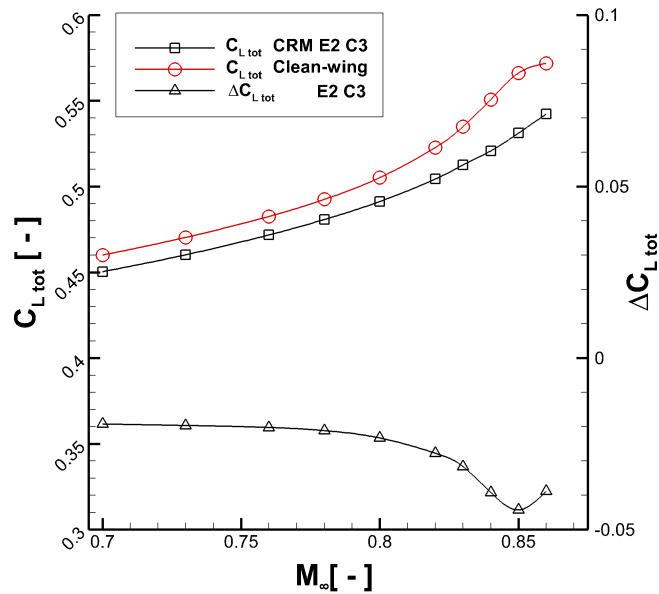


FIGURE 6.42: Aircraft total lift ($C_{L\ tot}$) for the clean-wing CRM and the CRM with E2 engine in position C3; Loss of lift ($\Delta C_{L\ tot}$) due to the presence of E2 engine under the wing of CRM as a function of flight Mach number (M_∞); $AoA_{A/C} = 2.5^\circ$

Both the increase in drag and the loss of lift due to the presence of the engine installation are the result of changes to the pressure distribution on the wing and nacelle (Figure 6.43). To confirm the change of the wing-loading between the clean-wing and engine-airframe configurations, the pressure distribution (C_p) at the wing cross-section inboard of the engine installation ($y/(b/2)=0.3$) was compared between the two configurations over a range of Mach numbers from 0.8 to 0.86. The reduction of the wing loading is predominantly an effect of reduced suction on the upper side of the wing. The presence of installation caused a reduction of the peak ΔC_p of +0.42, +0.4, +0.38, +0.35 for the flight Mach numbers of 0.8, 0.82, 0.84 and 0.86, respectively. Moreover, a significant upstream movement of the shock location ($\Delta x/c$) on the upper section of the wing was observed by -0.15, -0.2, -0.3 and -0.3 for the M_∞ of 0.8, 0.82, 0.84 and 0.86, respectively. As a result the peak suction on the upper side of the wing for the

engine-airframe configurations did not exceed $C_p = -0.7$ and the shocks present in the engine-airframe configuration were significantly weaker than the shocks in clean-wing configuration. The contribution of the wave drag at high Mach numbers is understood to be the main reason of an earlier drag rise for the clean-wing configuration as compared with the engine-airframe configurations. A relatively modest sensitivity of the effect of the engine installation to the flight Mach number was observed for the C_p on the lower side of the wing. Over the considered range of M_∞ from 0.8 to 0.86, the fan nozzle pressure ratio increased from 2.09 to 2.27, which had a very modest effect on the pressure distribution.

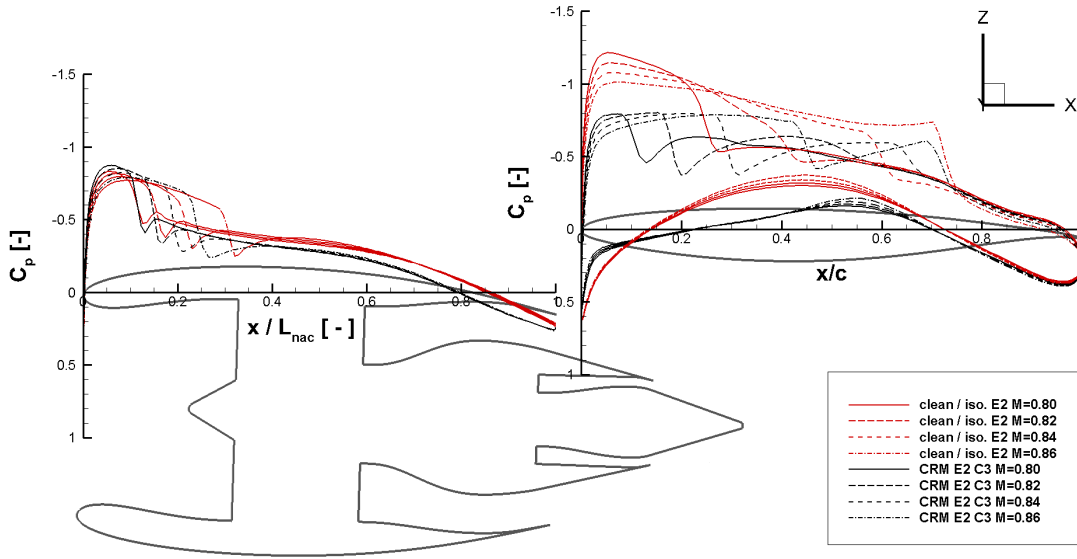


FIGURE 6.43: Pressure distribution (C_p) at the wing cross-section inboard of the engine ($y/(b/2)=0.3$) for the clean-wing and the CRM with E2 engine in C3 position and C_p at the topline of the nacelle for the isolated engine and E2 engine in C3 position; M_∞ from 0.8 to 0.86; $AoA_{A/C} = 2.5^\circ$

In addition, to highlight the origin of beneficial effect of the presence of the wing on the modified nacelle drag ($C_{D^* nac}$), the C_p at the topline of the fan-cowl of the wing-installed engine was compared with the isolated engine over the same range of Mach numbers (Figure 6.43). The C_p distribution on the topline of the nacelle was sensitive to the flight Mach number (M_∞). The wing pressure-field caused an increase in C_p at the afterbody of the fan-cowl by approximately $\Delta C_p = +0.05$ from $x/L_{nac} = 0.6$ to 1.0 for all the Mach numbers investigated. A rather constant effect of the wing pressure-field on the afterbody of the fan-cowl is accompanied with a notably different location of the shock at the leading edge of the nacelle for different M_∞ . The presence of the wing pressure-field shifted the shock downstream by $\Delta x/L_{nac}$ of +0.0, +0.02, +0.05 and +0.1 for the M_∞ of 0.8, 0.82, 0.84 and 0.86, respectively. As a result, the beneficial interference on the nacelle reduced with the increase of Mach number. The relatively constant effect of the buoyancy force at the afterbody was opposed by a significant shift of the shock at the leading edge. The reduction of the extent of suction prior to the

shock location resulted in a notable decrease of the forward-oriented suction force that minimised the nacelle modified drag ($C_{D^* nac}$). As a result, the $C_{D^* nac}$ for the wing-installed nacelle increased faster with an increase of M_∞ as compared with the isolated engine (Figure 6.41).

The effect of engine installation on the thrust was quantified and its sensitivity to the flight Mach number (M_∞) was analysed based on the thrust coefficient (C_V ; Figure 6.44). To maintain the constant intake mass flow capture ratio (MFCR), the engine power setting was increased with an increase of M_∞ . From the $M_\infty = 0.7$ to 0.86, the TET increased fairly linearly from 1480K to 1745K. The change in the engine power setting resulted in a linear increase in FNPR from 1.8 to 2.27 and a quadratic increase in CNPR from 1.37 to 1.84. The increase of nozzle pressure ratios results in the steady increase of C_V from 0.979 at $M_\infty = 0.7$ to 0.980 at $M_\infty = 0.86$. For the cases over the entire range of Mach numbers considered, the presence of the wing reduced the value of C_V by approximately -0.001. The reduction in the thrust coefficient was a result of the overlap of the engine exhaust system with the wing and the reduction of the pressure force at the core-cowl of the engine due to the chosen engine position C3. The effect appeared to be independent of the flight Mach number.

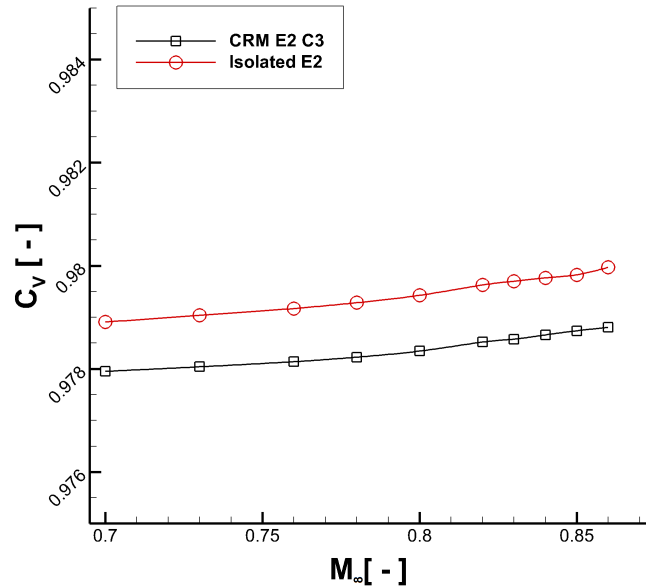


FIGURE 6.44: Thrust coefficient (C_V) for E2 engine in isolation and in configuration C3 under the wing of CRM as a function of flight Mach number (M_∞); $AoA_{A/C} = 2.5^\circ$

To sum up, the CRM configurations with and without an engine were tested for a range of flight Mach numbers. The presence of installation causes a reduction of local angle of attack for the wing and reduces the wing loading. As compared with the clean-wing configuration the drag rise for the airframe is substantially delayed, however, it is partially offset by the increase of the drag on the nacelle. Overall, the drag rise Mach number of $M_{DR} = 0.83$ and $M_{DR} = 0.833$ for the configuration without and with the engine was determined. Moreover, the sensitivity of the effects of engine installation on

drag, lift and thrust to the flight Mach number was quantified. The installation drag at constant angle of attack ($C_{D \text{ } inst|AoA}$) was broadly constant at $C_{D \text{ } inst|AoA} \approx 36dc$ for a range of M_∞ below 0.83. A further increase of M_∞ resulted in a decrease of $C_{D \text{ } inst|AoA}$ to zero at $M_\infty = 0.86$. However, the presence of installation caused the loss of lift, which has to be compensated and it was a source of the lift induced drag. A typical loss of lift $\Delta C_{L \text{ } tot} = -0.02$ was observed for the cases with M_∞ below 0.83. A constant reduction in C_V by -0.001 over the entire range of flight Mach numbers was observed and it was caused primarily by the overlap of the exhaust system with the wing. Overall, the choice of the cruise Mach number in the study at $M_\infty = 0.82$ is justified to avoid the negative effect of the drag rise and credibly determine the effects of engine installation on drag, lift and thrust. The rather constant effect of engine installation over the range of Mach numbers from 0.7 to 0.83 allows for a potential extension of the mission analysis to other flight Mach numbers based on the existing database.

6.8 The effect of engine power setting

A next essential step to determine the effects of engine installation within the cruise phase of flight is to determine the sensitivity of those effects to the engine power settings. Two engines, E1 and E2, in the baseline position (C3) were considered to determine the sensitivity of the overall installation effect. The engine power setting was modified in the performance model by the prescription of a range of turbine entry temperatures (TET ; Section 6.2). Furthermore, the aerodynamic analysis was performed and it was recognised that there were 3 key parameters (Table 6.2 in Section 6.2) that dominated the aerodynamics, mass flow capture ratio ($MFCR$), fan nozzle pressure ratio ($FNPR$) and core nozzle pressure ratio ($CNPR$).

For the baseline engine E1, the total aircraft drag ($C_{D\ tot}$) and overall vertical force ($C_{Z'}$) polar was computed (Figure 6.45). The reduction of engine power setting reduced the massflow that enters the engine and as a result the intake massflow capture ratio ($MFCR$) of the engine changed. The $MFCR$ gives an indication of intake spillage characteristic. Even though, the significant change in $MFCR$ is considered from 0.75 to 0.6, the $C_{D\ tot}$ increased by only 3dc over the entire range of $C_{Z'}$. To explain the rather modest effect in the drag domain, the decomposition of the $C_{D\ tot}$ was considered. The modified nacelle drag ($C_{D^*\ nac}$) of a wing-installed engine was compared with the isolated engine. For the isolated engine, the effect of spillage was modestly sensitive to the engine incidence ($AoA_{eng,ideal}$) (Figure 6.46). The trend of the $C_{D^*\ nac}$ for an isolated engine was not affected by the reduction in power setting and the $C_{D^*\ nac}$ increased by 6dc at $AoA_{eng,ideal} = 1.75^\circ$ and 11dc at $AoA_{eng,ideal} = 7^\circ$ over the range of power settings.

Clearly the under-wing installation of the engine had a significant effect on the drag ($C_{D^*\ nac}$) characteristics of the engine (Figure 6.46). The $C_{D^*\ nac}$ increased from 36dc for an isolated engine in PS1 at $AoA_{eng,ideal} = 1.75^\circ$ to 45dc when installed. However, as the aircraft incidence increased, the wing pressure-field reduced the $C_{D^*\ nac}$ of a wing-installed engine. For example, at $AoA_{eng,ideal} = 7^\circ$ the $C_{D^*\ nac}$ reduced from 40 dc for an isolated engine in PS1 to only 19 dc when installed.

It is important to determine if the spillage drag and the sensitivity to engine power setting depend on the engine installation. To compare with the isolated engine, the power setting was reduced for the engine under the wing. The reduction of the massflow capture ratio from 0.75 to 0.6 for the engine power settings from PS1 to PS4 in the baseline position (C3) resulted in an increase of the nacelle modified drag ($C_{D^*\ nac}$) by roughly +7 dc at $AoA_{eng,ideal} = 1.75^\circ$ (Figure 6.45), as compared with a +6dc increase for the isolated engine over the same range of power settings. Moreover, the increase of $\Delta C_{D^*\ nac}$ due to a reduction in power setting for the wing-installed engine was moderately dependent on the aircraft incidence (Figure 6.46). At $AoA_{eng,ideal} =$

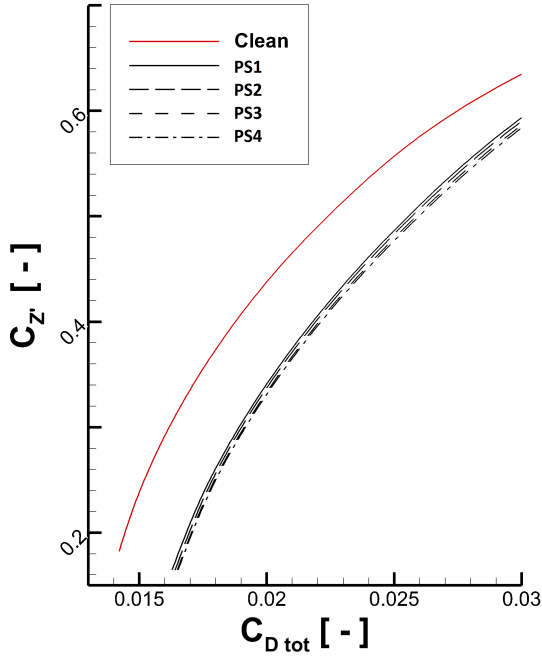


FIGURE 6.45: Lift-drag polar for E1 engine in position C3 in four engine power settings from PS1 to PS4; $Re = 45 \times 10^6$; $M_\infty = 0.82$

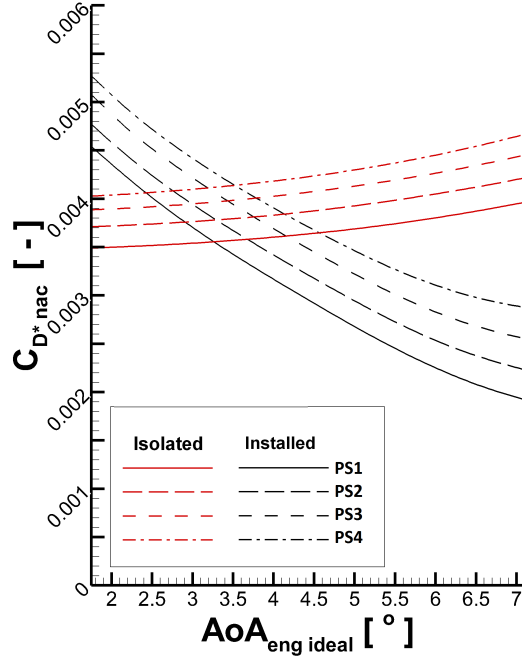


FIGURE 6.46: Modified engine drag ($C_{D^* nac}$) as function of engine ideal incidence ($AoA_{eng,ideal}$) for the E1 engine in isolation and under the wing in baseline position (C3) for a range of power settings from PS1 to PS4; $Re = 45 \times 10^6$; $M_\infty = 0.82$

6.75° the $\Delta C_{D^* nac}$ for the wing-installed engine equals +12dc, as compared with the isolated engine over the entire range of power settings. It was suggested that the spillage characteristic was broadly unaffected by the installation due to the position of the intake outside of the wing pressure-field. However, the overall trend of the $C_{D^* nac}$ was a combination of the effect of the wing pressure field and the spillage characteristics of an intake. Even though the sensitivity of the nacelle drag ($C_{D^* nac}$) to the engine power setting ranged from +7 to +12dc, the sensitivity of the total aircraft drag ($C_{D tot}$) across the same range of power settings was +3 to +5 dc. It was explained as a result of favourable aerodynamic effects on the airframe. The beneficial effect on the airframe was predominantly a result of the reduced relative positive pressure on the lower side of the wing and in consequence the reduction of pressure drag. Given a typical change of an engine power setting during a long-haul flight in range of $\Delta MFCR = 0.05$, the overall sensitivity in the drag domain equals approximately +1dc.

The effect of the engine power setting, and associated MFCR, on the pressure distribution was localised to the fan-cowl forebody (Figure 6.47a). The reduction of power setting caused a reduction in intake MFCR and increased the spillage drag. As a result the peak suction at $\xi/L_{nac} = 0.05$ increased by approximately $\Delta C_p = -0.08$ every $\Delta MFCR = 0.05$. The effect of the wing on the central and aft section of the fan-cowl was broadly constant for all power settings and as a result almost no difference was

observed beyond $\xi/L_{nac} > 0.3$. The aerodynamics on the suction side of the wing were indifferent to the changes of engine power setting from PS1 to PS4 (Figure 6.47b) and a constant reduction in peak suction $\Delta C_p = +0.25$ was observed for all power settings as compared with clean-wing configuration. However, a modest difference in the pressure distribution was observed on the lower side of the wing. The lowest power setting (PS4) had single suction peak $C_p = -0.17$ on the pressure side at $x/c = 0.55$. By increasing the power setting to PS1, the suction peak at $x/c = 0.55$ increases by $\Delta C_p = -0.03$, also a local pressure increased by $\Delta C_p = +0.03$ at $x/c = 0.3$.

For the completeness, the E2 engine was tested over a range of power settings (Table 6.2 in Section 6.2). As compared with the E1 engine, broadly similar effects of the variation in engine power setting were observed for the larger engine (E2) (Figure 6.48). As for the baseline engine (E1), the key effect of the engine power setting on the E2 engine is due to the increase of the intake spillage. The presence of the wing affected the peak suction at the leading edge of the fan-cowl $\xi/L_{nac} = 0.05$ by approximately $\Delta C_p = -0.03$ at every power setting considered as compared with the isolated E2 engine (Figure 6.48a). A similar effect was observed for the E1 engine. In the cases of both engines, it was explained by the presence of the upwash that increased the local engine incidence. Furthermore, the upper side of the wing was insensitive to the engine power setting (Figure 6.48b) similarly as for the baseline engine (E1). Moreover, a modest sensitivity of C_p distribution is observed at $x/c = 0.55$, where $\Delta C_p = +0.05$ was caused by a reduction of engine power setting from PS1 to PS4. Even though, the changes to the C_p are modest, the integration of the pressure force over the large wing area results in modest benefit in airframe interference that opposed the increased spillage drag of the engine.

As compared with the isolated engine, the installation of the engine under the wing affected the pressure distribution for the core-cowl (Figure 6.49). The isolated engine at PS1 exhibited two distinct zones on the core-cowl, one with expansion zone ($0.1 < \xi/L_{core} < 0.5$) and one compression zone ($0.5 < \xi/L_{core} < 0.9$). As the engine was installed under the wing (Figure 6.49), the peak of the suction and peak of the pressure reduced by $\Delta C_p = +0.1$ at $\xi/L_{core} = 0.25$ and $\Delta C_p = -0.2$ at $\xi/L_{core} = 0.7$, as a result of the wing pressure-field. Similarly, as for the fan-cowl, the buoyancy force that acted on the core-cowl resulted in a beneficial forward oriented force. Moreover, the reduction in the power setting from PS1 to PS4 led to reduced $FNPR$ (Table 6.2). As a result the flow physics changed and a constant constant value of $C_p = +0.25$ along the entire core-cowl length (Figure 6.49) was reported. To quantify the effects of the wing pressure-field, the modified gross propulsive force (GPF^*) for E1 engine in four power settings was assessed. The GPF^* for each power setting was non-dimensionalised by the GPF_{ref}^* (Figure 6.50) for the respective power setting of an isolated engine (Table 6.2) at $AoA_{eng,ideal} = 0^\circ$. The effect of upwash was neglected, as a typical $AoA_{upwash} = +0.5^\circ$ corresponds to a difference of only 0.02% of the GPF^* of an isolated engine. As for

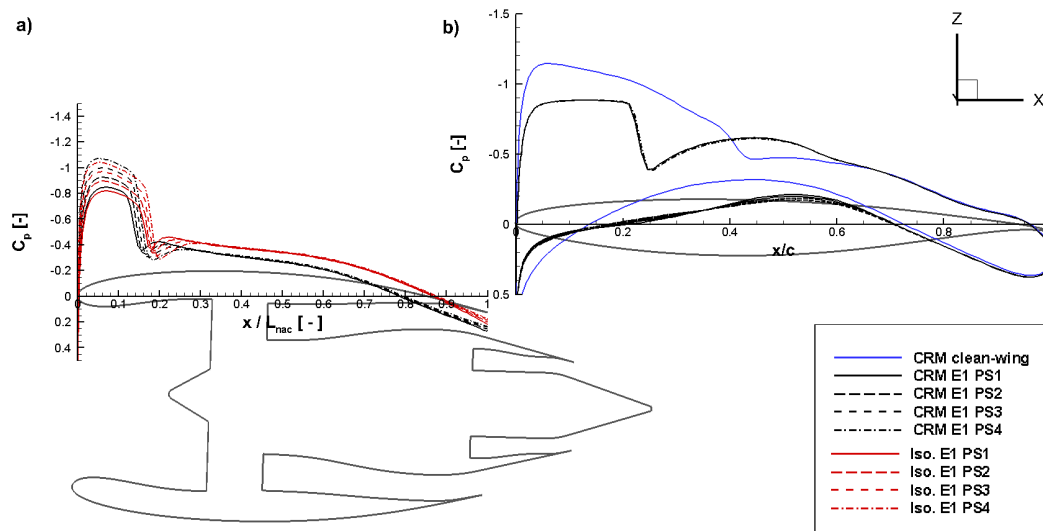


FIGURE 6.47: a) Pressure distribution (C_p) on the top line of the fan-cowl for E1 engine in baseline position (C3) in four engine power settings PS1 to PS4 at $AoA_{eng,ideal} = 4.25^\circ$ ($AoA_{A/C} = 2.5^\circ$), compared with the isolated E1 engine at $AoA_{eng,ideal} = 4.25^\circ$; b) Pressure distribution (C_p) on the wing at $y/(b/2)=0.3$ in presence of E1 engine in C3 position for PS1 to PS4 compared with the CRM clean-wing; $Re = 45 \times 10^6$; $M_\infty = 0.82$;

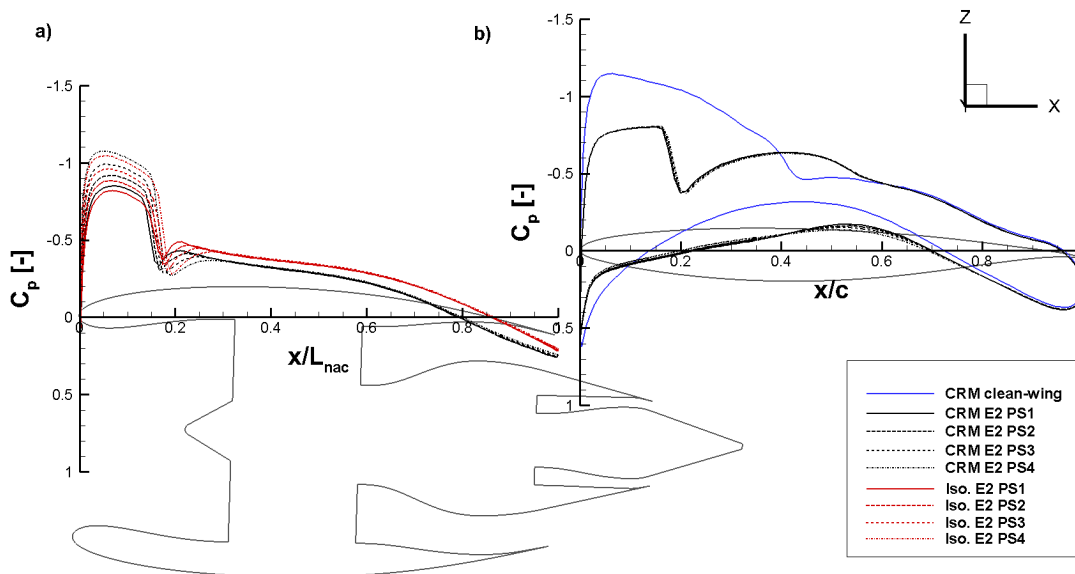


FIGURE 6.48: a) Pressure distribution (C_p) on the top line of the fan-cowl for E2 engine in baseline position (C3) in four engine power settings PS1 to PS4 at $AoA_{eng,ideal} = 4.25^\circ$ ($AoA_{A/C} = 2.5^\circ$), compared with isolated E2 engine at $AoA_{eng,ideal} = 4.25^\circ$; b) Pressure distribution (C_p) on the wing at $y/(b/2)=0.3$ in presence of E2 engine in C3 position for PS1 to PS4 compared with the CRM clean-wing; $Re = 45 \times 10^6$; $M_\infty = 0.82$;

the isolated engine (Figure 6.50), the sensitivity of GPF^* to $AoA_{eng,ideal}$ depended on the engine power setting and was not monotonic. No clear relation was found as the effect depended on the non-linear interaction of the shock waves in the jet with the wing pressure-field. Overall, a broad trend for all power settings was defined as the GPF^* reduced by 0.5% over the entire range of incidence. As compared with the isolated engine, the wing-installed engine had a substantially different trend. A loss of GPF^* by 1.75% was observed for PS1 at $AoA_{eng,ideal} = 1.75^\circ$ and as the $AoA_{eng,ideal}$ increased, a benefit of +1% in GPF^* was observed at $AoA_{eng,ideal} = 7^\circ$. As the engine power setting of an installed engine reduces from PS1 to PS4, a non-monotonic performance was noted. It is important to realise, that the wing pressure-field can notably increase the GPF^* by +0.25% at typical cruise conditions. The consequence of this finding is, that the core-cowl and plug can be shaped to maximise the benefit that arises from the wing by increasing the projected area in the thrust direction.

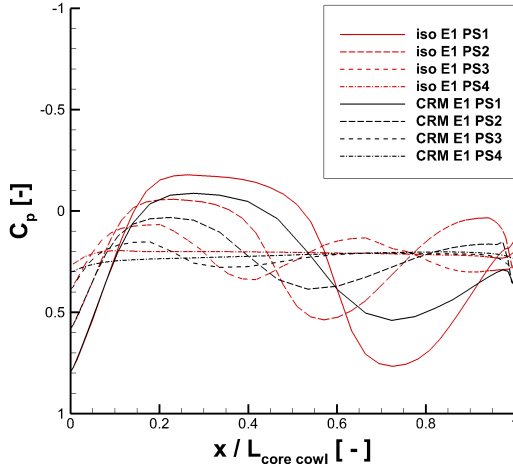


FIGURE 6.49: Pressure distribution (C_p) on the top line of the core-cowl for E1 engine in position C3 for engine power settings from PS1 to PS4 at $AoA_{eng,ideal} = 4.25^\circ$ ($AoA_{eng} = 2.5^\circ$), compared with isolated engine at $AoA_{eng,ideal} = 4.25^\circ$; $Re = 45 \times 10^6$; $M_\infty = 0.82$;

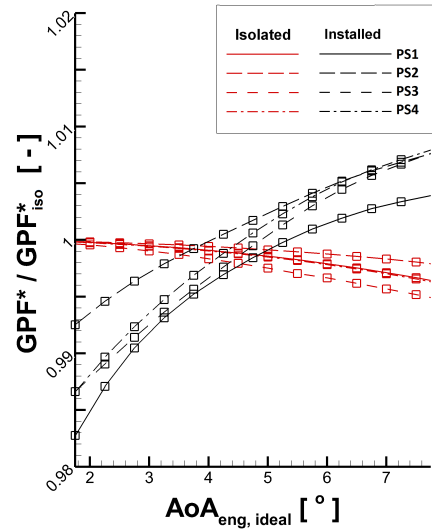


FIGURE 6.50: Non-dimensionalised modified gross propulsive force (GPF^*/GPF_{ref}^*) for E1 engine in position C3 (C3) in four engine power settings compared with isolated engine; $Re = 45 \times 10^6$; $M_\infty = 0.82$

To quantify the overall installation effect for the range of power settings, an installation force diagram (Figure 6.51) for the baseline engine E1 was created. The cruise operating point was chosen with the coefficient for the overall vertical force of ($C_{Z'} = 0.5$ at $M_\infty = 0.82$). The installation effects were considered for a re-trimmed aircraft and were referenced to the value of standard net thrust of the baseline engine E1 at cruise condition ($F_{N ref}$ at PS1) (Figure 6.51). The effect of the overall installation force was presented as relative to the respective standard net thrust at cruise ($F_{N iso|C_{Z'}=0.5}$) and therefore the force diagram for all power settings starts at zero. The beneficial thrust terms ($\theta_{exhaust}$) reduced with a reduction of the power setting from +11% $F_{N ref}$ for

PS1 to +8.8% $F_{N \text{ ref}}$ for PS4, as a result of the reduction in nozzle pressure ratios (FNPR and CNPR). The detrimental effect of modified nacelle drag (D_{nac}^*) on the overall installation force (F_{inst}) increased. The D_{nac}^* increased from -14.3% $F_{N \text{ ref}}$ for PS1 to -16.6% $F_{N \text{ ref}}$ for PS4, as a result of an increase in the intake spillage drag. As a consequence, the reduction of the power setting causes a difference in isolated NPF of -4.6% $F_{N \text{ ref}}$ for PS4 as compared with PS1. To include the interference terms due to the installation onto the aircraft, the interference for thrust (ΔF_N^*), modified nacelle drag (ΔD_{nac}^*) and airframe drag ($\Delta D_{A/F}$) were considered (Figure 6.51). Considering the overall interference effect (F_{interf}), a benefit of +1.2% of $F_{N \text{ ref}}$ for PS1 increased to +2.4% of $F_{N \text{ ref}}$ for PS4. The improvement in overall F_{interf} between the power settings was the outcome of a reduction of the FNPR and improvement of thrust (ΔF_N^*) and airframe ($\Delta D_{A/F}$) interference. A modestly detrimental effect in ΔF_N^* for PS1 of -1% $F_{N \text{ ref}}$ became neutral for PS4. However, the change was non-monotonic. The reduction of power setting had an detrimental effect on the nacelle interference and ΔD_{nac}^* reduced from +2.6% $F_{N \text{ ref}}$ for PS1 to +1.7% $F_{N \text{ ref}}$ for PS4. The detriments were partially offset by an improvement of $\Delta D_{A/F}$ from -0.3% $F_{N \text{ ref}}$ for PS1 to +1% $F_{N \text{ ref}}$ for PS4. Overall, the overall installation force (F_{inst}) was sensitive to engine power setting by 3.4% $F_{N \text{ ref}}$.

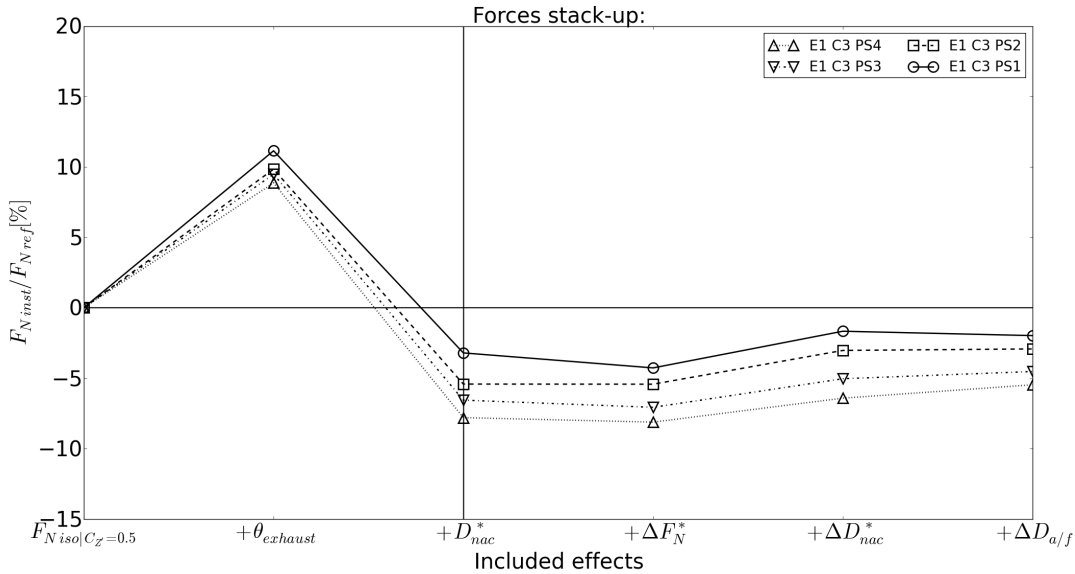


FIGURE 6.51: The overall installation force diagram for engine E1 in four power settings from PS1 to PS4; standard net thrust of isolated engine at cruise ($F_{N \text{ iso}}|_{C_{Z'}=0.5}$) plus the effect of exhaust forces ($+\theta_{\text{exhaust}}$), nacelle drag ($+D_{nac}^*$), and detailed interference effects ($\Delta F_N^* / \Delta D_{nac}^* / \Delta D_{A/F}$); $Re = 45 \times 10^6$; $M = 0.82$; $C_{Z'} = 0.5$

Similarly, to quantify the overall installation effect for the range of power settings for the larger engine E2, an installation force diagram was considered (Figure 6.52). The installation effects were also referenced to the value of standard net thrust of the baseline engine E1 at cruise condition ($F_{N \text{ ref}}$ at PS1) (Figure 6.52). The beneficial effect of θ_{exhaust} reduced with a decrease of power setting from +18% of $F_{N \text{ ref}}$ for PS1 to +13%

of $F_{N \text{ ref}}$ for PS4. The benefits in the thrust domain were offset by a detrimental effect of the nacelle drag (D_{nac}^*). The increase of drag (D_{nac}^*) for an isolated E2 engine over a range of power settings from PS1 to PS4 was -1.8% $F_{N \text{ ref}}$, as compared with a drag increase of -2.3% $F_{N \text{ ref}}$ for the E1 engine over the same range of power settings. The net propulsive force (NPF) of an isolated engine (Figure 6.52) for the larger E2 engine was more sensitive for the same range of massflow capture ratios (PS1 to PS4) with -6.7% $F_{N \text{ ref}}$, as compared with the detrimental effect of -4.6% of $F_{N \text{ ref}}$ for E1 over the power settings from PS1 to PS4. However, as considered over the same range of engine thrust from 100% to approximately 40% of $F_{N \text{ ref}}$, the large engine was less sensitive with -3.4% (PS1 to PS3; Figure 6.52) as compared with the -4.6% of $F_{N \text{ ref}}$ (PS1 to PS4) for E1. The installation of the E2 engine under the wing resulted in the overall beneficial interference (F_{interf}) and the interference effect was sensitive to the power setting. The F_{interf} improved from $+2.77\%$ $F_{N \text{ ref}}$ for PS1 to 3.0% $F_{N \text{ ref}}$ at PS4. The overall effect was the sum of component interference. The ΔF_N^* improved from -1.25% $F_{N \text{ ref}}$ at to -0.95% $F_{N \text{ ref}}$ at PS4. The ΔD_{nac}^* worsened from $+3.1\%$ $F_{N \text{ ref}}$ at PS1 to $+2.01\%$ $F_{N \text{ ref}}$ at PS4. The dominant effect appeared on the airframe and the $\Delta D_{A/F}$ improved from $+0.95\%$ $F_{N \text{ ref}}$ at PS1 to $+1.83\%$ $F_{N \text{ ref}}$ at PS4. Overall, the installation effect (F_{inst}) for the E2 engine was more penalising by -0.1% $F_{N \text{ ref}}$ over the same thrust range from $100\%F_{N \text{ ref}}$ to 40% $F_{N \text{ ref}}$. As a consequence, the larger engine was more sensitive to engine power settings and notable changes in the operating massflow capture ratio should be avoided to minimise the spillage drag.

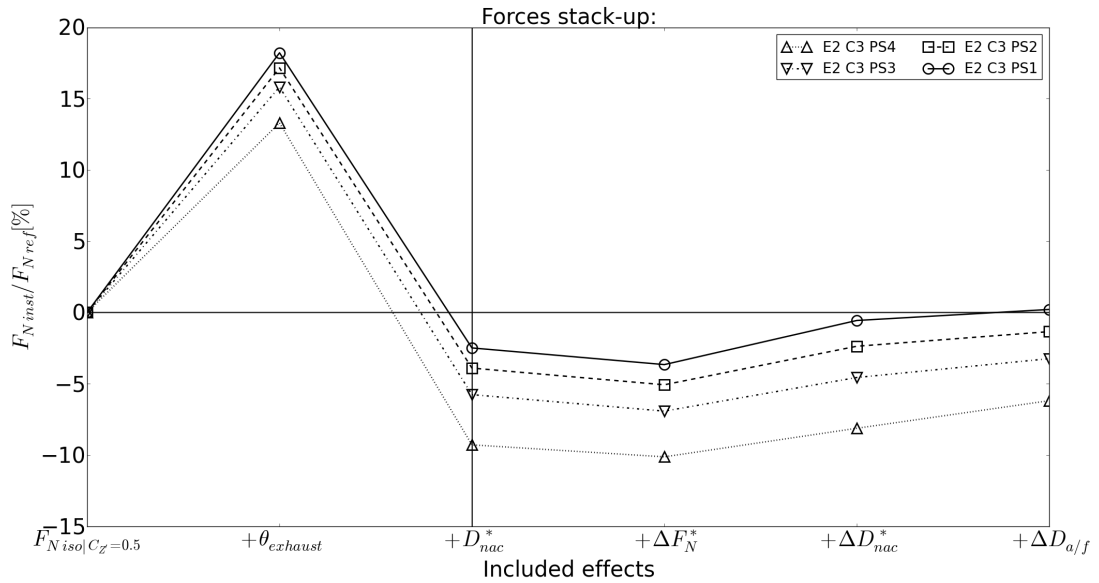


FIGURE 6.52: The overall installation force diagram for engine E2 in four power settings from PS1 to PS4; standard net thrust of isolated engine at cruise ($F_{N \text{ iso}}|_{Cz'=0.5}$) plus the effect of exhaust forces ($+\theta_{exhaust}$), nacelle drag ($+D_{nac}^*$), and detailed interference effects ($\Delta F_N^* / \Delta D_{nac}^* / \Delta D_{A/F}$); $Re = 45 \times 10^6$; $M = 0.82$; $C_{Z'} = 0.5$

6.9 Cruise fuel burn analysis

To stress the significance of the aerodynamic results, an aircraft performance calculation for a 300 seater aircraft was completed to quantify the impact on cruise fuel burn over the range of 10000 km. Based on the clean-wing airframe drag polar and the overall installation effect (F_{inst}) for the considered engines, the thrust requirement for the engines is defined. The differences in F_{inst} for the engine-airframe configurations from the previous section are then converted to the fuel burn with the use of the point mass model of the aircraft (Section 4.1). A single point calculation at the start of cruise ‘SoC’ condition was initially used for the baseline engine with fixed engine power setting to isolate the effect of aerodynamic forces on the fuel flow. For the E1 configuration the ‘SoC’ condition was evaluated to be at $C_{Z'} = 0.47$ and MFCR=0.75 for a fully balanced aircraft at $M_\infty = 0.82$ at an altitude of 10668 m (35000 ft) (Section 4.1.3). To quantify the effect of engine position in terms of fuel consumption, the engine fuel flow was quantified (Figure 6.53) based on the thrust requirement for the balanced aircraft for each engine position. For the baseline engine E1, the spatial sensitivity of aerodynamic forces to different engine position resulted in a greater thrust requirement for some of the engine positions as compared with the baseline position C3. As a result, the fuel consumption ($\dot{m}_{fuel}$) of the engine rose by approximately +1.6% in the worst position A2 as compared with the best installation position C3.

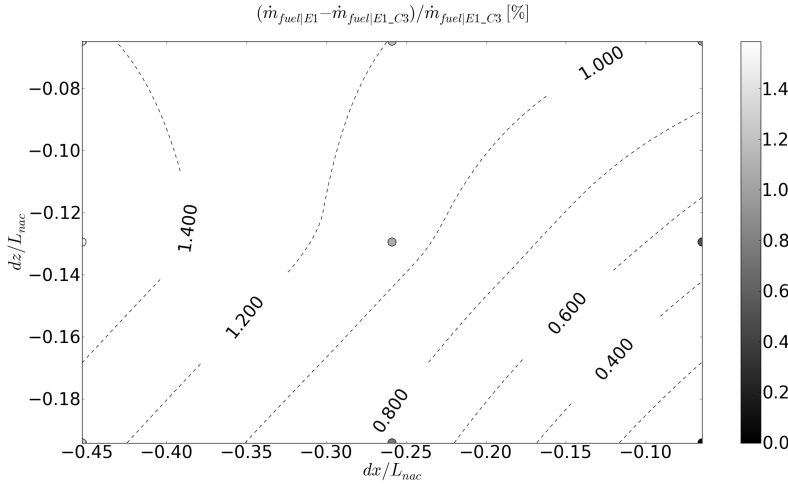


FIGURE 6.53: Spatial sensitivity map of the fuel consumption ($\dot{m}_{fuel}$) for the baseline engine E1 as compared with the baseline position C3; $Re = 45 \times 10^6$; $M_\infty = 0.82$; ($C_{Z'}$) = 0.47 for engine cruise power setting (MFCR=0.75, FNPR=2.71)

To compare the fuel consumption between the two engines the cycle performance at the ‘SoC’ design point was analysed. For the same thrust requirement for both engines, the benefits in $\dot{m}_{fuel}$ for E2 engine were about -5.8% based on the cycle analysis (Table 6.1) as compared with E1 engine. The single point calculation for a balanced aircraft at ‘SoC’ with use of the computed aerodynamic installation effects and the weight penalty

for the larger engine resulted in benefits in $\dot{m}_{fuel}$ ranging from -5.6% to -6.6% for E2 as compared with E1 at a corresponding position. The increase of aircraft weight for E2 configurations resulted in a greater lift requirement, and as a consequence greater drag and thrust requirement for the engine. Overall, the detrimental effect of weight was in order of +2% increase in $\dot{m}_{fuel}$ that partially offset the benefits from the cycle improvements and aerodynamics. As a result a spatial sensitivity map for the fuel consumption at the start of cruise conditions was created for the E2 as compared with E1 at the corresponding position and referenced to the $\dot{m}_{fuel}$ of E1 at C3 (Figure 6.54). The thermodynamic cycle improvements, aerodynamic differences and weight penalty were included (Figure 6.54). The -5.6% benefit in $\dot{m}_{fuel}$ for E2 engine as compared with E1 was observed at position C3. Due to different spatial sensitivities for the two engines, the benefit for the large engine E2 increases to a -6.6% reduction in fuel flow rate at position A2.

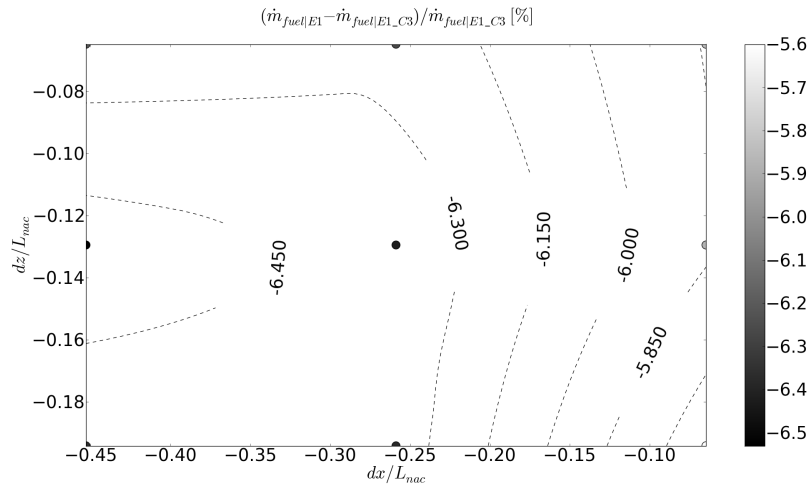


FIGURE 6.54: Spatial sensitivity map of the fuel consumption ($\dot{m}_{fuel}$) for the baseline engine E2 as compared with the baseline engine E1 in respective position; $Re = 45 \times 10^6$; $M_\infty = 0.82$; both engine-aircraft configuration at ‘SoC’ power setting and with balanced aircraft

To compute the integrated fuel consumption over the entire cruise phase, the throttle dependent aerodynamic effects were considered as explained in Section 4.1.3. A range of engine power settings for E1 and E2 engines in configuration C3 was considered, to cover the cruise phase from the start of cruise (‘SoC’) through the end of cruise (‘EoC’), as well as to give an indication about the effects at the start of descent (‘SoDe’). Furthermore, to determine the performance of the engine over the entire cruise phase of flight, the relation between the overall installation force (F_{inst}) and the engine power setting and lift coefficient was considered. Based on the computations for engine E1 in the baseline position C3, a sensitivity map of the overall installation force (F_{inst}) as a percentage of reference thrust (F_{N_ref}) was created (Figure 6.55a) to aid the cruise fuel burn analysis. The map covers the range of $C_{Z'}$ and range of engine power settings represented by the intake massflow capture ratio (MFCR). For the baseline engine E1,

the installation penalty (F_{inst}) increased for aircraft off-design conditions away from $C_{Z'} = 0.5$ and $MFCR=0.75$. The dominant trend was the increase of the installation penalty with a reduction in $MFCR$, as a result of increased spillage. The secondary effect observed was the jet interaction with the flow acceleration over the lower side of the wing at $MFCR=0.75$ and low lift coefficient ($C_{Z'} = 0.2$). At $C_{Z'} = 0.2$, the reduction of power setting initially alleviated the jet interaction for the $MFCR=0.7$, however, further reduction of power setting was dominated by the increasing spillage drag. Overall, over the entire range of conditions considered, the sensitivity to engine power setting and aircraft lift coefficient was in order of 3% of $F_{N\ ref}$.

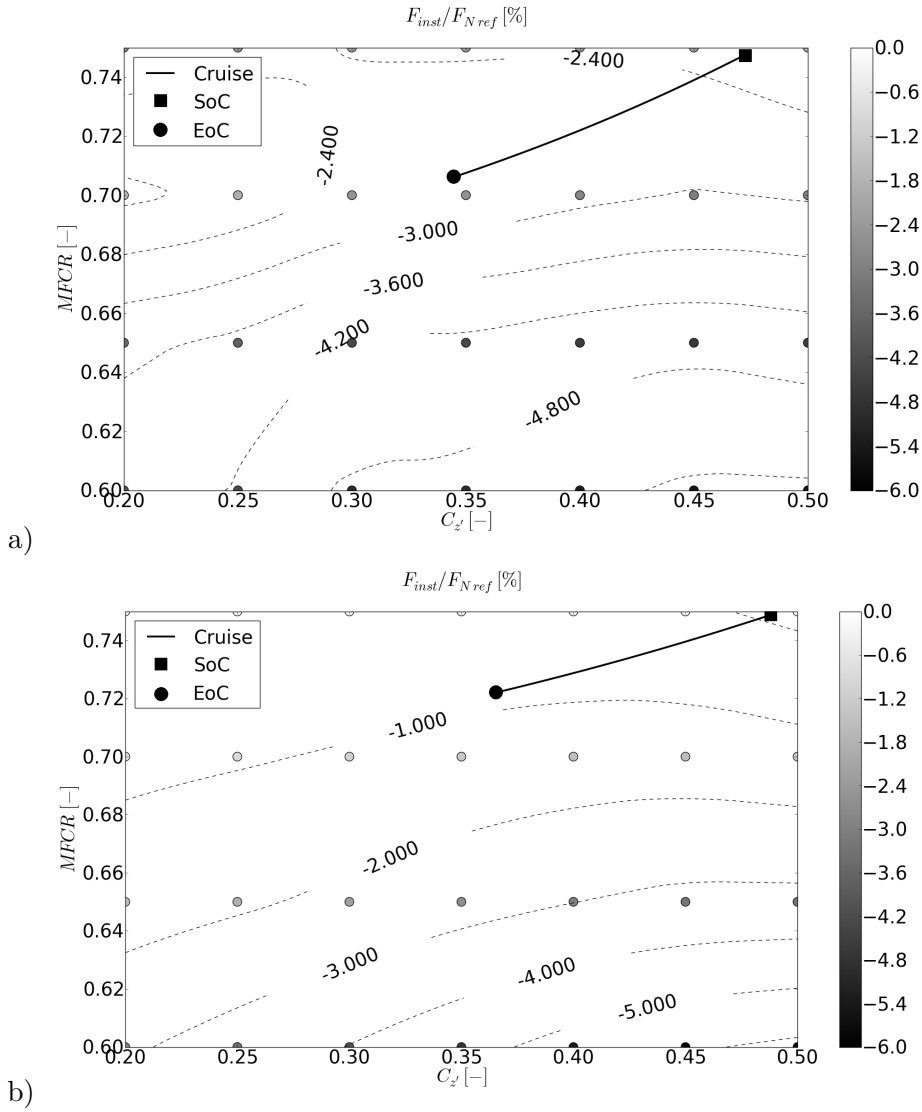


FIGURE 6.55: Overall installation effect (F_{inst}) as percentage of reference thrust ($F_{N\ ref}$) for a) baseline engine E1 and b) E2 engine over the range of vertical force coefficients coefficients ($C_{Z'}$) and range of power settings represented by MFCR; $Re = 45 \times 10^6$; $M_\infty = 0.82$;

Based on the computed relation between the overall installation force (F_{inst}) and the engine power setting, the integrated cruise fuel burn calculation was considered with the

weight penalty for large engine included. For the E1 configuration the ‘SoC’ condition was evaluated to be at $C_{Z'} = 0.47$ and $MFCR=0.75$ for a fully balanced aircraft at $M_\infty = 0.82$ at an altitude of 10668 m (35000 ft), according to the methodology (Section 4.1.3). As a result, the end of cruise ‘EoC’ point was established for the E1 configuration at $C_{Z'} = 0.34$ and $MFCR=0.7$. The integrated fuel burn for the cruise phase was evaluated. A similar off-design performance map was created for the large engine E2 (Figure 6.55b). The large engine exhibited a more monotonic behaviour with regards to power setting and lift coefficient as compared with E1 engine. The design of the engine for the low specific thrust resulted in more favourable nozzle pressure ratios and the jet interference effects at the high power setting. Over the flight envelope, the effect of the engine power setting on the overall installation penalty for the E2 engine was approximately twice as sensitive as the effect of power setting for the E1 engine and it was of the order of 1% of $F_{N \text{ ref}}$. Based on the off-design map for the overall installation effect (F_{inst}) as a function of the engine power setting and aircraft lift coefficient, the integrated cruise fuel burn calculation was considered. For the E2 configuration the ‘SoC’ condition was evaluated to be at $C_{Z'} = 0.49$ and $MFCR=0.75$ for a fully balanced aircraft. The greater value of the $C_{Z'}$ was the result of the increased weight of the aircraft for the E2 configuration as compared with the E1 configuration. As a result, the end of cruise ‘EoC’ point was established for the E1 configuration at $C_{Z'} = 0.36$ and $MFCR=0.72$. The integrated fuel burn for the cruise phase was evaluated and as compared with the E1 engine the E2 engine consumed -4.8% less fuel over the same mission. As a consequence, the estimated benefits from the thermodynamic cycle analysis in the order of -5.8% reduced to -4.8% fuel burn reduction in favour of the large E2 engine taking into account the effects of throttle dependent aerodynamic effects, installation terms and the engine weight penalty that were considered over a long-haul mission. The beneficial aerodynamic terms that were noted for the E2 engine were fully opposed by the penalty due to the engine weight. Overall, it is better to use the large engine and the potential savings over the cruise flight are -4.8% of total fuel burn.

6.10 Conclusions

The following conclusions arose for the numerical studies on the effects of installation for the under-wing turbofan engines:

- The engine installation affected the lift of the aircraft, the total drag of the aircraft and the thrust of the wing-installed engine. An overall installation force (F_{inst}) was proposed to address all the effects of installation for a re-trimmed aircraft configuration at an overall vertical force coefficient ($C_{Z'}$) that included the vertical component of thrust. Moreover, the component breakdown of F_{inst} was proposed, which outlined the contribution of the interference on airframe drag ($\Delta D_{A/F}$), interference on nacelle drag (ΔD_{nac}^*), and interference on engine thrust (ΔF_N^*)

- Considering the baseline engine, the movement of the engine closer to the wing vertically or upstream axially was detrimental as compared with the baseline position (C3). The sensitivity of F_{inst} to the position was approximately -1.7% of reference thrust relative to position C3.
- For the E2 engine, there were some notable differences in the interference terms which depended on the engine position and overall the F_{inst} varied by -1.3% $F_{N ref}$ across the full range of positions, as compared with E2 engine in position C3. This was less than the range of 1.7% which was calculated for the smaller E1 engine.
- As compared with the baseline (E1) engine, the large engine (E2) required from -2.2% to -3.2% $F_{N ref}$ less standard net thrust ($F_{N ref}$) to compensate for the overall installation penalties (F_{inst}) over the range of considered positions. The large penalties of the modified drag of the nacelle (D_{nac}^*) increased proportionally to the engine size, however, they were offset by the better interference terms and the beneficial effect on the exhaust term ($+\theta_{exhaust}$) due to the design of the core-cowl and plug as well as the favourable lower FNPR. The modified thrust requirement (F_N^*) for the larger engine was approximately +4.6% $F_{N ref}$ greater for the larger engine, as compared the baseline engine. A significantly greater attention needs to be dedicated to the evaluation of the $\theta_{exhaust}$ forces.
- The sensitivity of the overall installation force (F_{inst}) to engine power setting was assessed for E1. The overall sensitivity was 3.4% and it was primarily determined by an increase of spillage drag of the intake due to a reduction in mass flow capture ratio, and the reduction of beneficial pressure force on the exhaust due to reduction in nozzle pressure ratio. The effects of aerodynamic interference were of secondary magnitude and predominantly depended on the favourable nozzle pressure ratio of the fan nozzle.
- As for the larger engine E2, the installation effect (F_{inst}) depended with 3.5% $F_{N ref}$ over the same thrust range as E1 engine with 3.4% $F_{N ref}$. As a consequence, the larger engine was more sensitive to engine power settings and notable changes in the operating massflow capture ratio should be avoided to minimise the spillage drag.
- The estimated benefits from the thermodynamic cycle analysis were in the order of -5.8% less in fuel consumption for the larger E2 engine as compared with the baseline E1 engine. The integrated cruise fuel burn that took into account the effects of throttle dependent aerodynamic effects, installation terms and the engine weight penalty that were considered over a long-haul mission reduced the fuel burn reduction of -5.8% to -4.8% in favour of the large E2 engine.

CHAPTER 7

Discussion and synthesis

“If particulars are to have meaning,
there must be universals.”

Plato

This chapter describes a more generalised model proposed for the determination of the engine installation effects. Due to the limited dataset, the model was aimed to outline the general trend rather than quantify the detailed changes in the effects of engine installation. The author’s intention is to propose an approach for the evaluation of the effects of engine installation that can be improved with time by computation of extensive database. Based on the range of sensitivities determined in the Chapter 6, a reduced model for engine position, size and power setting was proposed. The installation drag (X_{inst}) was considered to include the drag of isolated nacelle and the sum of interference on nacelle drag, airframe drag and thrust. The submodels based on the sensitivity to engine size (Section 7.1), engine power setting (Section 7.2) were proposed and a basic verification was completed based on the available dataset (Section 7.5). Furthermore, the submodel for the engine position (Section 7.3) was completed. Overall, the method to calculate the installation drag (X_{inst}) was proposed based on the superposition of individual sensitivities (Section 7.4).

7.1 Model for engine size

As shown in Section 6.6 and Appendix E.3.1, the size of the engine significantly affected the magnitude of the engine installation effect. Therefore, the size factors ($\kappa_{\sigma|i}$) were a function (Equation 7.1) of the ratio of engine fan diameter (D_{FF}) and the fan diameter of the baseline engine (D_{base})

$$\kappa_{\sigma|i} = f\left(\frac{D_{FF}}{D_{base}}\right) \quad (7.1)$$

The model for engine size was based on the numerical exercise completed with the geometrically enlarged engine at a constant specific thrust (Appendix E.3.1). The engine fan diameter was increased by factors of 1.23 and 1.45 as compared with the baseline engine. The constant specific thrust of the engine isolated the effect of size on the installation effect ($X_{inst|posC3}$). As anticipated, the modified drag of the nacelle scaled linearly with the square of the engine diameter $\left(\frac{D_{FF}}{D_{base}}\right)^2$ (Figure 7.1), as the other engine performance parameters ($MFCR$, $FNPR$, $CNPR$, BPR , TET) were kept constant. A linear function was defined to approximate the results from the CFD data (Equation 7.2). Similar to previous sections, the results are presented as percentage of reference thrust of the baseline engine E1 ($F_{N\ ref}$), which are based on the thrust requirement during cruise phase of flight. In this context, the penalty of -14.35% $F_{N\ ref}$ for E1 increases linearly with the square of the fan diameter to a value of -30.25% $F_{N\ ref}$ for the 1.45xE1. Clearly, the modified drag of an isolated nacelle (D_{nac}^*) is the key contributing factor to the installation effect. The limitation of the approach with geometrically enlarged engines is that only a single design of the nacelle was considered. Naturally, the magnitude of the drag is dependent on the geometrical design of the nacelle as well. A greater parametric study that analyses the impact of the design on the drag of the nacelle (D_{nac}^*) lied outside of the scope of this project.

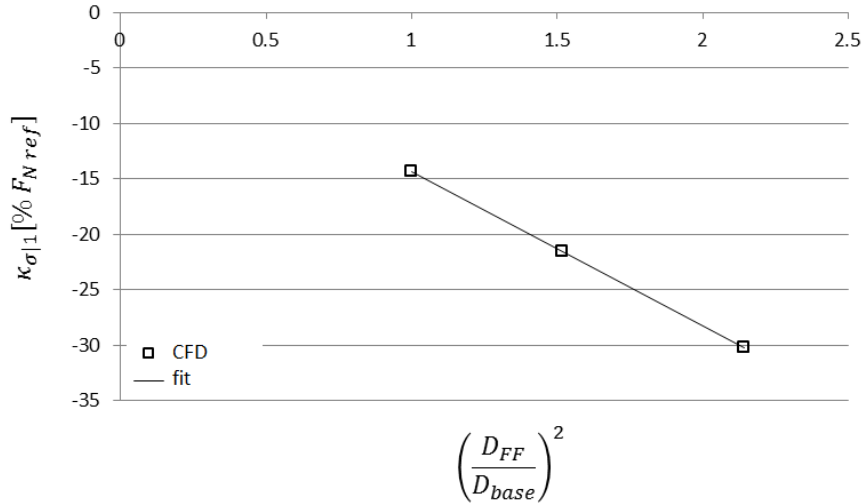


FIGURE 7.1: Proposed correlation for $\kappa_{\sigma|1}$ based on the E1, 1.23xE1 and 1.45xE1 CFD results for the $D_{nac}^*_{iso}$; $M_\infty = 0.82$; $Re = 45 \times 10^6$ $C_{Z'} = 0.5$

$$\kappa_{\sigma|1} = -13.888 \left(\frac{D_{FF}}{D_{base}}\right)^2 - 0.4686 \quad (7.2)$$

The next step was to determine the key sensitivity to engine size for the interference terms on thrust (ΔF_N^*), nacelle drag (ΔD_{nac}^*) and airframe drag ($\Delta D_{A/F}$). All three were found to be linearly related with the square of the engine diameter $\left(\frac{D_{FF}}{D_{base}}\right)^2$. As for the ΔF_N^* (Appendix E.3.1), the greater the engine diameter, the greater projected

area of the core-cowl and the plug was exposed to the overlap with the wing. As the position of the trailing edge of the nacelle was kept constant while the size of the engine was increased, the enlargement of the engine from E1 to 1.45xE1 resulted in a greater overlap of the exhaust section with the wing. As a result, the projected area of the exhaust was exposed to the flow acceleration on the lower side of the wing. The increase of engine size from E1 to 1.45xE1 resulted in a increase of the thrust losses from -1.07% to -5.27% $F_{N \text{ ref}}$ (Figure 7.2). A linear function was proposed to describe the relation (Equation 7.3).

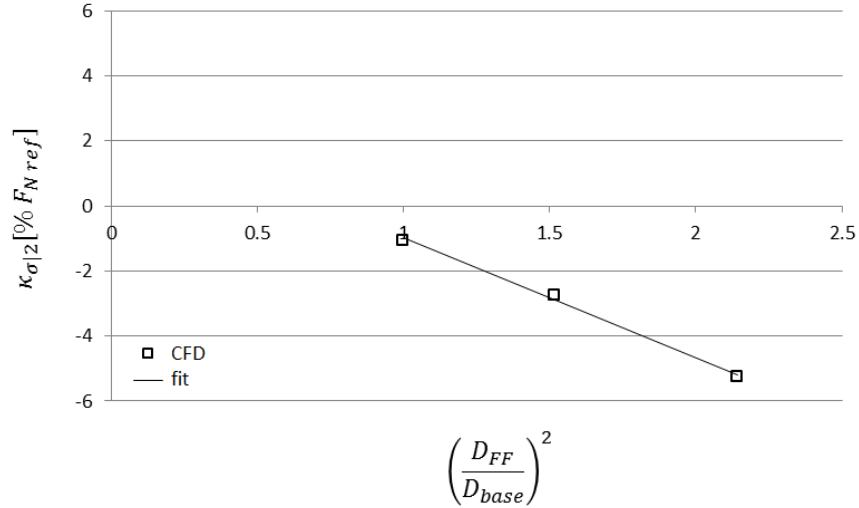


FIGURE 7.2: Proposed correlation for $\kappa_{\sigma|2}$ based on the E1, 1.23xE1 and 1.45xE1 CFD results for the ΔF_N^* ; $M_\infty = 0.82$; $Re = 45 \times 10^6$ $C_{Z'} = 0.5$

$$\kappa_{\sigma|2} = -3.6825 \left(\frac{D_{FF}}{D_{base}} \right)^2 + 2.6925 \quad (7.3)$$

Furthermore, the increase of engine size had a beneficial effect on ΔD_{nac}^* (Figure 7.3), which was linearly correlated with the square of the fan diameter $\left(\frac{D_{FF}}{D_{base}}\right)^2$ (Equation 7.4). The larger the engine diameter, the greater planform area of the nacelle was exposed to the beneficial positive relative pressure from the wing. As a result, the ΔD_{nac}^* increased from 2.6% to 5.23% $F_{N \text{ ref}}$.

$$\kappa_{\sigma|3} = 2.3049 \left(\frac{D_{FF}}{D_{base}} \right)^2 + 0.2249 \quad (7.4)$$

The presence of the nacelle affected the wing aerodynamics and the $\Delta D_{A/F}$ was also proportional to the square of the fan diameter $\left(\frac{D_{FF}}{D_{base}}\right)^2$ (Figure 7.4). A mildly detrimental effect of the nacelle on the airframe drag changed from -0.31% $F_{N \text{ ref}}$ for E1 to a positive effect of +4.17% $F_{N \text{ ref}}$ for 1.45xE1. A linear relation was fitted to the existing dataset (Equation 7.5). The $\Delta D_{A/F}$ depended significantly on the nacelle size, as the greater the nacelle, the larger planform area of the wing was affected. A change of a

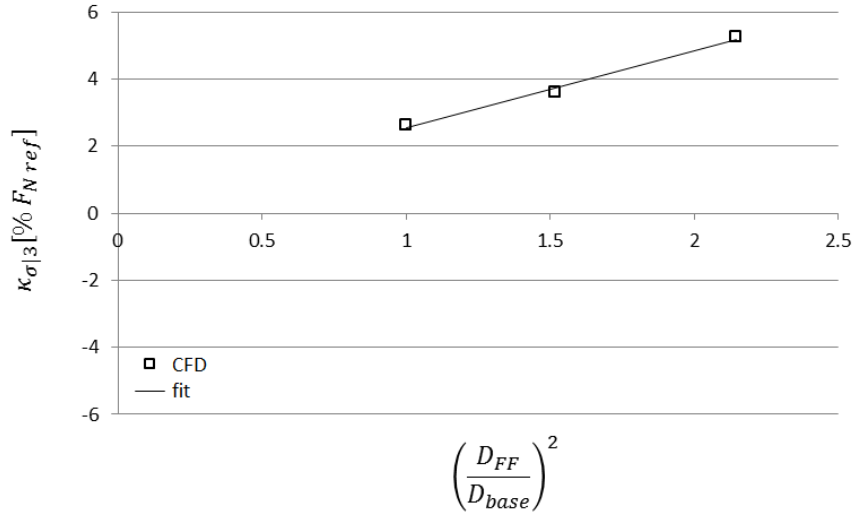


FIGURE 7.3: Proposed correlation for $\kappa_{\sigma|3}$ based on the E1, 1.23xE1 and 1.45xE1 CFD results for the ΔD_{nac}^* ; $M_\infty = 0.82$; $Re = 45 \times 10^6$ $C_{Z'} = 0.5$

local incidence on the wing due to the nacelle reduced the extent of the positive relative pressure at the wing leading edge and locally reduced the pressure drag of the wing.

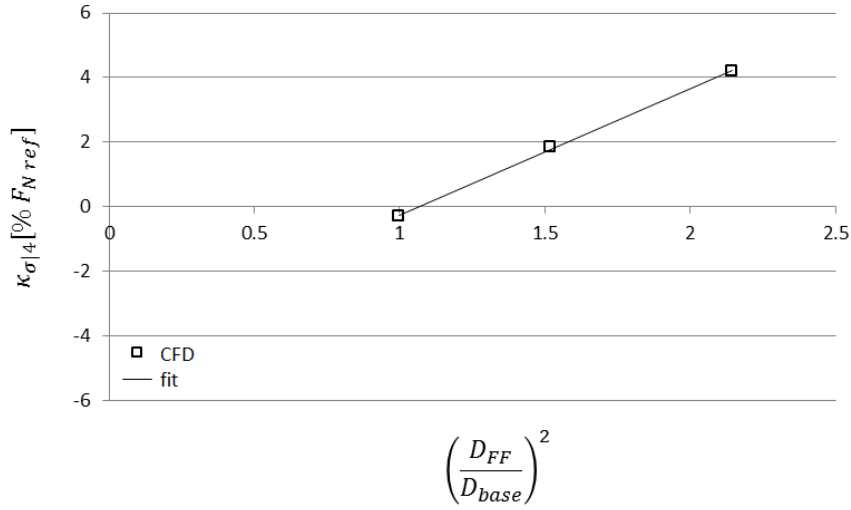


FIGURE 7.4: Proposed correlation for $\kappa_{\sigma|4}$ based on the E1, 1.23xE1 and 1.45xE1 CFD results for the $\Delta D_{A/F}$; $M_\infty = 0.82$; $Re = 45 \times 10^6$ $C_{Z'} = 0.5$

$$\kappa_{\sigma|4} = 3.9136 \left(\frac{D_{FF}}{D_{base}} \right)^2 - 4.1807 \quad (7.5)$$

To determine the quality of the curve fit proposed for the correlation, the basic check of the evaluation of the X_{inst} based on the size of the engine against the CFD points used for the correlation. An error that arose due to the assumption of linear correlation between the components of installation effect and the square of the engine fan diameter was quantified. Overall, the error on the overall installation effect (X_{inst}) was within

$\pm 0.08\%$ of $F_{N \text{ ref}}$ (Table 7.1), which corresponds to 0.2dc. Furthermore, the individual error on the evaluation of each of the contributing factors (D_{nac}^* , ΔF_N^* , ΔD_{nac}^* and $\Delta D_{A/F}$) to the installation effect was assessed (Table 7.2). The best agreement between the CFD and the proposed linear correlation with respect to the $\left(\left(\frac{D_{FF}}{D_{base}}\right)^2\right)$ was noted for the modified nacelle drag D_{nac}^* with 0.00% of $F_{N \text{ ref}}$ deviation in the model. The deviation on the interference terms was lower than 0.15%, 0.15% and 0.08% for ΔF_N^* , ΔD_{nac}^* and $\Delta D_{A/F}$ respectively. It is considered satisfactory, as the errors are below 0.4dc, which an order of magnitude lower than the current state of art in the drag prediction (Section 2.3).

	$X_{inst} [\%F_{N \text{ ref}}]$		
	CFD	Model	Δ
E1	-13.13	-13.08	0.04
1.23xE1	-18.88	-18.97	-0.08
1.45xE1	-26.11	-26.07	0.04

TABLE 7.1: The verification of the size model for the X_{inst}

	$D_{nac}^* [\%F_{N \text{ ref}}]$			$\Delta F_N^* [\%F_{N \text{ ref}}]$		
	CFD	$\kappa_{\sigma 1}$	Δ	CFD	$\kappa_{\sigma 2}$	Δ
E1	-14.36	-14.36	0.00	-1.07	-0.99	0.08
1.23xE1	-21.55	-21.55	0.00	-2.76	-2.90	-0.14
1.45xE1	-30.25	-30.25	0.00	-5.27	-5.20	0.06

	$\Delta D_{nac}^* [\%F_{N \text{ ref}}]$			$\Delta D_{A/F} [\%F_{N \text{ ref}}]$		
	CFD	$\kappa_{\sigma 3}$	Δ	CFD	$\kappa_{\sigma 4}$	Δ
E1	2.60	2.53	-0.07	-0.31	-0.27	0.04
1.23xE1	3.59	3.72	0.14	1.84	1.76	-0.08
1.45xE1	5.23	5.17	-0.06	4.17	4.21	0.04

TABLE 7.2: Verification of the size model (κ_{σ}) for the D_{nac}^* , ΔF_N^* , ΔD_{nac}^* and $\Delta D_{A/F}$

7.2 Model for engine power setting

To propose the model for the relation of engine installation effect ($X_{inst|posC3}$) with respect to the engine power setting, the results for both the E1 and E2 engines were analysed (Section 6.8) in the baseline position C3. The model was proposed for the baseline operating conditions of an aircraft $M_\infty = 0.82$, $C_{Z'} = 0.5$ and $Re = 45 \times 10^6$. The power setting factor ($\kappa_{\psi|i}$) was determined as a multiplication factor to the magnitude determined by the effect of size (Section 7.4). The factor (7.6) was meant to reflect the influence of the mass flow capture ratio ($MFCR$), fan nozzle pressure ratio ($FNPR$) and engine by-pass ratio (BPR) on $D_{nac\ iso}^*$, ΔD_{nac}^* , ΔF_N^* and $\Delta D_{A/F}$.

$$\kappa_{\psi|i} = f(MFCR; FNPR; BPR) \quad (7.6)$$

It was proposed to define a correlation for $D_{nac\ iso}^*$, ΔF_N^* , ΔD_{nac}^* , $\Delta D_{A/F}$ with regards to the engine performance parameters such as $MFCR$, $FNPR$ and BPR . The general form of the correlation function was proposed as a quadratic fit (Equation 7.7) to the existing results presented with respect to a single independent variable λ . The λ was a product of multiplication of the $MFCR$, $FNPR$ and BPR with corresponding exponents (Equation 7.8). The exponents were chosen independently for each of the correlations based on the physical influence on the performance parameter. The initial least squares method for the evaluation of the exponents (a_i , b_i , c_i) was used to determine the general trend in the data. However, the possible solutions were limited to the exponents within -2 and 2 and rounded to ± 0.25 . Furthermore, the coefficients of the quadratic polynomial (α_i , β_i , γ_i in Equation 7.7) were re-optimised with use of the least squares method to determine the best fit to the available data.

$$\kappa_{\psi|i} = \alpha_i \lambda_i^2 + \beta_i \lambda_i + \gamma_i \quad (7.7)$$

$$\lambda_i = MFCR^{a_i} \cdot FNPR^{b_i} \cdot BPR^{c_i} \quad (7.8)$$

For the isolated engine drag ($D_{nac\ iso}^*$), the λ_1 (Equation 7.9) was proposed with the following coefficients $a_1 = 2$, $b_1 = 0.25$ and $c_1 = 0.5$. The $D_{nac\ iso}^*$ was mostly sensitive to the $MFCR$, which determined the spillage drag of the nacelle. The increase of the $MFCR$ contributed to the increase of shape factor $\kappa_{\psi|1}$. However, the change of the fan nozzle pressure ratio ($FNPR$) affected the pressure distribution and effectively the drag of the nacelle generated at the fan-cowl afterbody. The reduction of $FNPR$ had a mildly beneficial effect on the overall trend of $\kappa_{\psi|1}$. Finally, the use of low specific thrust with high BPR reinforced the beneficial effect of the exhaust with a favourable $FNPR$. Overall, a curve fit with the use of a quadratic function was performed (Equation 7.10),

where the coefficients were estimated as $\alpha_1 = 0.0627$, $\beta_1 = -0.441$ and $\gamma_1 = +1.703$ by the least squares method.

$$\lambda_1 = MFCR^2 \cdot FNPR^{0.25} \cdot BPR^{0.5} \quad (7.9)$$

$$\kappa_{\psi|1} = 0.0627\lambda_1^2 - 0.441\lambda_1 + 1.703 \quad (7.10)$$

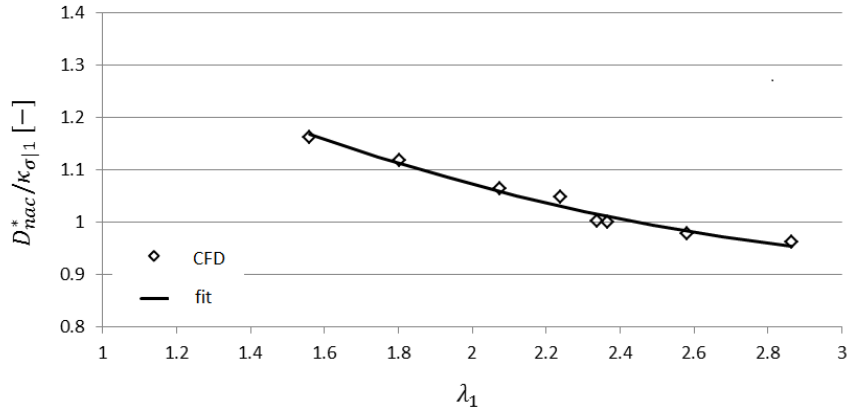


FIGURE 7.5: Proposed correlation for $\kappa_{\psi|1}$ based on the E1 and E2 CFD results for the D_{nac}^* ; $M_\infty = 0.82$; $Re = 45 \times 10^6$ $C_{Z'} = 0.5$

The correlation for the ΔF_N^* with regards to the λ_2 parameter was determined (Figure 7.6). The λ_2 was a product of multiplication of $MFCR$ and $FNPR$ with exponents equal to $a_1 = 0.5$ and $b_1 = 0.5$. The coefficients were selected based on the range of solutions from the initial least squares evaluation to match the physical behaviour in the flow. It was observed that a reduction of power setting and in particular the $FNPR$ reduced the severe effects on the interference on thrust (ΔF_N^*). In this context, the low values of $MFCR$ and $FNPR$ reduce the value of shape factor $\kappa_{\psi|2}$ and ultimately result in more favourable effect on the ΔF_N^* . Moreover, the BPR significantly affected the correlation and the exponent equaled $b_1 = -2$. It was explained, that an increase in by-pass ratio increased the distance between the wing and the core-cowl, which helped to alleviate the negative effects of the overlap between the exhaust and the wing. During the analysis, the CFD result for $\kappa_{\psi|2}$ of the E1 engine at PS2 did not fit any of the trends with the other points. After the thorough checks of the calculation, post-processing, accounting, the point was eliminated from the dataset during the creation of the correlation. The problem was explained as a non-monotonic interaction of the shocks in the jet with the wing-pressure field with respect to the change in $FNPR$. As a result a neutral interference on thrust was achieved at this particular configuration. It can be pertinent in the design work to determine similar points of neutral interference with the overall benefit for the configuration. The case highlights in particular, a need to perform a parametric study on the interference terms with respect to the $FNPR$

and $CNPR$. The final optimisation of the coefficients for the polynomial was completed with the use of least squares method ($\alpha_1 = 10114$, $\beta_1 = -78.45$ and $\gamma_1 = +0.4755$).

$$\lambda_2 = MFCR^{0.5} \cdot FNPR^{0.5} \cdot BPR^{-2} \quad (7.11)$$

$$\kappa_{\psi|2} = 10114\lambda_2^2 - 78.45\lambda_2 + 0.4755 \quad (7.12)$$

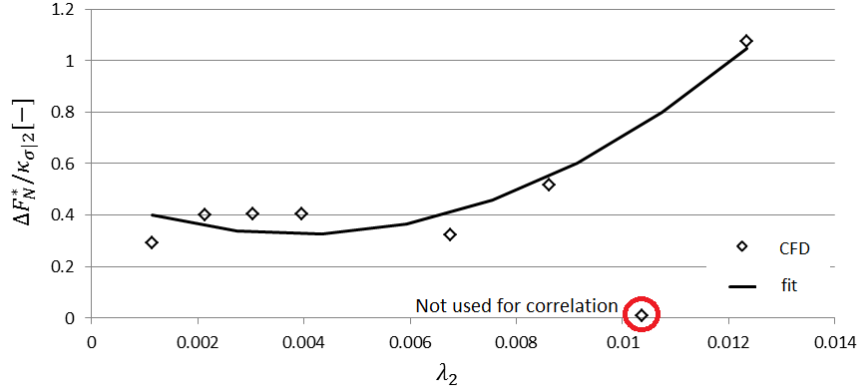


FIGURE 7.6: Proposed correlation for $\kappa_{\psi|2}$ based on the E1 and E2 CFD results for the ΔF_N^* ; $M_\infty = 0.82$; $Re = 45 \times 10^6$ $C_{Z'} = 0.5$

As for the ΔD_{nac}^* , the correlation with respect to λ_3 (Equation 7.13) was also found in the form of a second order polynomial (Figure 7.7 and Equation 7.14). Based on the initial least squares optimisation for the exponent coefficients (a_3 , b_3 and c_3), the λ_3 was inversely related with the $MFCR$ and the exponent equals $a_3 = -1$. The reduction of $MFCR$ resulted in an increased spillage drag and a reduction in the shape factor $\kappa_{\psi|3}$. The increase of spillage drag was more severe for the wing-installed engine as compared with the isolated due to the presence of a local up-wash of the flow to the engine. Simultaneously, the interference drag of the nacelle (ΔD_{nac}^*) worsened for low $FNPR$. The presence of the wing was favourable for high $FNPR$ and the interaction with the wing pressure-field counteracted the detrimental jet entrainment drag for the high $FNPR$. The reduction in $FNPR$ reduced the benefits, even though the low $FNPR$ was in general favourable for the isolated engine drag D_{nac}^* . The engine with high BPR had a lower value of ΔD_{nac}^* as expected from the size study. In general, the high BPR was favourable for the modified nacelle drag (D_{nac}^*), however less of an interference benefit (ΔD_{nac}^*) was available due to the presence of the wing. It was easier to obtain beneficial interference from the worse design, rather than from the favourable design with low specific thrust. Similarly, the least squares optimisation was used for the determination of the coefficients of the proposed polynomial ($\alpha_3 = 1.9885$, $\beta_3 = -1.0552$ and $\gamma_3 = 0.1922$).

$$\lambda_3 = MFCR^{-1} \cdot FNPR^{-0.75} \cdot BPR^{0.25} \quad (7.13)$$

$$\kappa_{\psi|3} = 1.9885\lambda_3^2 - 1.0552\lambda_3 + 0.1922 \quad (7.14)$$

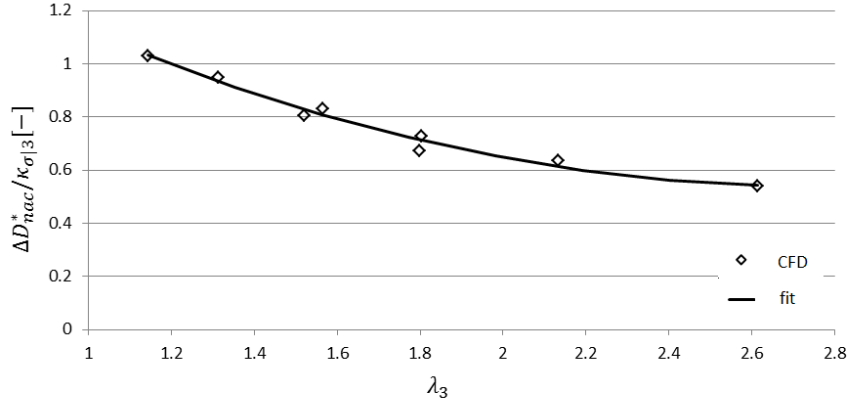


FIGURE 7.7: Proposed correlation for $\kappa_{\psi|3}$ based on the E1 and E2 CFD results for the ΔD_{nac}^* ; $M_\infty = 0.82$; $Re = 45 \times 10^6$ $C_{Z'} = 0.5$

Even though the interference on the airframe drag ($\Delta D_{A/F}$) followed the general trend for size ($\kappa_{\sigma|4}$), it was the least monotonic effect of the installation drag (X_{inst}) with respect to engine power setting ($\kappa_{\psi|4}$). The results of the computations were analysed as a function of λ_4 (Equation 7.15), which is a second-order polynomial (Equation 7.16). The greatest impact from the engine performance parameters was observed for the *BPR*. The high *BPR* resulted in fairly constant effect with respect to the other parameters such as *FNPR* and *MFCR*. The $\kappa_{\psi|4}$ for high *BPR* engine ranged from 0.15 to 0.4. For the low *BPR* engines, the channel between the airframe and the core cowl is smaller, which makes the engine more sensitive to the variation of the *FNPR*. The mildly negative effect evaluated based on the size ($\kappa_{\sigma|4}$) was significantly affected by the change of the *FNPR* and the $\kappa_{\psi|4}$ ranges from -3 to 1 for the *FNPR* from 2.12 to 2.71, respectively. Due to the high non-monotonic behaviour in the results, an extensive study should be dedicated to enrich the dataset for the airframe interference with a preferable variation of the engine *BPR* and *MFCR* as the key factors. Clearly, the restriction on the range of λ_4 is necessary with the bounds between $\lambda_4 = 45$ to $\lambda_4 = 70$. It is anticipated as a serious risk, that the misuse of the correlation outside the spectrum of tested λ_4 may lead to unfeasible evaluations.

$$\lambda_4 = MFCR^{0.5} \cdot FNPR^2 \cdot BPR^1 \quad (7.15)$$

$$\kappa_{\psi|4} = -0.00596\lambda_4^2 + 0.8728\lambda_4 - 30.875 \quad (7.16)$$

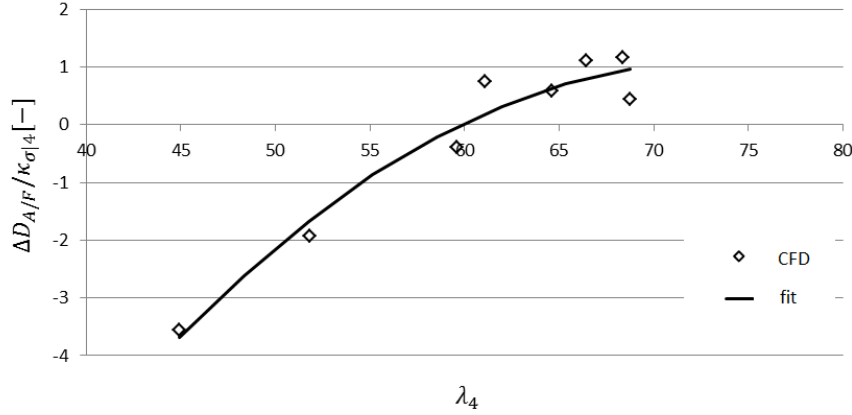


FIGURE 7.8: Proposed correlation for $\kappa_{\psi|4}$ based on the E1 and E2 CFD results for the $\Delta D_{A/F}$; $M_\infty = 0.82$; $Re = 45 \times 10^6$ $C_{Z'} = 0.5$

7.3 Model for engine position

To complete the model, the relation for engine position was proposed in form of the maps for the interference terms (ΔF_N^* , ΔD_{nac}^* and $\Delta D_{A/F}$). The maps were obtained for the baseline engine E1 in the power setting PS1 (Section 6.5.3). It has to be stressed that the position effect was analysed independently from the effect of engine power setting. In the reality the two effects are coupled and multiple engine positions are required to be tested at a range of engine power setting to determine a more accurate trend. The current decoupled approach was meant to provide the basic guidance and to inform about the key sensitivities of the interference terms with respect to the engine position. The position dependent factor $\Delta\kappa_{position}$ of the overall installation effect (X_{inst} , Equation 7.21) was proposed as a sum of the individual interference effects (ΔF_N^* , ΔD_{nac}^* and $\Delta D_{A/F}$) referenced to the corresponding interference term at a baseline position of the engine (C3) (Equation 7.17), where $\kappa_{pos|\Delta F_N^*}$, $\kappa_{pos|\Delta D_{nac}^*}$ and $\kappa_{pos|\Delta D_{A/F}}$ were the relative values of interference for the thrust (ΔF_N^*), modified nacelle drag (ΔD_{nac}^*) and airframe drag ($\Delta D_{A/F}$), respectively (Equations 7.18-7.20).

$$\Delta\kappa_{position} = \kappa_{pos|\Delta F_N^*} + \kappa_{pos|\Delta D_{nac}^*} + \kappa_{pos|\Delta D_{A/F}} \quad (7.17)$$

$$\kappa_{pos|\Delta F_N^*} = (\Delta F_N^* - \Delta F_{N|C3}^*) / F_{N \text{ ref}} [\%] \quad (7.18)$$

$$\kappa_{pos|\Delta D_{nac}^*} = (\Delta D_{nac}^* - \Delta D_{nac|C3}^*) / F_{N \text{ ref}} [\%] \quad (7.19)$$

$$\kappa_{pos|\Delta D_{A/F}} = (\Delta D_{A/F} - \Delta D_{A/F|C3}) / F_{N \text{ ref}} [\%] \quad (7.20)$$

The details of the sensitivity of the individual interference effects were discussed in Section 6.5.3 for the computed engine positions. However, for the purpose of the model, spatial maps were created with the use of a natural neighbour interpolation based on

Delaunay triangulation. With the use of the spatial maps the relative interference can be evaluated for an arbitrary position within the tested range. The relative interference on thrust ($\kappa_{pos|\Delta F_N^*}$ in Figure 7.9) improved as the engine was moved away from the baseline position. The potential improvement of 1.9% was the result of the movement of the engine upstream and the reduction of the overlap with the wing. Similarly, the modified nacelle drag reduced and the relative interference benefit ($\kappa_{pos|\Delta D_{nac}^*}$) increased by approximately +2.5% for the engine moved further upstream from the wing (Figure 7.10). The two beneficial effects ($\kappa_{pos|\Delta F_N^*}$ and $\kappa_{pos|\Delta D_{nac}^*}$) were opposed by a detrimental effect on the airframe drag ($\kappa_{pos|\Delta D_{A/F}}$ in Figure 7.11). The $\kappa_{pos|\Delta D_{A/F}}$ reduced by -7% when the engine was moved upstream. In addition the movement of the engine vertically closer to the wing increased the interaction with the jet and also worsened the $\kappa_{pos|\Delta D_{A/F}}$. Overall, the superposition of the described sensitivities led to an overall sensitivity of the $\Delta\kappa_{position}$ by +1.7% relative to the baseline position C3 (Figure 6.27 in Section 6.5.3). The “blind test” verification of the model is suggested for an arbitrary position within the range. Moreover, the existence of similar interference maps in any other study were not known to the author. The verification of the maps was based on the agreement of general trends of the changes in the wing pressure distributions due to the movement of the engine (Section 6.5) as compared with the trends observed in the previous research (Section 2.3.2). Overall, the presented sensitivity maps allow for the preliminary design evaluation of the installation effect (X_{inst}) for an arbitrary position.

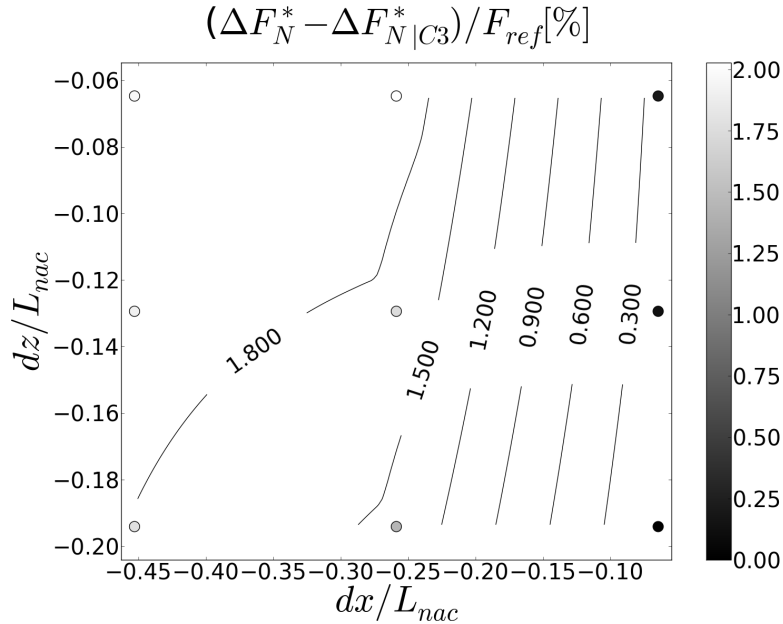


FIGURE 7.9: The sensitivity map for the ΔF_N^* relative to E1 in position C3; $M_\infty = 0.82$; $Re = 45 \times 10^6$ $C_{Z'} = 0.5$

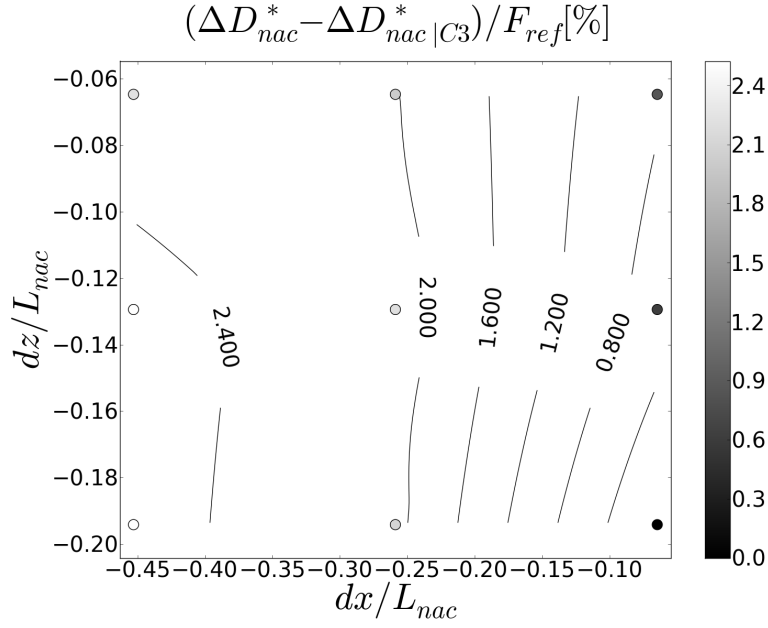


FIGURE 7.10: The sensitivity map for the ΔD_{nac}^* relative to E1 in position C3; $M_\infty = 0.82$; $Re = 45 \times 10^6$ $C_{Z'} = 0.5$

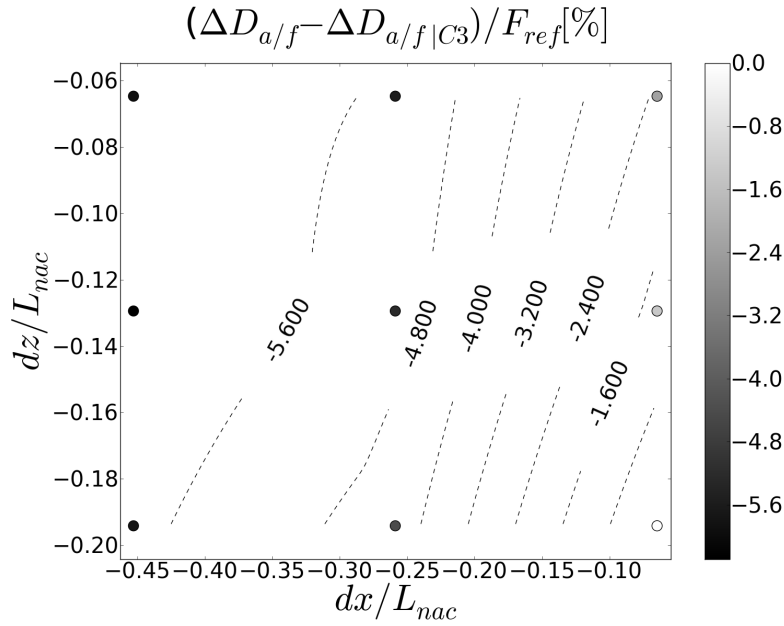


FIGURE 7.11: The sensitivity map for the $\Delta D_{A/F}$ relative to E1 in position C3; $M_\infty = 0.82$; $Re = 45 \times 10^6$ $C_{Z'} = 0.5$

7.4 Complete model

The general form of the model for engine installation (X_{inst}) was proposed (Equation 7.21) as a superposition of the sensitivity to engine position ($\Delta F_{position}$, Equation 7.17) and the evaluation of the installation penalty for the baseline position ($X_{inst|posC3}$). The installation penalty for the engine in the baseline position ($X_{inst|posC3}$) was decomposed (Equation 7.22) into the key contributors such as modified drag of an isolated engine ($D_{nac iso}^*$), interference on modified thrust (ΔF_N^*), interference on the modified nacelle drag (ΔD_{nac}^*) and interference on the airframe drag ($\Delta D_{A/F}$).

$$X_{inst} = X_{inst|posC3} + \Delta \kappa_{position} \quad (7.21)$$

$$X_{inst|posC3} = D_{nac iso}^* + \Delta F_N^* + \Delta D_{nac}^* + \Delta D_{A/F} \quad (7.22)$$

The general form of a correlation for the key contributors to the $X_{inst|posC3}$ was proposed as a multiplication of the size factor ($\kappa_{\sigma|i}$) and power setting factor ($\kappa_{\psi|i}$) and it took the following form for the $D_{nac iso}^*$ (Equation 7.23), ΔF_N^* (Equation 7.24), ΔD_{nac}^* (Equation 7.25) and $\Delta D_{A/F}$ (Equation 7.26).

$$D_{nac iso}^* = \kappa_{\sigma|1} \kappa_{\psi|1} \quad (7.23)$$

$$\Delta F_N^* = \kappa_{\sigma|2} \kappa_{\psi|2} \quad (7.24)$$

$$\Delta D_{nac}^* = \kappa_{\sigma|3} \kappa_{\psi|3} \quad (7.25)$$

$$\Delta D_{A/F} = \kappa_{\sigma|4} \kappa_{\psi|4} \quad (7.26)$$

Overall, the model allows to determine the $D_{nac iso}^*$, ΔF_N^* , ΔD_{nac}^* and $\Delta D_{A/F}$ based on the engine size, power-setting and position. However, the validation and verification of the model are essential to determine the credibility of the evaluations.

7.5 Verification of the low order models

The validation of the proposed model for the engine size and power setting is not available due to the time constraints on the project. It is suggested to validate the model with use of “blind test” methodology, where the model is used to evaluate the installation effect (X_{inst}) for a prospective configuration prior to the design and computation of the actual result. Nevertheless, a basic verification for the curve-fitting process for the individual effects of installation, as well as the verification of the stack-up for the size and power submodels into $X_{inst|C3}$ was completed at the baseline engine position C3. For this reason, the evaluation of the modified nacelle drag for an isolated engine (D_{nac}^*),

interference on the thrust (ΔF_N^*), interference on the modified nacelle drag (ΔD_{nac}^*) and interference on the airframe drag ($\Delta D_{A/F}$) was completed and compared against the CFD result for each datapoint used in the creation of the model.

The prediction of the D_{nac}^* in position C3 was found to be within $\pm 0.4\% F_{N \text{ ref}}$ error as compared with the computations (Table 7.3), which corresponds to approximately 1dc. The average error of the prediction is $\pm 0.2\% F_{N \text{ ref}}$. It is considered a satisfactory result with the agreement representative of current standards in drag prediction.

Position C3	Model [% $F_{N \text{ ref}}$]	CFD [% $F_{N \text{ ref}}$]	Δ [% $F_{N \text{ ref}}$]
E1 PS1	-14.36	-14.51	-0.153
E1 PS2	-15.26	-15.19	0.074
E1 PS3	-16.05	-15.96	0.084
E1 PS4	-16.67	-16.76	-0.086
E2 PS1	-20.71	-20.57	0.134
E2 PS2	-21.06	-21.18	-0.119
E2 PS3	-21.59	-21.87	-0.288
E2 PS4	-22.59	-22.20	0.392

TABLE 7.3: Verification of the curve-fit for the effect of modified nacelle drag (D_{nac}^*)

The prediction of the interference on thrust ΔF_N^* in position C3 resulted in an average error within $\pm 0.2\% F_{N \text{ ref}}$ with an exception for the single case of E1 engine at PS2 (Table 7.4) with $+0.7\% F_{N \text{ ref}}$. The E1 engine at PS2 was eliminated from the dataset during the creation of the model. Even though all necessary checks for the simulations have been done, the reason of the non-monotonic behaviour of the interference on thrust with respect to power setting was not found. It was assessed that the cause is in the nonlinear interaction of the jet with the wing pressure field due to the presence of the shocks in the by-pass stream. The overall result of $\pm 0.2\% F_{N \text{ ref}}$ accuracy for the model was found to be acceptable on the basis that the interference on thrust is a modest component of the X_{inst} . As a result, the potential error in the prediction has a relatively modest effect on the overall outcome of the model.

Position C3	Model [% $F_{N \text{ ref}}$]	CFD [% $F_{N \text{ ref}}$]	Δ [% $F_{N \text{ ref}}$]
E1 PS1	-1.04	-1.07	-0.029
E1 PS2	-0.74	-0.01	0.732
E1 PS3	-0.54	-0.51	0.033
E1 PS4	-0.40	-0.32	0.086
E2 PS1	-0.94	-1.17	-0.232
E2 PS2	-0.96	-1.17	-0.214
E2 PS3	-1.03	-1.16	-0.131
E2 PS4	-1.16	-0.84	0.314

TABLE 7.4: Verification of the curve-fit for the interference effect on the modified thrust (ΔF_N^*)

The prediction of the interference on the modified nacelle drag (ΔD_{nac}^*) in position C3 had an average error of $\pm 0.02\%F_{N\ ref}$ (Table 7.5). The interference on the nacelle drag is the second most important contributing factor to the X_{inst} . It was found to be satisfactory that the little deviation in the proposed correlation was found. The determination of the beneficial effect of the wing pressure-field on the fan-cowl afterbody and the assessment of the interference due to the upwash angle are the core of the installation work. However, it is suggested to expand the dataset and base the updated model on a larger spectrum of nacelle designs. The investigated nacelles had a conventional design and potential change of the design should be reflected in a more general model for engine installation.

Position C3	Model [$\%F_{N\ ref}$]	CFD [$\%F_{N\ ref}$]	Δ [$\%F_{N\ ref}$]
E1 PS1	2.63	2.60	-0.027
E1 PS2	2.34	2.40	0.056
E1 PS3	2.04	2.03	-0.010
E1 PS4	1.73	1.70	-0.027
E2 PS1	3.11	3.09	-0.012
E2 PS2	2.72	2.70	-0.013
E2 PS3	2.32	2.36	0.040
E2 PS4	2.01	2.01	-0.004

TABLE 7.5: Verification of the curve-fit for the interference effect on the modified nacelle drag (ΔD_{nac}^*)

The result of the assessment for the submodel of the interference of the airframe ($\Delta D_{A/F}$) in position C3 was the least acceptable (Table 7.6). It was found that the average error of the $\Delta D_{A/F}$ was about $\pm 0.35\%F_{N\ ref}$. The result was found unacceptable as the interference on the airframe drag was the the third most important contributor to the installation effect (X_{inst}) and it can be of similar magnitude to the interference on the nacelle drag (ΔD_{nac}^*). The reason for such a poor agreement was found to be a highly non-monotonic behaviour of the interference term with the respect to the engine power setting. The available dataset was not rich enough to determine a clear trend for all the cases, which leads to the rise of error for individual cases during the model evaluation. The broad trend for the $\Delta D_{A/F}$ was outlined and it was understood that a more detailed parametric study with respect to $FNPR$ and BPR of an engine needs to be completed.

Eventually, the assessment of the overall prediction for the installation effect in position C3 ($X_{inst|C3}$) was completed (Table 7.7). Even though, some large discrepancies were found at the component level, the overall installation effect is predicted within $\pm 1.3\%F_{N\ ref}$ margin, where the average deviation is $\pm 0.5\%F_{N\ ref}$, which corresponds to 1.25dc. Overall, this very basic verification of the model allowed to determine the general stability of the prediction. It was considered acceptable and satisfactory that the proposed model determined the installation drag with an accuracy comparable to the numerical method used for the creation of the dataset. It is believed that the proposed

Position C3	Model [% $F_{N \text{ ref}}$]	CFD [% $F_{N \text{ ref}}$]	Δ [% $F_{N \text{ ref}}$]
E1 PS1	-0.25	-0.31	-0.061
E1 PS2	0.01	0.11	0.099
E1 PS3	0.44	0.51	0.072
E1 PS4	0.98	0.95	-0.035
E2 PS1	1.69	0.77	-0.912
E2 PS2	1.12	1.03	-0.093
E2 PS3	0.35	1.32	0.974
E2 PS4	1.41	1.93	0.522

TABLE 7.6: Verification of the curve-fit for the interference effect on the modified nacelle drag ($\Delta D_{A/F}$)

model is suitable for the preliminary studies of the engine-aircraft performance and as a guidance in the design of engine installation.

Position C3	Model [% $F_{N \text{ ref}}$]	CFD [% $F_{N \text{ ref}}$]	Δ [% $F_{N \text{ ref}}$]
E1 PS1	-13.16	-13.13	0.036
E1 PS2	-13.58	-12.77	0.813
E1 PS3	-14.02	-14.01	0.010
E1 PS4	-14.45	-14.34	0.110
E2 PS1	-16.72	-18.01	-1.290
E2 PS2	-18.30	-18.50	-0.201
E2 PS3	-20.23	-19.06	1.171
E2 PS4	-19.93	-19.49	0.440

TABLE 7.7: Verification of the model for the installation effect ($X_{inst|C3}$)

At the moment the verification of the superposition of the position model with the size and power-setting models is not possible. The verification would require an additional computational campaign to test engines in different positions at a range of power settings. In this sense the position model is de-coupled from the models for the size and power-setting. It is strongly recommended that selected engine positions and sizes are tested at a range of power-settings to verify and validate the proposed model.

7.6 Conclusions

The model for the overall engine installation effect (X_{inst}) based on the engine position, size and power setting was proposed. The work on the creation of the model can be summarised with the following conclusions:

- The general sensitivity of the component interference was outlined with respect to engine position
- The linear sensitivity of the interference effects (ΔF_N^* , ΔD_{nac}^* and $\Delta D_{A/F}$) with respect to the square of the engine fan diameter $\left(\frac{D_{FF}}{D_{base}}\right)^2$ was determined.
- The correlation of the interference effects (ΔF_N^* , ΔD_{nac}^* and $\Delta D_{A/F}$) due to the change in power setting were determined with respect to the key engine performance parameters such as mass flow capture ratio ($MFCR$), fan nozzle pressure ratio ($FNPR$) and by-pass ratio (BPR).
- The basic verification of the model for the engine size and power setting was completed to determine the error of $\pm 0.5\% F_{N_{ref}}$ (1.25dc) due to the approximation of the data with proposed correlations.
- The superposition of the effects of engine position, size and power setting was proposed to determine the overall engine installation effect (X_{inst}).
- An extensive validation and verification campaign is required to increase the credibility of the proposed model. Computation of additional engines with different sizes over a range of engine positions and power settings is advised to verify the evaluations from the proposed model.
- The existence of a similar model within the public domain is currently not known. It is expected that the proposed model can significantly contribute to the state of the knowledge and inform design decisions at a preliminary design stage.

Summary and conclusions

“Reasoning draws a conclusion,
But does not make the conclusion certain,
Unless the mind discovers it
By the path of experience.”

Roger Bacon

8.1 Main accomplishments

This work significantly improved the application of the numerical methods related to the evaluation of engine installation at a preliminary design stage. The most important accomplishments are listed below:

- The work contributed to the development of the engine design methods:
 - An engine cycle optimisation routine which used the genetic algorithm was created to improve the design and optimisation of engine cycles.
 - CST curves were implemented in the fully parametric definition of an axisymmetric engine geometry to automatically design the external aerolines of a modern turbofan engine.
 - A semi-automated CFD process was established to evaluate the aerodynamics of the potential engine designs in a timely manner. Given the axisymmetric engine geometry, the structural mesh is automatically generated and the computation with RANS solver is initiated based on the specified pre-set. Following the computation, the post processing scripts are initiated by the user to evaluate the critical performance parameters of the computed configuration.

- The work established a numerical methodology for the investigation of the engine-airframe integration for turbofan engines:
 - An extensive validation activity was conducted to increase the credibility of the employed numerical method and led to improvement of the meshing and computation strategies.
 - A semi-automated process was established for the meshing and CFD computation of the wing-installed engines
 - The modified near-field method for the evaluation of engine drag and thrust was extended for the application to the engine configuration with separate exhaust in the presence of the airframe.
 - The complete system of the thrust and drag book-keeping for the engine-airframe was developed based on the modified near-field method with the determination of the component interference terms for the aircraft configurations considered at a constant overall vertical force.
- An assessment of the magnitude of engine installation effects with respect to the key parameters such as engine position, size and power setting was completed:
 - Two engine designs (E1 and E2) were tested for the specified aircraft mission over a range of 9 positions to determine the sensitivity of the overall installation effect to the engine position. Also, the impact of the engine position on the individual interference terms was highlighted.
 - A range of engine power settings for two engines was considered to determine the sensitivity of the installation effect to the throttle dependent performance parameters such as mass flow capture ratio, fan nozzle and core nozzle pressure ratios.
 - An additional numerical experiment was conducted with the geometrically enlarged version of the baseline engine to isolate the interference terms that depend solely on the size of the nacelle.
- The evaluation of the engine-aircraft performance during the cruise phase of flight was quantified:
 - The sensitivity of engine fuel burn to the engine position was quantified for E1 and E2 engines over the range of engine positions at a 'start of cruise'.
 - The integrated cruise fuel burn was calculated for the E1 and E2 engines for the long-haul flight. The penalty due to the engine weight, as well as the addition of the throttle dependent terms were incorporated in the evaluation of the integrated outcome.

8.2 Main conclusions

- A successful numerical strategy has been developed for the assessment of installation effects on the aircraft and nacelle drag. Based on the validation activities, the absolute accuracy for the total aircraft drag ($C_{D\ tot}$) was determined as less than 10dc. The accuracy in the prediction of installation drag coefficient was 1dc (4% of the installation drag and 0.4% of total aircraft drag) over the range of aircraft lift coefficients from 0.4 to 0.6. Whereas, the accuracy in determination of thrust coefficient (C_V) was 0.01 (0.1%) over the range of fan nozzle pressure ratios (FNPR) from 1.4 to 2.8 that encompasses a full range of engine power settings considered in the study.
- An overall installation force (F_{inst}) was proposed to address all the effects of installation for a re-trimmed aircraft configuration at an overall vertical force coefficient ($C_{Z'}$) that included the vertical component of thrust. Moreover, the breakdown of F_{inst} was proposed, which outlined the contribution of the interference on airframe drag ($\Delta D_{A/F}$), interference on nacelle drag (ΔD_{nac}^*), and interference on engine thrust (ΔF_N^*)
- The sensitivity of the overall installation effect (F_{inst}) to the engine position was quantified for the E1 and E2 engines and it equaled 1.7% and 1.3% $F_{N\ ref}$ across the full range of positions for E1 and E2, respectively. The engine position varied axially from $\Delta X/L_{nac|E1} = -0.45$ to $\Delta X/L_{nac|E1} = -0.065$ and vertically from $\Delta Z/L_{nac|E1} = -0.065$ to $\Delta Z/L_{nac|E1} = -0.19$.
- The estimated benefits for the larger E2 engine as compared with the baseline E1 engine were in the order of -5.8% less in fuel consumption, based on the analysis of the thermodynamic cycle. Furthermore, the integrated cruise fuel burn that took into account the effects of throttle dependent aerodynamic effects, installation terms and the engine weight penalty that were considered over a long-haul cruise phase reduced the fuel burn reduction of -5.8% to -4.8% in favour of the larger E2 engine.
- The superposition of the effects of engine position, size and power setting was proposed to determine the engine installation effect (X_{inst}). A single model to integrate all the effects was proposed based on the found correlations. A linear sensitivity of the interference effects (ΔF_N^* , ΔD_{nac}^* and $\Delta D_{A/F}$) with respect to the square of the engine fan diameter $\left(\frac{D_{FF}}{D_{base}}\right)^2$ was determined. The correlation of the interference effects (ΔF_N^* , ΔD_{nac}^* and $\Delta D_{A/F}$) due to the change in power setting were determined with respect to the key engine performance parameters such as mass flow capture ratio ($MFCR$), fan nozzle pressure ratio ($FNPR$) and by-pass ratio (BPR). The basic verification of the proposed model for the

engine size and power setting was completed to determine the error of $\pm 0.5\% F_{N \text{ ref}}$ (1.25dc) due to the approximation of the data with proposed correlations.

8.3 Novelty

The work significantly improved the methodology for the studies on the engine installation. Firstly, the semi-automated design methodology for the creation and optimisation of engine cycle, generation of engine geometry and evaluation of the aerodynamic performance of isolated engines was put in place (Appendix B). Secondly, based on the experience from previous studies (Section 2.3.3), an improvement to the numerical methodology were made, which ultimately led to an accuracy of 1dc in the prediction of the engine installation at a cruise condition (Section 5.3.1). A semi-automated process was established to evaluate engine-airframe configurations. Moreover, the work proposed a complete system for the evaluation for all of the effects of engine installation on the lift, drag and thrust forces. As compared with the accounting system used in previous studies^{53,55} (Appendix A.2), the detailed decomposition of the overall installation term was proposed (Section 4.4.2), which was possible due to the use of the modified near-field method (Section 4.4.1).

No publicly available study was found to systematically address the engine installation effects for realistic turbofan engines on commercial aircraft. The key sensitivities of the engine installation effect were addressed individually either experimentally (Section 2.2.2) or numerically (Section 2.3.2), however, there was no study that would address systematically the sensitivity to the engine position, size and power setting. Moreover, there is no study publicly known that would propose a coherent model for the evaluation of the overall installation effect, based on the quantified sensitivities. In this context, the current work contributes significantly to the current knowledge in the field of engine installation. The analysis of the engine installation and the proposed model should improve the decisions taken at a preliminary design stage of the turbofan design.

8.4 Publications

The work described in the thesis led to the following publications:

1. Stańkowski, T.P., MacManus, D.G., “Engine installation module development for an aircraft mission analysis framework”, In: Advanced Aero Concepts, Design and Operations Conference, 27 June 2014, Bristol, United Kingdom.
2. Stańkowski, T.P., MacManus, D.G., Sheaf, C.T, Christie, R., “Aerodynamics of aero-engine installation”, Proceedings of the Institution of Mechanical Engineers,

Part G: Journal of Aerospace Engineering (published online on 24.02.2016: DOI: 10.1177/0954410016630332)

3. Stańkowski, T.P., MacManus, D.G., Sheaf, C.T., Grech, N., “Aerodynamics of aero-engine installation”, In: 54th AIAA Aerospace Science Meeting, 4-8 January 2016, San Diego, California, USA, AIAA-2016-0764
4. Stańkowski, T.P., MacManus, D.G., Sheaf, C.T., Grech, N., “Aerodynamic interference for aero-engine installation”, In: 54th AIAA Aerospace Science Meeting, 4-8 January 2016, San Diego, California, USA, AIAA-2016-0766
5. Stańkowski, T.P., MacManus, D.G., Robinson, M., Sheaf, “The aerodynamic effects of VHBR engine installation to the Common Research Model” - paper submitted to the AIAA Journal of Aircraft

Moreover, the work contributed to the projects undertaken in parallel by other authors by the improvement of the engine design methodology:

1. Heidebrecht, A., Stańkowski, T., MacManus, D., “Parameteric geometry and CFD process for turbofan nacelles”, In: Proceedings of ASME 2016 Turbo Expo: Turbomachinery Technical Conference and Exposition, June 13-17 2016, Seoul, South Korea, GT2016-57784.
2. Goulos, I., Stańkowski, T.P., Otter, J., MacManus, D., “Aerodynamic Design of Separate-Jet Exhausts for Future Civil Aero-Engines—Part I: Parametric Geometry Definition and Computational Fluid Dynamic Approach”, In: Journal of Engineering for Gas Turbines and Power, August 2016, Vol. 138 No.081201-1
3. Goulos, I., Otter, J., Stańkowski, T.P., MacManus, D., “Aerodynamic Design of Separate-Jet Exhausts for Future Civil Aero-Engines—Part II: Design Space Exploration Surrogate Modelling, and Optimization”, In: Journal of Engineering for Gas Turbines and Power, August 2016, Vol. 138 No. 081202-1

CHAPTER 9

Future work

“Every new beginning comes from
some other beginning’s end.”

Seneca

The research led to the discovery of new research questions that need to be addressed to enable more accurate quantification of the engine installation. To mention the most interesting directions of the future research:

- A wider range of Mach number and operating altitudes should be tested to allow for the cruise phase evaluation with a step climb procedure
- A greater range of engine positions should be considered to refine the resolution of the overall installation maps and broaden their scope to include the wing-overlapped engine positions
- The effect of engine power setting should be tested at a greater range of engine positions to determine the relation between the effect of engine power setting with effect of engine position. The particular interest should lie in the verification of the sensitivity of the engine installation effect to the engine power setting at different engine positions.
- A non-axisymmetric nozzle design with the bifurcation in the by-pass duct and the engine pylon should be addressed to evaluate more realistic configurations and quantify the effect of engine pylon on the engine-aircraft performance.
- The use of non-axisymmetric intake geometries should be incorporated to the installation studies to minimise the spillage drag at a cruise condition
- The reduced order model for the evaluation of force on the post-exit streamtube should be proposed based on the parameteric study of the design of exhausts.

- The multiparameter approach for the generation of the reduced order model for engine installation effect should be followed. Given a greater number of computation will be performed, a model that includes a cross-correlation of the effect of engine position and the effect of the engine power setting needs to be evaluated. A potential optimum power settings at different engine positions can be found due to the beneficial interaction between the wing and the jet.
- The development of the engine optimisation code should be continued to enable the variation of a greater number of engine parameters and increase the flexibility of the engine models that are subject of optimisation. The implementation of extensive maps of optimum fan nozzle pressure ratio as a function of COT and BPR is advised.
- The further development of the geometry generation tool and automated meshing routines. The creation of the link between the geometry generation, automated meshing, CFD computation and automatic evaluation of the results should be implemented and improved to create a wider platform for the evaluation engine aerodynamics. The GEMINI framework should be improved and expanded to incorporate the three dimensional engine geometries with an automated meshing method and automated solver computation.
- The improvement of the turbulence modelling with the use of the transitional model $k - \omega$ SST $\gamma - \theta$ for the engine-airframe applications is required. The creation of user defined functions within the solver can improve the accuracy in the determination of the airframe drag due to the more accurate determination of the transition point.

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APPENDIX A

Drag extraction methods

This appendix gathers the information about some of the drag extraction methods mentioned within the current thesis, however not directly pertinent to the core studies. The description of other drag methods was discussed in Section 2.4.

A.1 The Wake Integral Method

The drag of a nacelle can be determined by the comparison of the fluid momentum downstream of the body with the momentum of the freestream flow (Equation A.1).

$$D = \int \int \rho V (V_\infty - V) dA \quad (\text{A.1})$$

Applying Equation A.1 to a body of revolution leads to Equation A.2, where velocity (V) is a function of radial (r) and circumferential (θ) position.

$$D = \int_0^\theta \int_r^R \rho V (V_\infty - V) r dr d\theta \quad (\text{A.2})$$

The method was used by Langley et al.¹⁹ in the studies on the nacelle designs. The pitot-static rake was used in these tests (Figure A.1). Local static pressure (p) and ambient pressure (p_∞), as well as the total pressure and temperature within the wind tunnel $T_{t\infty}$ and $p_{t\infty}$ were known. To eliminate the error of integration of negligible velocity deviations over large areas at high radia, the cut-off level for the integration was derived. The outer edge of the wake was found by scanning rake measurements from the outermost rake inwards until a total head loss of 0.013% was found (Figure A.1). Only the measurements from the wake that met the total head loss were integrated. The remaining flow properties required for the method can be determined based on the isentropic assumption. The local Mach number can be calculated using Equation A.3 which enables the local static temperature to be calculated from Equation A.4. The local velocity and density can be found using Equations A.5 and A.6, respectively. Now

all the variables in Equation A.2 are known and drag can be calculated. The method was tested on the numerical results and typically 3-5 measurement points were used for the calculation of the drag.

$$M = \sqrt{\left[\left(\frac{p_{t\infty}}{p} \right)^{\frac{\gamma}{\gamma-1}} - 1 \right] \frac{2}{\gamma-1}} \quad (\text{A.3})$$

$$T = \frac{T_t}{1 + \frac{\gamma-1}{2} M^2} \quad (\text{A.4})$$

$$V = M \cdot \sqrt{\gamma R T} \quad (\text{A.5})$$

$$\rho = \frac{p}{R T} \quad (\text{A.6})$$

Then Equation A.2 can be used to calculate the drag with V_∞ being the average of all good readings outside the wake.

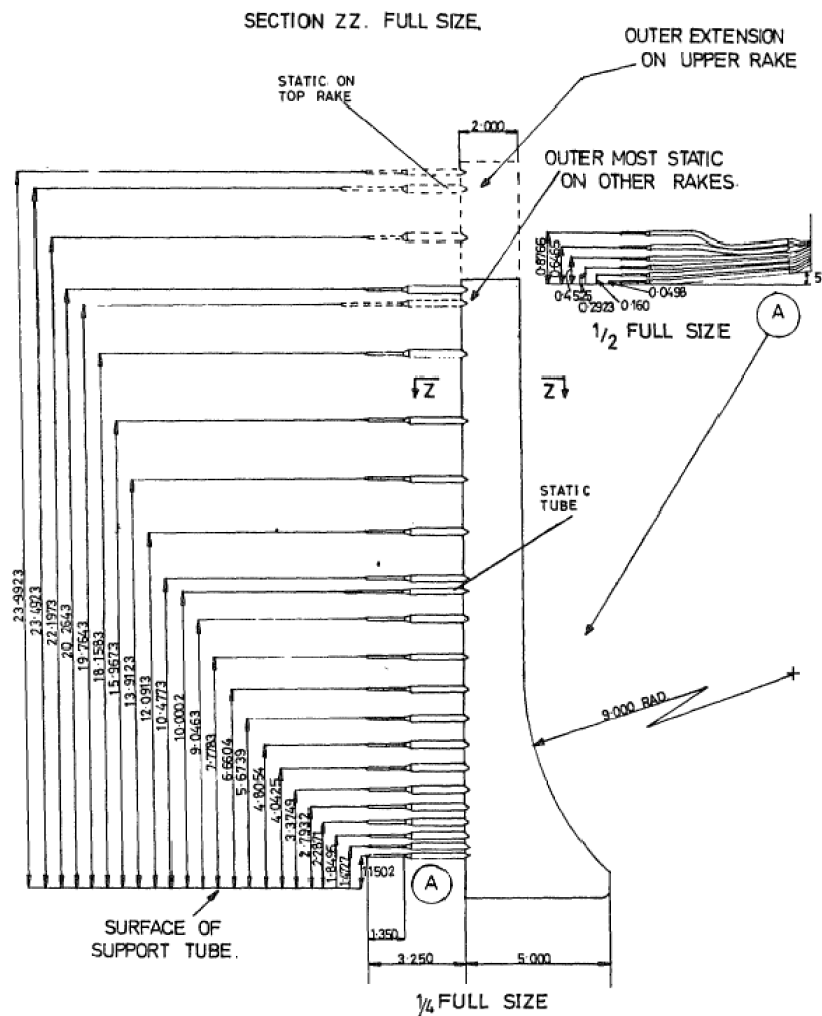


FIGURE A.1: ARA experimental pitot-static rake²⁴

A.2 The drag extraction in ALVAST TPS studies

The thrust and drag book-keeping system that was used in the ALVAST studies is analysed^{53,55} to check the compatibility with the system used in the current project (Section 4.4). The decomposition of the drag system is vital to determining the common ground for the comparison of ALVAST results with the results from the current project in Section 6.6.2.

Based on the the pre-entry and post-exit streamtubes the engine domain around the engine-airframe configuration was split into a drag domain and a thrust domain (Figure A.2). The drag domain is located outside of the engine streamtubes and it wets the surface of the airframe, nacelle and parts of the pylon. The thrust domain contains of the intake diffuser and the core-cowl, the plug and the parts of the pylon washed by the by-pass flow (Figure A.2).

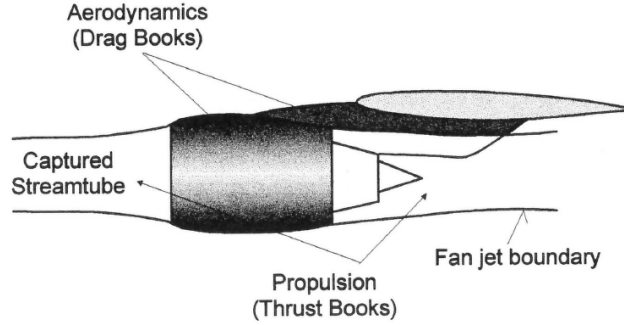
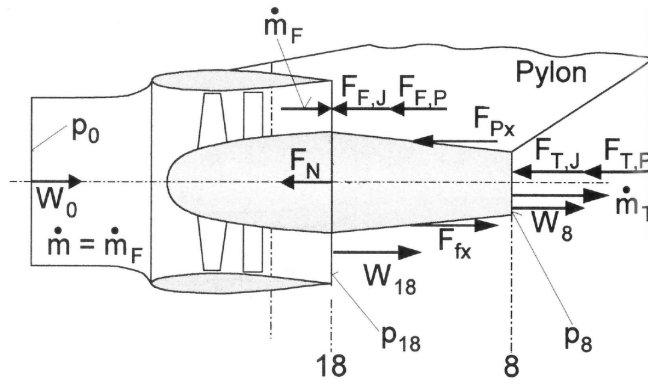


FIGURE A.2: The split between the thrust and drag domains⁵³

The definition of thrust used in the ALVAST study is compatible with the definition of the modified net thrust (F_N^*) used within the current project (Figure A.3). The modified thrust (Equation A.7) is the product of the change of the fan massflow rate ($\dot{m}_F$), mass flow rate of the core provided from the separate feed ($\dot{m}_T$), the gauge pressure force at fan exit (A_{18}) and core exit (A_8), as well as the pressure (F_{Px}) and viscous (F_{Fx}) forces that act on the core-cowl.

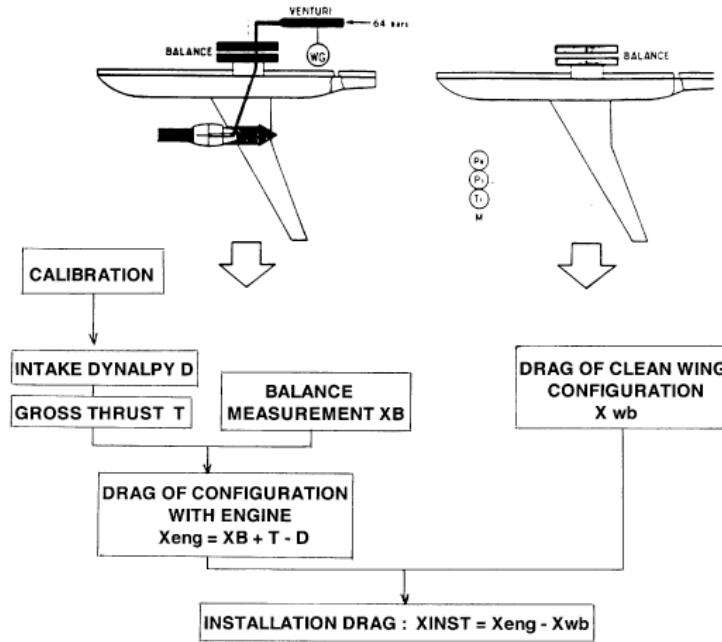
$$F_N^* = \dot{m}_F (w_{18} - w_0) + \dot{m}_T \cdot w_8 + (p_{18} - p_0) A_{18} + (p_8 - p_0) A_8 + F_{Px} + F_{Fx} \quad (\text{A.7})$$

To measure the installation drag of the engine-aircraft configuration, the force along the X axis (X_B) was measured by the balance for the configuration with the engine (Figure A.4). The measured force included the net thrust of the engine (F_N^*), that can be derived from calibration (Equation A.8) as gross thrust ($Thrust$) reduced by the dynamic head ($Dynalpy$). The isolated engine thrust (F_N^*) is added to the measured force to obtain the drag of the configuration with the engine (Equation A.9). It has

FIGURE A.3: Thrust accounting for a Turbine Powered Simulator (TPS) on ALVAST⁵³

to be noted that the interference on thrust (ΔF_N^*) is not accounted for and only the isolated engine thrust was used to obtain the drag of the engine configuration. Finally, the installation drag (X_{inst} , Equation A.10) can be obtained by subtracting the drag of the clean wing configuration (X_{wb}) from the configuration with the engine (X_{eng}). To create a link with the current work, the installation term was expressed in terms quantified with the use of the modified near-field method (Equation A.11). The X_{inst} was expressed in terms of the modified nacelle drag of an isolated engine ($D_{nac iso}^*$) and the sum of interference terms (ΔD_N^* ; $\Delta D_{A/F}$; ΔF_N^*) as in Section 4.4.

DETERMINATION OF THE INSTALLATION DRAG

FIGURE A.4: The experimental procedure for the thrust and drag assessment⁵⁵

$$F_N^* = Thrust - Dynalpy \quad (A.8)$$

$$X_{eng} = X_B + Thrust - Dynalpy = X_B + F_N^* \quad (A.9)$$

$$X_{inst} = X_{eng} - X_{wb} \quad (A.10)$$

$$X_{inst} = -INPF + F_{N \text{ iso}}^* - D_{A/F \text{ iso}} = D_{nac \text{ iso}}^* + \Delta D_{nac}^* + \Delta D_{A/F} - \Delta F_N^* \quad (A.11)$$

APPENDIX B

Engine design frameworks

The engine design process (Section 4.2) was aided by use of two computational frameworks. The cycle optimisation routine Automatch was used for a detailed fine tuning of the engine thermodynamic cycle to the desired characteristics (Section B.1). Based on the obtained cycles the engine aerolines were designed with use of Geometric Engine Modeller Including Nozzle Installation (GEMINI) (Section B.2).

B.1 Cycle optimisation routine - Automatch

To quantify the engine fuel consumption during mission, it is necessary to use engine thermodynamic cycles. Within current research the engine performance code Turbomatch (Section 4.2.1) is used to define the cycles. However, the task of creating and fine tuning the turbomatch models can be time consuming, therefore it was decided to create an automatching routine in Python to accelerate the process.

The Automatch is the python code that integrates the Turbomatch performance code into the genetic algorithm wrapper code. The turbomatch inputs are created automatically for a given engine architecture and the outputs of the performance analysis are automatically read and evaluated based on the assumed objective function. A single objective function has been used to assess the engine and some flexibility in the determination of the objective function was planned within the Automatch. The routine minimises the error objective function to determine the engine cycle, which is closest to meeting all the posed constraints. The typical performance constraints imposed within Automatch are engine massflow rate ($\dot{m}_{tot}$), engine thrust (F_N), engine gross thrust (F_{gross}), overall pressure ratio (OPR) and specific fuel consumption (SFC). The objectives can be set for any of the three engine operating points defined by user and typically these are start of cruise, top of climb and sea level static engine test. Furthermore, a range of variables has to be defined along with the prescribed bounds to narrow down the range of solutions to current technology level. The typical variables in the Automatch are pressure ratios in the IP and HP compressors (IPR , HPR), the

combustor outlet temperature (COT), the auxiliary power output (P_{aux}) and engine by-pass ratio (BPR). If required, each of the variables can be set to be constant. For example, the constant values of BPR or COT were used when optimising the engine cycle to represent an existing aero-engine with a given specification. Based on the set of inputs and initial generation is set with use of random seeding within the bounds. The genetic algorithm evaluates the performance of each individual within the generation and based on the cross-over and mutation procedures creates a new generation of potential solutions. A typical convergence process of current routine takes about 1000 generations, within which 45 individual engine solutions are analysed (Figure B.1). Typically, a good convergence can be determined based on the reach of the plateau for the best solutions over the number of generations, as well as the median of the solutions within each generation that approaches the best solution. There were more than 45000 individual engine performance solutions computed and evaluated based on the objective function. The introduced routine evaluates the multitude of cycles and presents the final optimised cycle within 3 hours. As compared with the typical time of 2-3 working weeks used to determine a single engine cycle, the time saving is substantial.

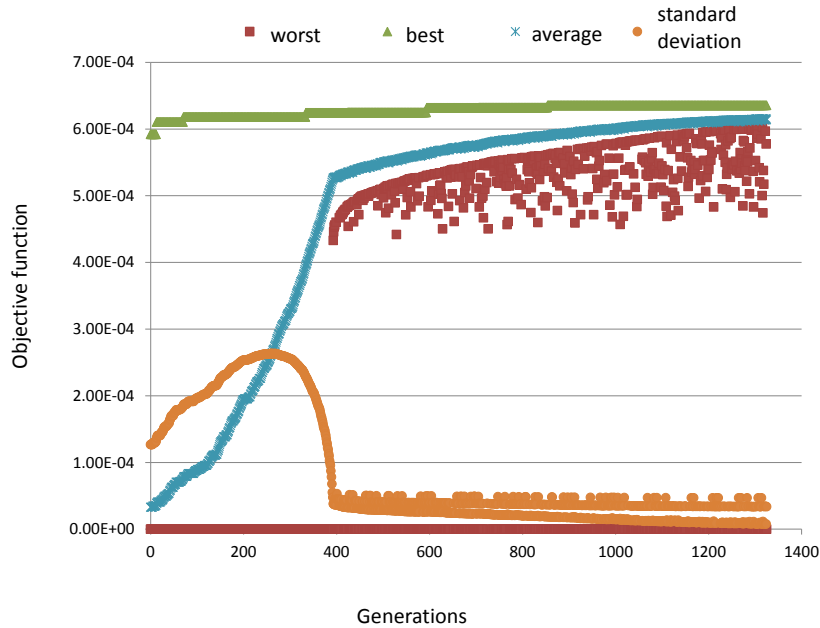


FIGURE B.1: A typical convergence study from genetic algorithm; Plot of best and worst individual within each generation compared with an average result

B.2 GEMINI

Geometric Engine Modeller Including Nozzle Installation (GEMINI) is a framework that enables a rapid design of the external geometry of the modern turbofan engine. The key aerolines of the engine are defined such as the engine intake and spinner geometries,

fan cowl, core cowl, plug, as well as internal aerolines within the by-pass duct and core nozzle. Current project substantially contributed to the development of the tool by creating the set of engine design rules, aiding the implementation of geometry generation^{92,93} and setting up the automated CFD process to evaluate the engine aerodynamic performance^{94,95}. As a result, the GEMINI framework (Figure B.2) was developed to relatively quickly design and evaluate the aerodynamic performance of the aero-engines with separate-jet exhaust.

The design of an engine geometry is based on the performance input (Section B.2.1). The engine key points are computed based on the defined flow areas of the engine fan, as well as inlets and outlets for core and by-pass nozzles. The geometry is completed with use of the Class Shape Transformation (CST) curves^{90,91}. A detailed methodology on geometry generation is presented in Section B.2.2.

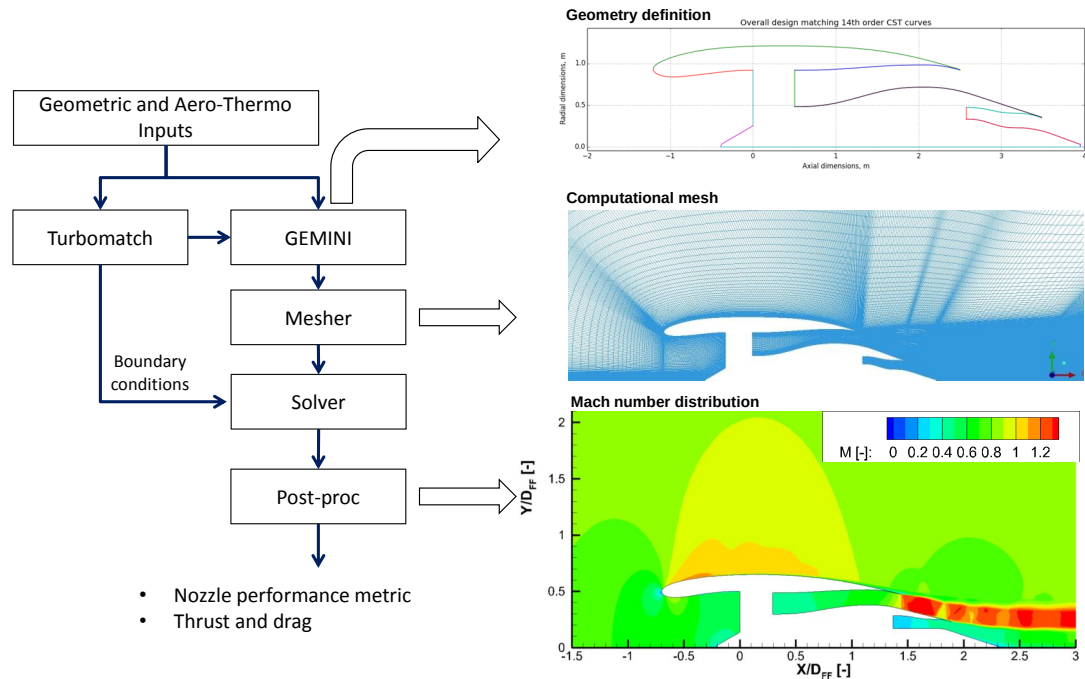


FIGURE B.2: The layout of engine design platform GEMINI based on Goulos et al.⁹⁵

B.2.1 Engine cycle input

The design process for the engine geometry is started by acquiring thermodynamic data from the engine performance code through a dedicated input. Basic information such as the total pressure, total temperature and engine massflows at each engine station are read by the code. The cycle information is read directly from the Turbomatch results file.

To enable a correct assignment of engine station vectors from Turbomatch bricks to the key engine stations such as fan face or nozzle entries, the specific number of Turbomatch stations need to be provided by the user to inform the code about engine architecture. In addition to the cycle input, the user needs to specify a list of Mach number at chosen engine stations such as fan inlet Mach number, fan outlet Mach number and turbine outlet Mach number. Based on the provided information the static pressures and temperatures are computed. The engine fuel-to-air ratio is also read from the turbomatch result, to evaluate the values of specific heat at constant pressure and heat capacity ratio of the gas in the core flow. Eventually, the thermodynamic computation within the GEMINI is concluded with computation of the required flow-capacities within the engine.

B.2.2 Geometry generation

Based on the required flow-capacities, an inverse design approach is employed to determine the axial and radial coordinates of key engine points and obtain a 2D axisymmetric representation of the intake aerolines and exhaust aerolines for the core and by-pass ducts. The aerolines are constructed with use of Class Shape Transformation (CST) curves^{90,91}. The approach inherits the intuitiveness and flexibility of Qin's aerofoil parameterisation method¹¹⁴. However, the applicability of the approach is extended to the parametric geometry definition of intakes and nozzles. In a general form, the CST function $y(x)$ is a product of class function ($C_{N2}^{N1}(x)$) and the shape function ($S(x)$). The CST function can be normalized within the Cartesian space $x \in (0, 1)$ and finally written as in Equation B.1. The normalised coordinates $x = X/L$ and $y = Y/L$ are the radial and axial dimensions normalised by the axial lengthscale, while y_{offset} determines the imposed offset in the radial direction between the curve's end-points in the non-dimensional Cartesian space. The class function ($C_{N2}^{N1}(x)$) defines the baseline aero shape from the leading edge to the trailing edge and it is defined as in Equation B.2. Furthermore, the shape function ($S(x)$) corresponds to the shape perturbation that is imposed on the class function. The shape function is defined as in Equation B.3. It can be observed in Equation B.3 that the shape function $S(x)$ corresponds to the n^{th} order Bernstein polynomial $BP_n = \sum_r^n \sum_{r=0}^n \{K_{r,n} x^r (1-x)^{n-r}\}$ with different A_r for $r = 0, \dots, n$ applied to the respective binomial coefficients $K_{r,n} x^r$. The terms the binomial coefficients are denoted as $K_{r,n}$ and they are expressed as in Equation B.4. Overall, the method offers a considerably detailed way of aeroline control. The variation of weighting coefficients enables a significant perturbation of the shape. Moreover, an example unity shape function is deconstructed (Figure B.3), where the all A_r weighting coefficients were equal unity.

Based on those basic CST principles a more refined methods for nacelle aerolines^{92,93} and exhaust aerolines^{94,95} were developed and are implemented in the GEMINI framework.

$$y(x) = C_{N2}^{N1}(x) \cdot S(x) + x \cdot y_{offset}, x \in (0, 1) \quad (B.1)$$

$$C_{N2}^{N1}(x) = x^{N1} \cdot (1 - x)^{N2}, x \in (0, 1) \quad (B.2)$$

$$S(x) = \sum_{r=0}^n \{A_r K_{r,n} x^r (1 - x)^{n-r}\}, x \in (0, 1) \quad (B.3)$$

$$K_{r,n} = \frac{n!}{r!(n - r)!} \quad (B.4)$$

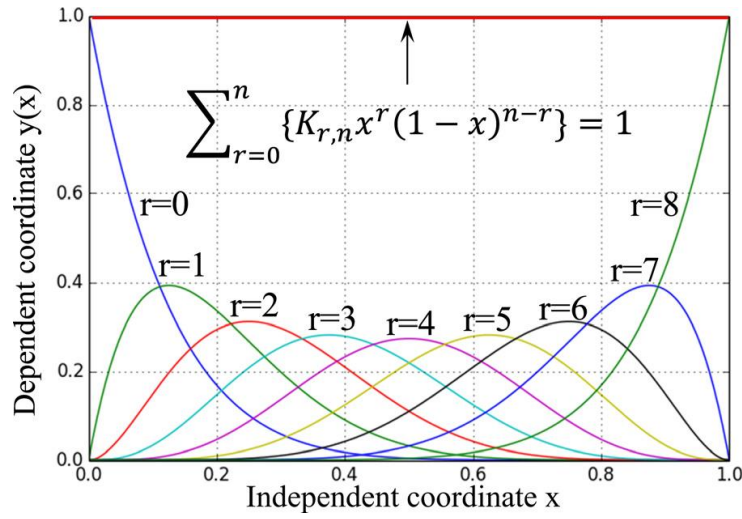


FIGURE B.3: The geometric representation of Bernsein polynomial⁹¹

B.2.3 CFD process

To assess the aerodynamic performance of the designed engines, the GEMINI framework uses 2D axisymmetric Computational Fluid Dynamics (CFD). A coherent numerical methodology was developed for the entire current research (see Chapter 4). For a designed geometry a structured 2D mesh is created. The mesh generation has been automated and it followed the meshing guidelines adopted for the 3D nacelles within current research. In addition to following the internal meshing criteria a range refinement factors dedicated to some areas of the engine were implemented. Following the mesh generation, the GEMINI framework automatically starts the aerodynamic evaluation with means of the CFD software. The key settings of the currently used commercial code (Fluent) are as follows:

- 2D axisymmetric problem

- steady state computation
- density based solver
- Green-Gauss node based gradient method
- second order spatial discretisation for flow and turbulent terms
- Reynolds-Averaged Navier-Stokes (RANS) turbulence model $k - \omega$ *SST*

Based on the computation, the obtained engine designs are assessed based on the overall net propulsive force (NPF) for the engine, as well as, with use of non-dimensional parameters such as drag coefficient ($C_{D\ nac}^*$), thrust coefficient (C_V), the bypass (C_d^{BPD}) and core nozzle (C_d^{Core}) discharge coefficients.

APPENDIX C

Domain and grid studies

C.1 Domain studies

A range of domain studies was carried out to determine the suitable extent of the domain for the isolated nacelle and engine studies. The outcomes of the domain work enabled the establishment of the numerical methods described in Section 4.3.1.

C.1.1 ARA Cows

Initially, the study of isolated nacelles was considered for validation of numerical methods based on the experimental campaign at the ARA (Section 2.1.1). A range of domain sizes was considered with the domain sizes of 60, 90, 120 and 150 of nacelle maximum diameters (D_{max}). A typical nacelle configuration was computed at $M_\infty = 0.85$ and for the range of MFCR from 0.4 to 0.75. The study showed that a domain of $60D_{max}$ was sufficiently large to ensure no spurious effects on the flow field. For example, for Cowl 1, the increase in $C_{D^* nac}$ for a domain of $150D_{max}$ at typical conditions of $M_\infty = 0.85$ and MFCR=0.7 increased the drag coefficient by less than 0.1%.

Domain size	$C_{D^* nac}$
$60D_{max}$	0.039914
$90D_{max}$	0.039918
$120D_{max}$	0.039919
$150D_{max}$	0.039919
Max error	0.01%

TABLE C.1: The results of domain study for the ARA Cowl 1 at M=0.4 and MFCR=0.7125

C.1.2 DSFRN

To determine the impact of the domain on the thrust performance parameters, the domain study with use of the Dual Separate Flow Reference Nozzle (DSFRN) was done in cooperation with Otter et al.¹⁰⁶. The study was done at a FNPR of 2.2 for three domain diameters, $20D_{max}$, $40D_{max}$ and $60D_{max}$, where D_{max} was the maximum diameter on the DSFRN nacelle (Table C.2). Between the computations for $20D_{max}$ and $60D_{max}$ there was a reduction of -0.3% in C_V^* , -0.001% in C_d^{BPD} and a decrease of -0.11% in C_d^{Core} . The results were monotonic with respect to the domain diameter. Overall, the $60D_{max}$ was adopted for further study on the DSFRN (Section 5.1.3).

R_{dom}	C_d^{BPD}	Δ [%]	C_d^{Core}	Δ [%]	C_V^*	Δ [%]
20	0.9842	-0.0010	0.9384	-0.1086	2.6626	-0.2975
40	0.9842	0.0002	0.9382	-0.0833	2.6576	-0.1094
60	0.9842	0.0000	0.9374	0.0000	2.6547	0.0000

TABLE C.2: The domain study for the DSFRN at FNPR=2.2

C.1.3 Isolated engine

To determine the impact of the domain on calculation of the performance parameters of a realistic engine configuration, studies were conducted for the separate-jet configuration. The isolated engine (E1) was simulated with the use of three domains of different size 10, 30 and $50D_{max}$, where D_{max} is the engine maximum diameter. The impact on the engine performance was quantified by the monitoring of the key performance parameters of an isolated engine (Table C.3): discharge coefficient of the by-pass duct (C_d^{BPD}), core duct (C_d^{Core}), thrust coefficient (C_V^*), modified nacelle drag ($C_{D^* nac}$). The C_V^* and $C_{D^* nac}$ are the most important parameters with regards to the determination of the engine net propulsive force. The C_V^* reduced monotonically with the size of the domain and the percentage difference was -0.00176% and -0.00024% for the $10D_{max}$ and $30D_{max}$, respectively, as compared with the domain of $50D_{max}$. Furthermore, the effect of the domain was also assessed for $C_{D^* nac}$ and the results indicated an increase of the $C_{D^* nac}$ with an increase of domain diameter by +0.6% and +1.23% from 10 and 30 to $50D_{max}$, respectively. The non-monotonic behaviour of $C_{D^* nac}$ was explained by a pressure influence from the pressure far-field for the $10D_{max}$ domain. Ultimately, the extent of the domain size was set to 50 nacelle D_{max} for this work.

R_{dom}	C_d^{BPD}	Δ [%]	C_d^{Core}	Δ [%]	C_V^*	Δ [%]	$C_{D^* nac}$	Δ [%]
$10D_{max}$	0.98818	4.50E-03	0.73728	-0.0060	1.01910	-0.00176	0.003601	0.5999
$30D_{max}$	0.98823	3.25E-05	0.73725	-0.0028	1.01908	-0.00024	0.003578	1.2348
$50D_{max}$	0.98823	0	0.73723	0	1.01908	0	0.003622	0

TABLE C.3: Sensitivity of discharge coefficient of the by-pass duct (C_d^{BPD}), core duct (C_d^{Core}), thrust coefficient (C_V^*), modified nacelle drag ($C_{D^* nac}$) of an isolated engine with respect to the domain size (R_{dom})

C.2 Grid convergence

To determine the sensitivity of the numerical solutions to the spatial resolution of the meshes, a grid convergence study was conducted for a range of configurations considered in the research. The mesh studies were completed for the isolated nozzle (Appendix C.2.2), the isolated engine (Appendix C.2.3) and for the aircraft (Appendix C.2.4).

The method advocated by Roache¹⁰⁹ was followed in the mesh studies and it was briefly outlined in this section. Firstly, the ratio of grid refinement r is defined in terms of the grids sizes (N) across two levels of refinement (N_1, N_2) (Equation C.1), where d is the dimensionality and $d = 3$ for 3D problems. Furthermore, the order of computation was defined for the second order accurate code (Equation C.2).

$$r = \left(\frac{N_1}{N_2} \right)^{1/d} \quad (C.1)$$

$$p = 2 \quad (C.2)$$

Richardson's Extrapolation¹⁰⁹ was used to estimate the value of the coefficient in question for an infinitely fine mesh.

$$f_{cor-med} \cong f_{med} + \frac{f_{med} - f_{cor}}{r^p - 1} \quad (C.3)$$

$$f_{med-fine} \cong f_{fine} + \frac{f_{fine} - f_{med}}{r^p - 1} \quad (C.4)$$

The grid convergence indices are then calculated with the use of Equation C.5¹⁰⁹. A safety factor of $F_S = 1.25$ was used due to the use of three levels of grid refinement are considered¹⁰⁹.

$$GCI_{med-fine} = F_S \left| \frac{f_{fine} - f_{med-fine}}{f_{med-fine}} \right| \quad (C.5)$$

$$GCI_{cor-med} = F_s \left| \frac{f_{med} - f_{cor-med}}{f_{cor-med}} \right| \quad (C.6)$$

The grid convergence indicies can then be used to check whether the solutions lie in the asymptotic range and hence satisfy Equation C.7:

$$AR \cong \frac{GCI_{cor-med}}{GCI_{med-fine}} \cong r^p \quad (C.7)$$

C.2.1 DLR-TPS

The CFD capabilities for the simulation of isolated engine configurations with separate exhaust systems were investigated by Vulgaris et al.⁹⁹, based on an open source test case^{32,33}. A set of 2D axisymmetric computation was carried for a range of power settings for the turbine power simulator (TPS). The mesh independence study for the TPS case was conducted with three grids and a constant refinement ratio of 1.5 was used. The medium for a 2D grid has around 1.6×10^3 cells. The obtained GCI aspect ratio was 0.999 for all the surface integrals tested such as force on the core cowl, force on the fan cowl and force on the plug. As a result, it was considered that the asymptotic range was achieved. The separate-exhaust meshing criteria were developed based on the ‘fine’ mesh density and the 3D meshing guidelines were created by introduction of circumferential node distribution.

C.2.2 DSFRN

To determine the grid dependency, a total of three mesh resolutions were investigated at a FNPR of 2.2 such that grid independency could be assessed with Richardson Extrapolation¹⁰⁸ used to calculate the grid independent solution (Table C.4). The first cell height remained fixed across all the meshes investigated. The element count for the three meshes was as follows: 8.6×10^6 , 10.3×10^6 , 14.5×10^6 . A monotonic increase of thrust coefficient (C_V^*) was observed with the mesh refinement, with a GCI of -0.0037%, -0.0014% and -0.0012% for the coarse, medium and fine meshes, respectively. A safety factor of 1.25 was used throughout. To assess the impact of the domain on the evaluation of the losses in the ducts, the discharge coefficients of the DSFRN were computed and the GCI was evaluated (Table C.4). The GCI for C_d^{BPD} was -0.0423% and the GCI for C_d^{Core} -0.5405% with the use of the fine mesh. Overall, it is assessed that the fine mesh resolution presents the necessary accuracy for the numerical studies of engine performance. The results presented in this thesis were computed on the 14.5 million element mesh, which is the fine resolution with the same meshing guidelines as for the isolated engines.

GCI	coarse	medium	fine
C_V^*	-0.0037%	-0.0014%	-0.0012%
C_d^{BPD}	-0.0667%	-0.0531%	-0.0423%
C_d^{Core}	-0.9465%	-0.6789%	-0.5405%

TABLE C.4: Grid Convergence Index (GCI) for the thrust coefficient (C_V^*) and discharge coefficients for the by-pass (C_d^{BPD}) and core (C_d^{Core}) nozzles of the DSFRN at $FNPR = 2.2$, $CNPR = 1.8$

C.2.3 Isolated engine

The sensitivity of the computation of key engine performance parameters to the spatial resolution of the grid was tested for the isolated E1 engine at a typical cruise condition $M_\infty = 0.82$, $AoA_{eng} = 4.25^\circ$, $FNPR = 2.71$, $CNPR = 1.37$. The three mesh densities were considered with 3.02×10^6 , 9.5×10^6 and 33.5×10^6 elements for the “medium”, “fine” and “superfine” mesh resolution, respectively. The GCI was computed (Table C.5) for the thrust coefficient (C_V^*), modified nacelle drag coefficient ($C_{D^* nac}$), as well as for discharge coefficients of the by-pass (C_d^{BPD}) and core (C_d^{Core}). The C_V reduced monotonically with the mesh size and for the fine mesh the GCI equaled 0.0288%. A satisfactory result of the GCI was also observed for the C_d^{BPD} with -0.0234% and C_d^{Core} with -1.6553% for the fine mesh. The value of GCI for the discharge coefficients reduced monotonically with the increase of mesh size. Meanwhile, the GCI for the $C_{D^* nac}$ equaled +8.2005%, the GCI reduced monotonically with mesh size and the GCI for the superfine mesh equaled +3.5468%. However, due to the very large size of the superfine mesh, it was not possible to incorporate the superfine mesh into the studies. The use of medium size meshes for the engine studies was considered unacceptable with a GCI for $C_{D^* nac}$ in order of +19.5813%. Overall, the fine mesh was used within the study, as the GCI was considered acceptable for the preliminary design studies.

GCI	medium	fine	superfine
$C_{D^* nac}$	19.5813%	8.2005%	3.5468%
C_V^*	0.0774%	0.0288%	0.0124%
C_d^{BPD}	-0.0757%	-0.0234%	-0.0101%
C_d^{Core}	-3.5990%	-1.6553%	-0.7159%

TABLE C.5: Grid Convergence Index (GCI) for the thrust coefficient (C_V^*), modified nacelle drag coefficient ($C_{D^* nac}$) and discharge coefficients for the by-pass (C_d^{BPD}) and core (C_d^{Core}) nozzles of the isolated E1 engine at $M_\infty = 0.82$, $AoA_{eng} = 4.25^\circ$, $FNPR = 2.71$, $CNPR = 1.37$

C.2.4 Common Research Model

The outcomes described in this Section affect the definition of the meshing strategy (Section 5.2.1) implemented in the research. To quantify the grid convergence, the

method advocated by Roache et al.¹⁰⁸ was followed for the CRM in WBT0 configuration at a typical cruise condition $M_\infty = 0.85$ and $AoA = 2.5^\circ$ for four mesh resolution (Table C.6).

Configurations	“coarse”	“medium”	“fine”	“superfine”
CRM WB	6.872	10.223	14.711	x
CRM WBNP	15.731	22.477	29.712	x
CRM WBT0	6.937	10.655	16.479	24.663

TABLE C.6: The mesh size in million elements for chosen CRM⁴⁶ configurations

The Richardson Extrapolation¹⁰⁹ was conducted to estimate the grid independent solution. The aircraft drag coefficient $C_{D\ tot}$ reduced monotonically with increasing mesh size (Table C.7). Using a factor of safety of 1.25, the second order grid convergence index (GCI) at $AoA_{A/C} = 2.5^\circ$ was 6.54%, 5.99%, 4.06% and 3.05% for the coarse, medium, fine and superfine solutions, respectively. Similarly, the GCI for the lift coefficient $C_{L\ tot}$ (Table C.8) at $AoA_{A/C} = 2.5^\circ$ was 2.18%, 3.19%, 4.83% and 3.63% for the coarse, medium, fine and superfine solutions, respectively. It was concluded that the “fine” mesh resolution is acceptable and the “superfine” mesh resolution presented too high computational cost for current standards.

	$GCI\ C_{D\ tot}$			
$AoA_{A/C} [^\circ]$	coarse	medium	fine	superfine
0.0	5.24%	4.28%	-1.37%	-1.03%
0.5	7.84%	6.90%	1.43%	1.08%
1.0	6.18%	5.61%	1.67%	1.25%
1.5	4.11%	3.57%	0.45%	0.34%
2.0	13.59%	12.82%	11.03%	8.28%
2.5	6.54%	5.99%	4.06%	3.05%

TABLE C.7: Grid Convergence Index (GCI) for the drag coefficient ($C_{D\ tot}$) for the clean-wing with a tail (WBT0) at a range of aircraft angle of attack ($AoA_{A/C}$); $M_\infty = 0.85$, $Re = 5 \times 10^6$

	$GCI\ C_{L\ tot}$			
$AoA_{A/C} [^\circ]$	coarse	medium	fine	superfine
0	11.99%	15.72%	16.22%	12.18%
0.5	14.15%	17.01%	17.42%	13.09%
1	13.54%	15.51%	15.78%	11.86%
1.5	8.52%	10.16%	9.77%	7.34%
2	5.70%	6.98%	6.95%	5.22%
2.5	2.18%	3.19%	4.83%	3.63%

TABLE C.8: Grid Convergence Index (GCI) for the lift coefficient ($C_{L\ tot}$) for the clean-wing with a tail (WBT0) at a range of aircraft angle of attack ($AoA_{A/C}$); $M_\infty = 0.85$, $Re = 5 \times 10^6$

Furthermore, the WB configuration was tested at a lower Mach number $M_\infty = 0.83$ over the range of angles of attack (Table C.9). The noted GCI at a typical cruise incidence for the fine mesh was 5.6%, whereas for the full range of incidence it did not exceed 9% for the fine mesh.

$AoA_{A/C}$ [°]	$GCI C_{L tot}$			$GCI C_{D tot}$		
	coarse	medium	fine	coarse	medium	fine
0.0	-2.19%	0.39%	0.30%	14.02%	11.85%	9.09%
0.5	-1.92%	0.12%	0.10%	13.06%	11.20%	8.59%
1.0	-	-	-	-	-	-
1.5	-1.41%	0.17%	0.13%	10.69%	9.38%	7.20%
2.0	-1.13%	0.36%	0.27%	9.62%	8.54%	6.56%
2.5	-1.36%	0.10%	0.08%	8.10%	7.30%	5.60%
3.0	-2.99%	-1.46%	-1.12%	5.32%	5.00%	3.84%
3.5	-5.78%	-3.84%	-2.95%	0.62%	1.16%	0.89%
4.0	-8.19%	-6.28%	-4.82%	-3.44%	-2.37%	-1.82%
4.5	-7.91%	-6.21%	-4.77%	-3.85%	-2.68%	-2.05%
5.0	-0.77%	0.42%	0.32%	10.29%	11.56%	8.87%

TABLE C.9: Grid Convergence Index (GCI) for the lift and drag coefficients ($C_{L tot}$; $C_{D tot}$) for the clean-wing (WB) configuration of CRM at a range of aircraft angle of attack ($AoA_{A/C}$); $M_\infty = 0.82$, $Re = 5 \times 10^6$

Considering the WBNP configuration tested at a Mach number of $M_\infty = 0.83$ over the range of angles of attack (Table C.10), the noted GCI at a typical cruise incidence for the fine mesh was 2.3%, whereas for the full range of incidence it did not exceed 5% for the fine mesh. The improvement in the GCI values for the WBNP as compared with the WB configuration was a result of additional fuselage refinement caused by the through-flow nacelle blocking.

$AoA_{A/C}$ [°]	$GCI C_{L tot}$			$GCI C_{D tot}$		
	coarse	medium	fine	coarse	medium	fine
0.0	3.21%	6.68%	5.27%	9.36%	5.45%	4.30%
0.5	1.84%	4.54%	3.58%	7.74%	4.61%	3.64%
1.0	0.74%	2.98%	2.35%	7.02%	4.60%	3.62%
1.5	0.17%	2.17%	1.71%	5.76%	3.80%	2.99%
2.0	-0.33%	1.52%	1.20%	4.78%	3.18%	2.50%
2.5	-0.75%	0.90%	0.71%	4.19%	2.92%	2.30%
3.0	-2.03%	-0.30%	-0.23%	2.60%	1.62%	1.27%
3.5	-3.91%	-1.90%	-1.50%	0.11%	0.01%	0.01%
4.0	-4.19%	-1.81%	-1.43%	-1.60%	-0.93%	-0.73%
4.5	-2.45%	-0.16%	-0.13%	-1.81%	-1.00%	-0.78%
5.0	-0.76%	1.20%	0.95%	-1.64%	-1.01%	-0.79%

TABLE C.10: Grid Convergence Index (GCI) for the lift and drag coefficients ($C_{L tot}$; $C_{D tot}$) for the CRM configuration with a through flow nacelle (WBNP) at a range of aircraft angle of attack ($AoA_{A/C}$); $M_\infty = 0.82$, $Re = 5 \times 10^6$

As a result of the study, the fine meshes were used for the computations.

C.3 Iterative convergence

The iterative convergence was monitored during the computation based on the scaled residuals for all the computed cases, as described in the methodology (Section 4.3.5). Typically, the scaled residuals for the continuity and momentum terms leveled below 10^{-4} during the second-order computation (Figure C.1). The residual of the turbulent kinetic energy (k) leveled below 10^{-5} , whereas the convergence of the specific turbulence dissipation rate (ω) leveled below 10^{-3} . To check the impact of the convergence on the aerodynamic forces, the rearward-oriented force on all the walls was monitored. The typical oscillation in the rearward force was 0.001 aircraft drag count (Figure C.2) and that corresponds to approximately 0.0004% of the total aircraft drag or 0.003% of the isolated engine drag. Overall in the project, the amplitude of the oscillation did not exceed an equivalent of 0.2 aircraft drag counts for all of the computations, which is 0.1% of total aircraft drag. The massflow convergence was monitored and the pressure outlet and the typical oscillation did not exceed 0.004% of the total massflow. Similarly, the convergence for the pressure inlets converged with no oscillation greater than approximately 0.004% of the respective massflows.

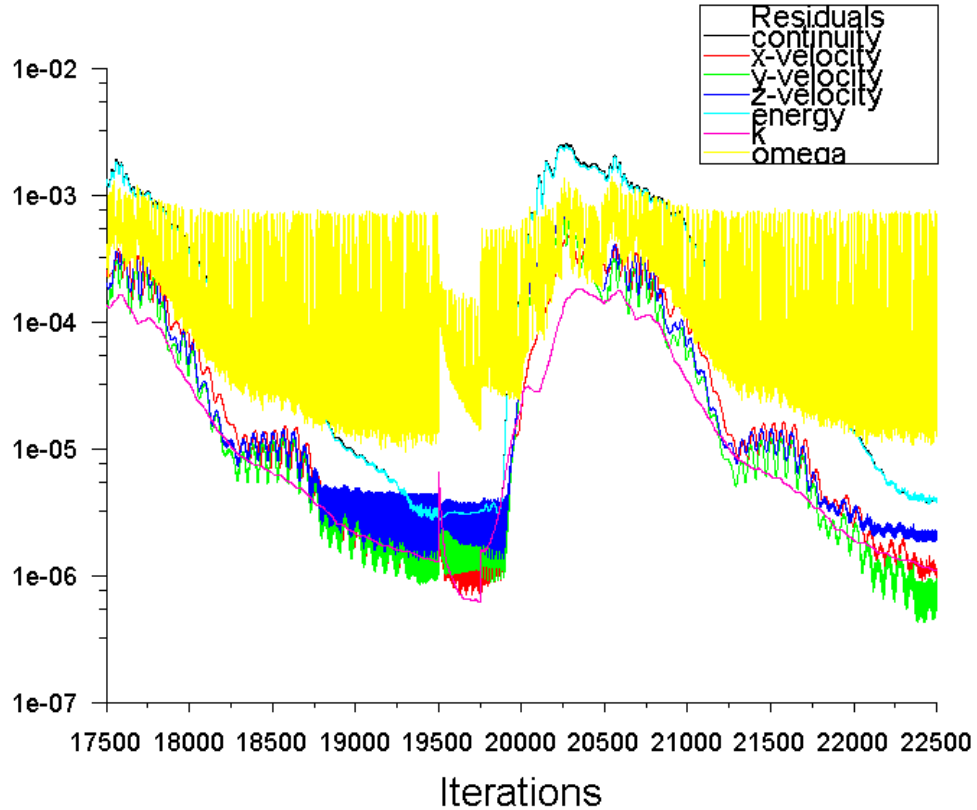


FIGURE C.1: A sample iterative convergence for engine-aircraft configuration

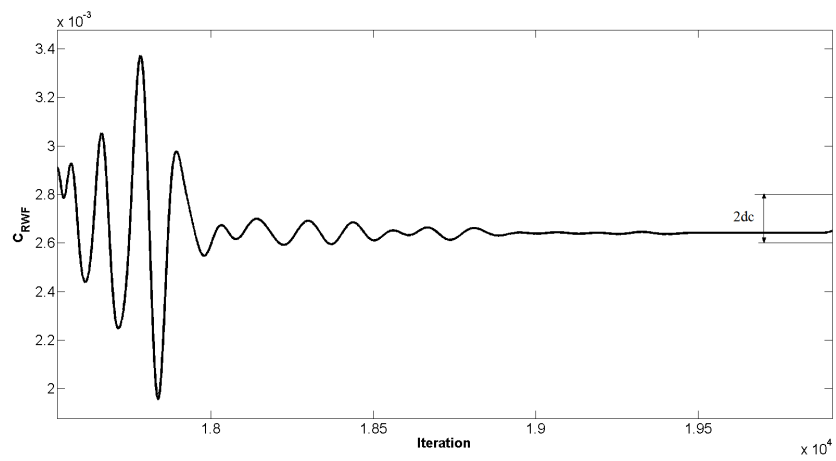


FIGURE C.2: A sample iterative convergence of the rearward oriented force (C_{RWD}) for engine-aircraft configuration

APPENDIX D

Detailed boundary conditions

This chapter contains information about additional boundary conditions that were used for the off-design performance of the engines.

D.1 Boundary conditions for the flight Mach number study

The completion of the sensitivity study to the flight Mach number (M_∞) required the operation of the engine off-design. It is impossible to maintain the relation between all the operating conditions of the engine and aircraft during the variation of flight Mach number. Therefore, it was decided to maintain the constant intake mass flow capture ratio (MFCR) to minimise the changes to the spillage drag and maintain the most comparable conditions within the drag domain. Simultaneously, it was not possible to match the thrust of the engine with the drag of the aircraft, thus the engine was operated independently based on the performance model and the power setting dictated by the MFCR. As a result of the engine performance calculation for the E2 engine, the boundary conditions in terms of MFCR, fan nozzle pressure ratio (FNPR) and core nozzle pressure ratio (CNPR) were determined (Table D.1)

M_∞	TET	$MFCR$	$FNPR$	$CNPR$
0.7	1480	0.75	1.80	1.37
0.73	1527	0.75	1.88	1.43
0.76	1577	0.75	1.97	1.51
0.78	1610	0.75	2.03	1.57
0.8	1642	0.75	2.09	1.63
0.82	1698	0.75	2.15	1.69
0.83	1692	0.75	2.18	1.73
0.84	1710	0.75	2.21	1.77
0.85	1729	0.75	2.24	1.813
0.86	1745	0.75	2.27	1.84
0.87	1762	0.75	2.30	1.883
0.88	1779	0.75	2.33	1.92
0.89	1795	0.75	2.36	1.96
0.9	1812	0.75	2.40	2.00

TABLE D.1: Engine boundary conditions for the E2 engine for a range of flight Mach numbers (M_∞)

APPENDIX E

Additional results

E.1 Validation of drag methods for isolated nacelles

A substantial amount of work dedicated to the validation of numerical methods for isolated nacelles was done outside of the scope of current project. Robison et al¹¹⁵ expanded the validation activity (Section 5.1) and increased the level of confidence that is attributed to the utilised numerical methods. The numerical setup for the work of Robison et al¹¹⁵ is consistent with the methods used in the current project. Overall, the validation of the numerical methods was done for the available nacelle geometries (Section 2.1.1). As a result, the ARA²⁴ (see Appendix E.1.1) were tested numerically for the experimental conditions (Table E.1). The same corrections for the friction drag of the sting were used in the validation exercise in the current research (Section 5.1.1).

E.1.1 ARA

Experimental data²⁴ for ARA cowls were reported including the viscous drag of $0.12L_{nac}$ of cylindrical sting section. Robison et al¹¹⁵ proposed that the drag for nacelle and sting will be referred to as adjusted drag coefficient ($C_{D'}$). The adjusted drag coefficient $C_{D'}$ predicted by the CFD was within 4% of the experimental data prior to drag rise (Figure E.1 and Table E.2). The accuracy of $C_{D'}$ determination was $\Delta C_D = -0.0004$ at MFCR=0.6 and $\Delta C_D = -0.0015$ at MFCR=0.7 for $M_\infty = 0.8$. Furthermore, it is the nacelle modified drag coefficient ($C_{D^* nac}$) that is of interest within the study, as no sting is present in real life application. The $C_{D^* nac}$ presents a fairly constant offset of $\Delta C_D = 0.003$ within the considered range of Mach numbers from 0.7 to 0.9 as compared

Geom	Mach	MFCR	Re	AoA
ARA	0.65-0.95	0.4-0.78	$1.5 - 2.2 \times 10^6$	0°

TABLE E.1: Isolated nacelle configurations in validation study

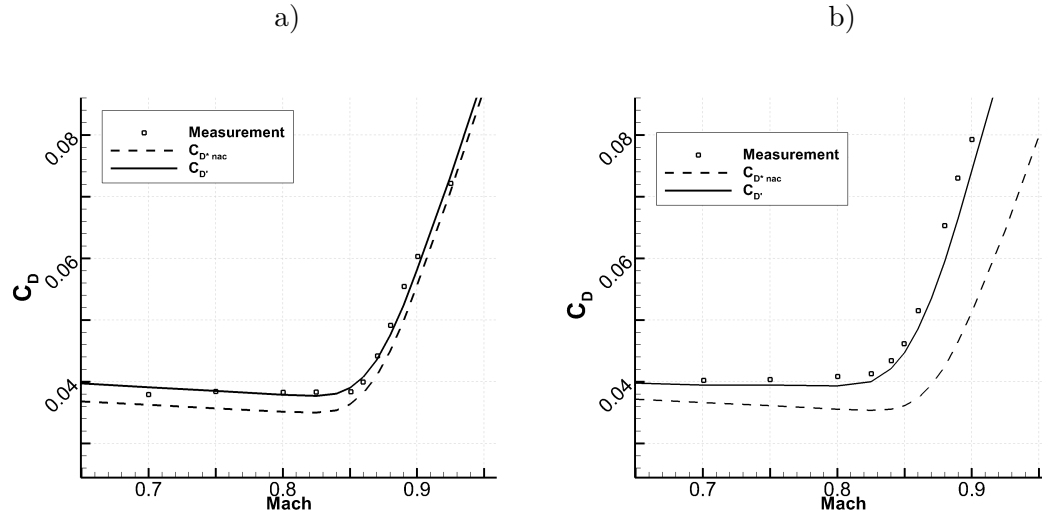


FIGURE E.1: Drag coefficient ($C_{D'}$) as a function of freestream Mach number for ARA Cow1 for CFD analysis¹¹⁵ and experimental data²⁴ at a) MFCR=0.6; b) MFCR=0.7

with the $C_{D'}$ due to the presence of the sting. The difference is predominantly the result of viscous drag on the surface of the sting. Overall, the applied correction for the sting explained the difference between the modified nacelle drag ($C_{D'}$) and the measurements.

MFCR	Dataset	$M_\infty = 0.7$	$M_\infty = 0.8$	$M_\infty = 0.9$
0.6	Measurements	0.03796	0.03829	0.06024
	$C_{D^* \text{ nac}}$	0.03625	0.03514	0.05501
	$C_{D'}$	0.03912	0.03788	0.05832
0.7	Measurements	0.04023	0.04088	0.07971
	$C_{D^* \text{ nac}}$	0.03665	0.03553	0.05163
	$C_{D'}$	0.03948	0.03937	0.0747

TABLE E.2: Effect of sting corrections on the evalation of the nacelle drag

E.2 DSFRN - validation activity

A satisfactory agreement between numerical results by Otter¹⁰⁶ and the experimental data for the Dual Separate Flow Reference Nozzle (DSFRN) from the 2nd PAW³⁵ was noted (Table E.3).

C_V					C_d^{BPD}			C_d^{Core}		
FNPR	CNPR	EXP	CFD	$\Delta C(\%)$	EXP	CFD	$\Delta C(\%)$	EXP	CFD	$\Delta C(\%)$
1.40	1.17	0.984	0.985	-0.064	0.968	0.962	0.573	0.807	0.805	0.236
1.60	1.33	0.987	0.987	-0.034	0.974	0.968	0.580	0.861	0.858	0.359
1.80	1.50	0.988	0.989	-0.018	0.981	0.975	0.556	0.890	0.887	0.380
2.00	1.67	0.990	0.990	-0.038	0.987	0.982	0.503	0.910	0.907	0.333
2.20	1.83	0.990	0.990	-0.041	0.989	0.984	0.458	0.943	0.938	0.560
2.40	2.00	0.991	0.992	-0.087	0.989	0.985	0.434	0.944	0.937	0.758
2.60	2.16	0.991	0.991	-0.056	0.989	0.985	0.431	0.941	0.933	0.831
2.80	2.33	0.988	0.989	-0.040	0.990	0.985	0.439	0.966	0.963	0.316

TABLE E.3: Summary of results from DSFRN computations¹⁰⁶ and comparison to experimental³⁴ nozzle performance

E.3 Additional results on the engine installation

This section contains the analysis of additional computational results of engine-airframe configuration. The analysis of the installation effect for the geometrically enlarged engines was located in Appendix E.3.1. The results of geometrically scaled engine were used in the development of the reduced order model for the effect of engine installation (Section 7.1). Furthermore, the E2 engine was compared with the enlarged version of E1 engine to isolated the effect of low specific thrust (Appendix E.3.2).

E.3.1 The effect of size with constant efflux

To isolate the effect of size from the effects of the change in cycle and exhaust design, the geometrically enlarged versions of E1 engine were considered in position C3 (Figure E.2a). The computation was completed for engines with 1.23 and 1.45 larger fan diameter. The engines were operated with the constant mass flow capture ratio to maintain the same spillage condition between the engines, as well as with constant efflux (total pressure and total temperature) to maintain the shape of the post-exit stream tube constant. The constant efflux of the engines resulted in unrealistically large thrust for the larger engines due to proportionally larger massflow, however, the key interest of the exercise lay in the determination of the mutual interference effects between the larger engines and the airframe. The presence of the larger engine visibly affects the wing aerodynamics. As compared with the E1 engine (Figure E.2b), the wing with 1.23xE1

engine (Figure E.2c) exhibits reduced local Mach number on the upper side of the wing, as well as a modest deceleration on the lower side of the wing. The installation of the largest 1.45xE1 engine resulted in further deceleration of the flow over the upper and lower side of the wing as compared with the other two engines.

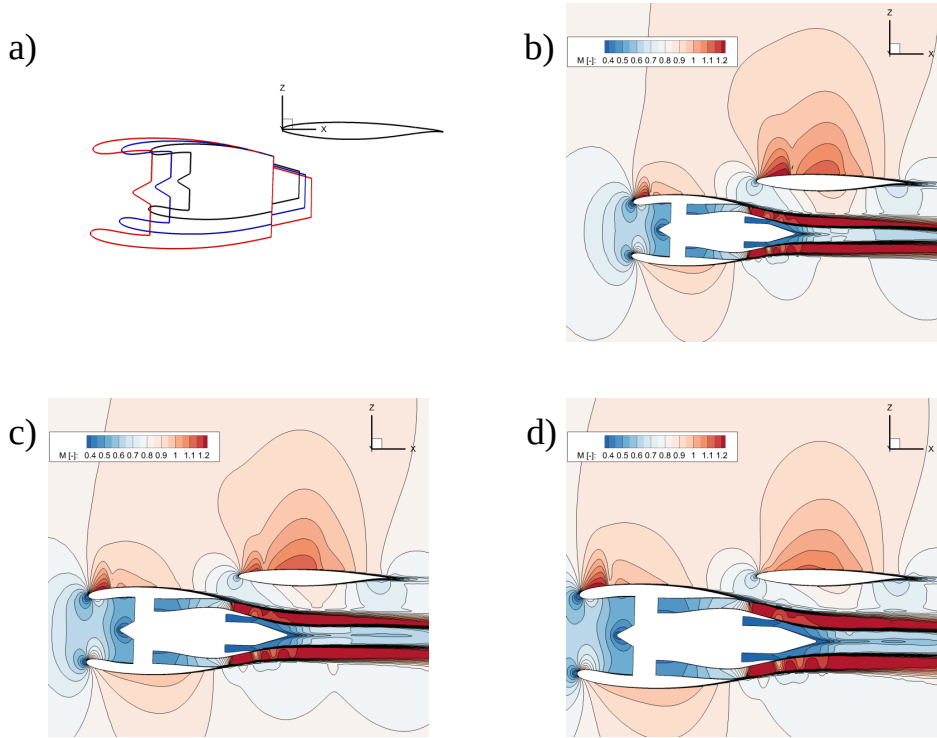


FIGURE E.2: a) Comparison between three size engines and contours of Mach number (M) for b) E1 engine c) 1.23xE1 engine and d) 1.45xE1 engine; $M_\infty = 0.82$; $AoA_{A/C} = 2.5^\circ$

As anticipated, the presence of the large engines affected the lift-drag polar (Figure E.3). The total aircraft drag ($C_{D\ tot}$) increased with the size of the engine. At a typical cruise lift coefficient of 0.5 (Table E.4), the $C_{D\ tot}$ increased from 224dc for the clean-wing configuration to 252.8dc, 262.4dc and 281.3dc for the E1, 1.23xE1 and 1.45xE1 engines, respectively. At lower values of lift coefficient ($C_{L\ tot}$), the drag penalty for the larger engines 1.23xE1 and 1.45xE1 is less notable and it is 5.9dc and 20.2dc as compared with the E1 configuration at $C_{L\ tot} = 0.3$. To quantify the penalty for the presence of the engine as compared with the clean-wing configuration, the installation drag at constant lift ($C_{D\ inst}$) was considered for all three configurations (Figure E.4). At a typical cruise condition ($C_{L\ tot} = 0.5$; $AoA_{A/C} \approx 2.5^\circ$), the $C_{D\ inst}$ for the E1 engine is 28.9dc and it increased with size to 38.4dc and 56.3dc for the 1.23xE1 and 1.45xE1 sizes, respectively. The installation drag ($C_{D\ inst}$) depends on the aircraft incidence and for the E1 configuration it increases from the 20dc at $AoA_{A/C} = 0^\circ$ to 30dc at the peak value at $AoA_{A/C} = 3^\circ$. A further increase of aircraft incidence causes a notable

	$C_{D\ tot} [dc]$			
$C_{L\ tot} [-]$	Clean	E1	1.23xE1	1.45xE1
0.2	144.3	163.8	167.6	179.2
0.3	161.7	184.9	190.8	205.1
0.4	187.7	214.4	222.3	239
0.5	223.9	252.8	262.4	281.3

TABLE E.4: Total aircraft drag ($C_{D\ tot}$) at chosen total lift coefficients ($C_{L\ tot}$) for three engine sizes E1, 1.23xE1 and 1.45xE1

reduction of the installation penalty to 0dc at $AoA_{A/C} = 4^\circ$. The peak penalty is a result of the additional drag that occurs as a result of re-trimming the aircraft to the same lift coefficient as the clean-wing configuration, whereas the rapid reduction of the penalty is the result of the increasing with incidence contribution of the lift from the nacelle. The relation is broadly similar for the larger engines with the $C_{D\ inst}$ of 23dc and 35dc at $AoA_{A/C} = 0^\circ$, that increased to the peak value of 40dc at $AoA_{A/C} = 3.25^\circ$ and 60dc at $AoA_{A/C} = 3.5^\circ$ for the 1.23xE1 and 1.45xE1, respectively.

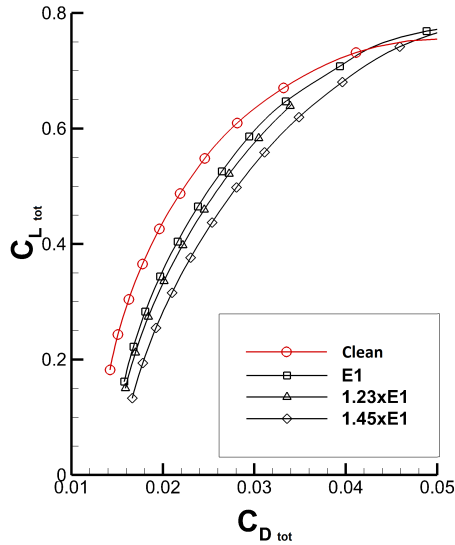


FIGURE E.3: Lift-drag polar for three engine sizes E1, 1.23xE1 and 1.45xE1 in position C3; $Re = 45 \times 10^6$; $M_\infty = 0.82$

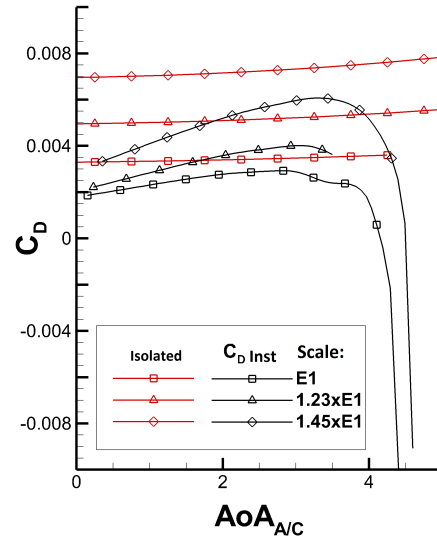


FIGURE E.4: The engine installation drag at constant lift ($C_{D\ inst}$) as function of aircraft incidence ($AoA_{A/C}$) for three engine sizes E1, 1.23xE1 and 1.45xE1; all in position C3; $Re = 45 \times 10^6$; $M_\infty = 0.82$

The presence of the engine causes a reduction of the total lift ($C_{L\ tot}$; Figure E.5). The loss of lift ($\Delta C_{L\ tot}$) is mostly constant with the aircraft incidence ($AoA_{A/C}$) for the range of $AoA_{A/C}$ from 0° to 3° . For the baseline engine E1, the $\Delta C_{L\ tot}$ equals -0.02 and the increase of engine size resulted in a greater loss of -0.03 and -0.05 for 1.23xE1 and 1.45xE1 engines, respectively. The greatest $\Delta C_{L\ tot}$ for all three engine sizes was noted at $AoA_{A/C} = 3.5^\circ$ and it equaled -0.025, -0.035 and -0.055 for the E1, 1.23xE1 and 1.45xE1

engines. Based on the results for the E1 and 1.45xE1, it was determined that further increase of $AoA_{A/C}$ resulted in beneficial effect and neutralisation of the loss of lift. It is a result of appearance of the maximum wing loading at a higher aircraft incidence for the engine-airframe configurations as compared with the clean-wing configurations, as well as the increasing with incidence contribution of the lift from the nacelle. Primarily, the reduction of the total lift ($C_{L\ tot}$) is the result of the loss of lift on the wing (Figure E.6). Based on the spanwise distribution of the lift coefficient it is noted that the presence of the engine caused a notable reduction inboard of the nacelle. The reduction of the lift is monotonic with the size of the engine. At a inboard position of $y/(b/2) = 0.2$, the local lift coefficient for the clean-wing configuration of 0.65 reduced due to the presence of the engine to 0.61, 0.58 and 0.55 for E1, 1.23xE1 and 1.45xE1, respectively. The local lift coefficient (Figure E.6) appeared to be constant with the spanwise position inboard of the engine ($y/(b/2) < 0.3$), however, beyond the $y/(b/2) = 0.35$ there is a gradual reduction of the local lift. As the spanwise position increased the lift distribution for the wing in the presence of the engines tend asymptotically to the lift distribution of the clean-wing. At $y/(b/2) = 0.6$, the local lift coefficient reduced to 0.49, 0.48 and 0.47 for E1, 1.23xE1 and 1.45xE1, as compared with 0.5 for the clean-wing. Beyond $y/(b/2) = 0.8$, the difference between the engines becomes negligible.

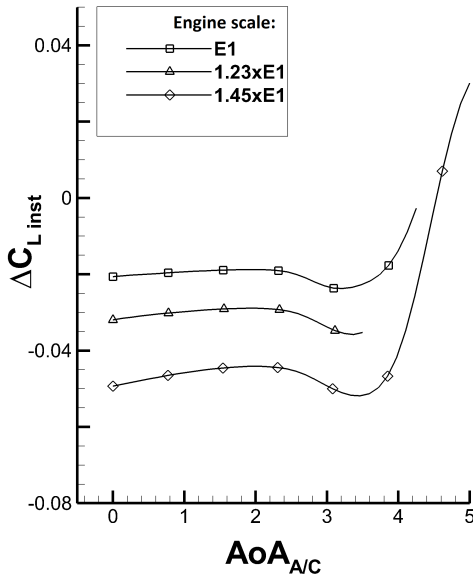


FIGURE E.5: Loss of lift ($\Delta C_{L\ tot}$) for three engine sizes E1, 1.23xE1 and 1.45xE1 in position C3; $Re = 45 \times 10^6$; $M_\infty = 0.82$

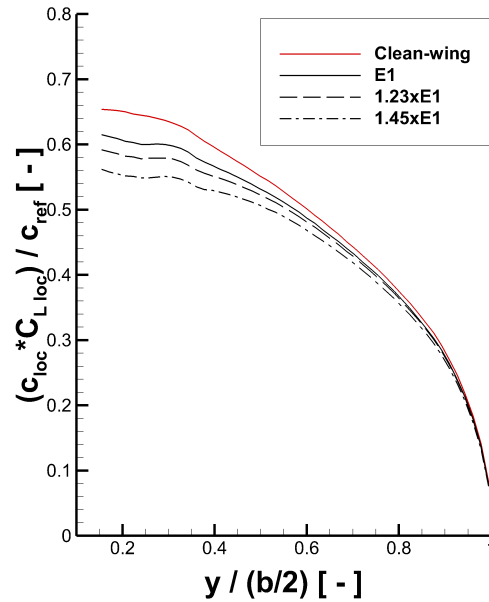


FIGURE E.6: The spanwise lift distribution over the wing of the clean-wing CRM compared with the distribution for the wing in presence of engine in one of three sizes E1, 1.23xE1 and 1.45xE1; all in position C3; $Re = 45 \times 10^6$; $M_\infty = 0.82$; $AoA_{A/C} = 2.5^\circ$

The reduction of the lift of the wing is clearly caused by the change in the pressure distribution on the wing (Figure E.7). Inboard of the engine ($y/(b/2)=0.2$), the presence

of the engine caused a modest acceleration of the flow over the upper side of the wing. The peak suction increased by ΔC_p of -0.15, -0.1 and 0.0 for the E1, 1.23xE1 and 1.45xE1 engines respectively, as compared with the clean-wing. The presence of the engine resulted in stronger shock, which position was shifted upstream as compared with the clean-wing by $\Delta x/c$ of -0.2, -0.225 and -0.25 for the E1, 1.23xE1 and 1.45xE1 engines, respectively. On the lower side of the wing, the presence of the engines caused an

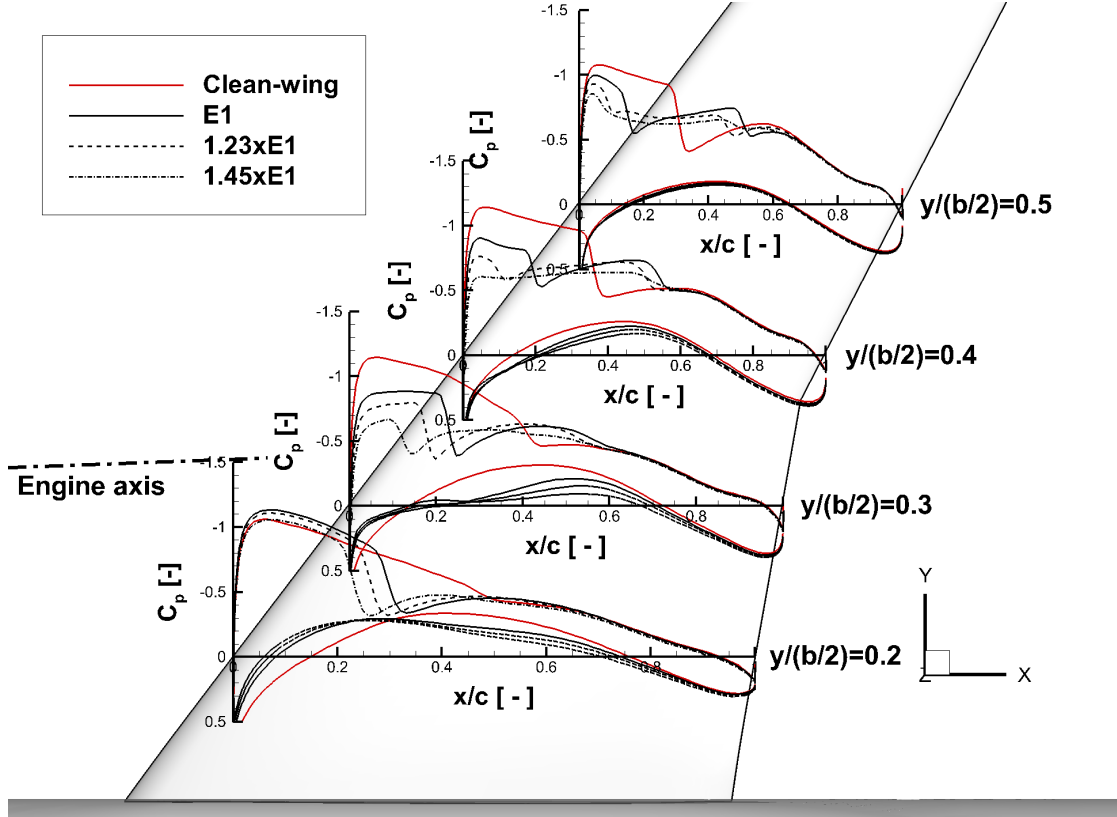


FIGURE E.7: Pressure distribution (C_p) at the wing cross-sections of the wing with $y/(b/2)$ of 0.2, 0.3, 0.4 and 0.5 for the clean-wing and the CRM with three sizes of the E1 engine: E1, 1.23xE1 and 1.45xE1; $M_\infty = 0.82$; $AoA_{A/C} = 2.5^\circ$

acceleration of the flow at the wing leading edge of approximately $\Delta x/c = -0.3$ at $x/c = 0.1$ followed by a reduction of the peak acceleration at the profile maximum thickness at $x/c = 0.5$. However, a modest difference between the engine sizes is observed. Close to the engine installation ($y/(b/2)=0.3$; Figure E.7), the presence of the engine notably affected the upper side of the wing by causing a reduction of local incidence. The reduction of the suction peak at the wing leading edge ($x/c = 0.1$) was ΔC_p of +0.3, +0.4 and +0.5 for the engine sizes of E1, 1.23xE1 and 1.45xE1. The shock position was moved upstream by approximately $\Delta x/c$ of -0.15, -0.2 and -0.25 for the E1, 1.23xE1 and 1.45xE1 engines, respectively. On the lower side of the wing and as compared with the clean-wing, there is a comparable acceleration of $\Delta C_p = +0.2$ for all three engines due to broadly the same shape of the gully. However, further downstream at $x/c = 0.55$, the peak acceleration for the engine configurations reduced as compared with the clean-wing ΔC_p and it was +0.1, +0.15 and +0.2 for the E1, 1.23xE1 and 1.45xE1, respectively.

As for the further outboard sections ($y/(b/2)=0.4$; Figure E.7), the effect of engine installation is predominantly visible on the upper side of the wing. The presence of the baseline engine E1 caused a reduction in the peak suction ΔC_p of $+0.2$ at the leading edge followed by a re-acceleration at $x/c = 0.5$ with a modest shocklet ΔC_p of -0.3 , as compared to clean-wing. The presence of the larger engine $1.23 \times E1$ resulted in a broadly similar two shocklet pressure distribution with rather weaker suction at the leading edge ΔC_p of $+0.35$, as compared with clean-wing. Finally, the presence of the largest engine resulted in a flat profile of the pressure distribution with a $C_p = -0.5$ over the range of x/c from 0.05 to 0.5 . The noted drop in the peak suction as compared with the clean-wing configuration was $\Delta C_p = +0.6$ at $x/c = 0.1$.

To investigate the impact of engine size on the installation effects, a more detailed single point analysis at cruise condition was performed based on the performance of isolated engines and the interference terms (ΔF_N^* , ΔD_{nac}^* , $\Delta D_{A/F}$) (Figure E.8). The cruise operating point was chosen at the steady state flight condition of a superposition model of the baseline engine. The overall lift coefficient obtained is $C_{z'} = 0.5$ at $M_\infty = 0.82$. To isolate the effect of the geometric scaling, constant values of the total pressure and total temperature were used for the jet regardless of the engine size. As a result the standard net thrust increases broadly proportionally with engine fan area. The aim of the comparison is to determine the magnitude of interference effects for different size engines. Therefore, the results for an engine of each size were non-dimensionalised by a single value of reference thrust $F_{N \text{ ref}} = 54028 \text{ N}$ of the baseline engine (E1) at the cruise condition. It was determined that the modified standard net thrust (F_N^*) was

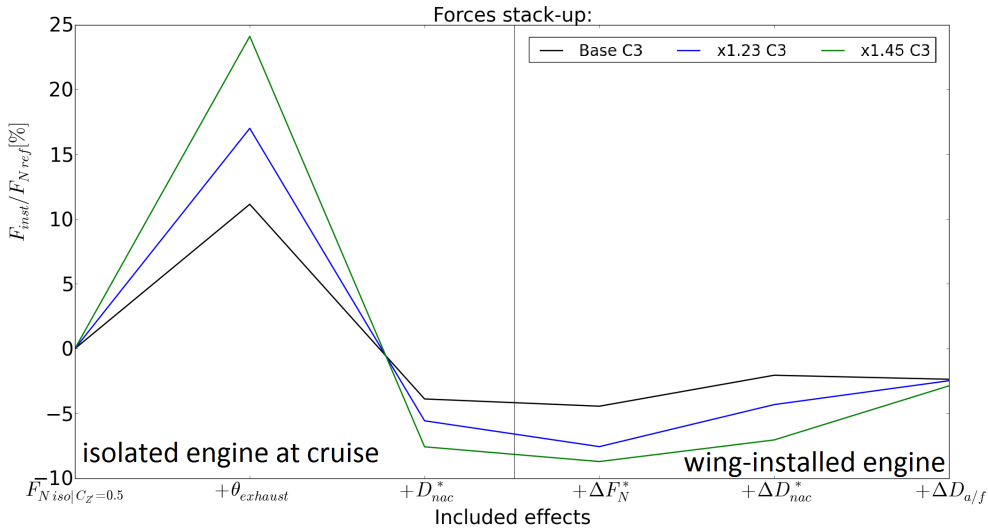


FIGURE E.8: Installation force diagram for three engine sizes E1, $1.23 \times E1$ and $1.45 \times E1$ in baseline position C3; standard net thrust of isolated engine at cruise ($F_{N \text{ iso}|C_{z'}=0.5}$) plus the effect of exhaust forces ($+\theta_{exhaust}$), nacelle drag (D_{nac}^*), and detailed interference effects (ΔF_N^* / ΔD_{nac}^* / $\Delta D_{A/F}$); $Re = 45 \times 10^6$; $M = 0.82$; $C_{z'} = 0.5$

$+11\%$ greater than the $F_{N \text{ ref}}$ for the baseline engine as a result of the beneficial forces that act on the exhaust ($+\theta_{exhaust}$). Moreover, the $\theta_{exhaust}$ increase was approximately

proportional with engine size, as a result of the same efflux conditions for all the engine sizes. Furthermore, the modified nacelle drag ($+D_{nac}^*$) was added to obtain the NPF. The isolated nacelle drag also broadly scaled with size. As a result the NPF for three engines was -3.6%, -5.5% and -7.5% less than the standard net thrust for baseline E1, 1.23xE1 scale and 1.45xE1 scale respectively.

The overall interference broadly scaled with size and greater interference benefits are observed for larger engines. As compared with +1.5% benefit for the baseline engine (E1), the two larger engines exhibited +3.09% and +4.7% for the 1.23xE1 and 1.45xE1 configurations, respectively. However, the break-down of the overall interference was different and depends on the engine size. The ΔD_{nac}^* was a beneficial force for all three engine sizes and the benefit broadly scaled with engine size. The beneficial contribution of +2.3% increased to roughly +3.2% for the size of 1.23xE1 and +5.5% for 1.45xE1. However, the significant benefit for a large size engine was observed in the effect of interference on airframe drag ($\Delta D_{A/F}$). The detrimental effect of -0.3% for the Baseline engine changed to roughly +1.9% for 1.23xE1 and 4.2% for 1.45xE1. The ΔF_N^* was determined as -0.5% for baseline engine. The 1.23xE1 engine exhibited a penalty of -2% and it was explained by the greater overlap of the exhaust with the wing as the exhaust was scaled proportionally with the fixed geometric point at the nacelle trailing edge. The largest engine 1.45xE1 had the greatest exhaust overlap with the wing and -5.5% effect on ΔF_N^* . Overall, the final force depends on the engine size (Figure E.8). The spread between baseline engine (E1) and the largest engine (1.45E1) was 0.5%. Even though it was anticipated that the aerodynamic interference would be severely more penalizing for the larger engines than for the baseline, it was not. The large penalties of the modified drag of the nacelle (D_{nac}^*) increased proportionally to the engine size, however, they were offset by the beneficial effect on the exhaust term ($+\theta_{exhaust}$). The 1.45xE1 engine exhibited -3.9% less net propulsive force than the baseline engine. However, the beneficial aerodynamic interference resulted in the total difference between wing-installed engines of 0.5% in the standard net thrust ($F_{N\ ref}$).

E.3.2 The effect of specific thrust

To highlight the impact of the exhaust design on the performance of the very-high by-pass ratio engine, a comparison between the large engine E2 and the geometrically enlarged version of E1 engine (1.23xE1) were considered (Figure E.9). The main aim of the comparison was to determine impact of the exhaust system on the overall installation effect (F_{inst}). The nacelles for both engines are the same size and they are geometrically similar. The engines were tested in isolation and under the wing in position C3 at $M_\infty = 0.82$. Both engines were operated at the baseline power setting PS1 with a intake mass flow capture ratio of 0.75. Therefore, as expected the greatest difference between the engines was observed in the jet and on the lower side of the wing (Figure E.9). Rather negligible difference was observed in the flow acceleration at the nacelle leading edge due to the constant MFCR between the engines. Moreover, the upper side of the wing was not exposed to the changes in the jet and remains largely unaffected.

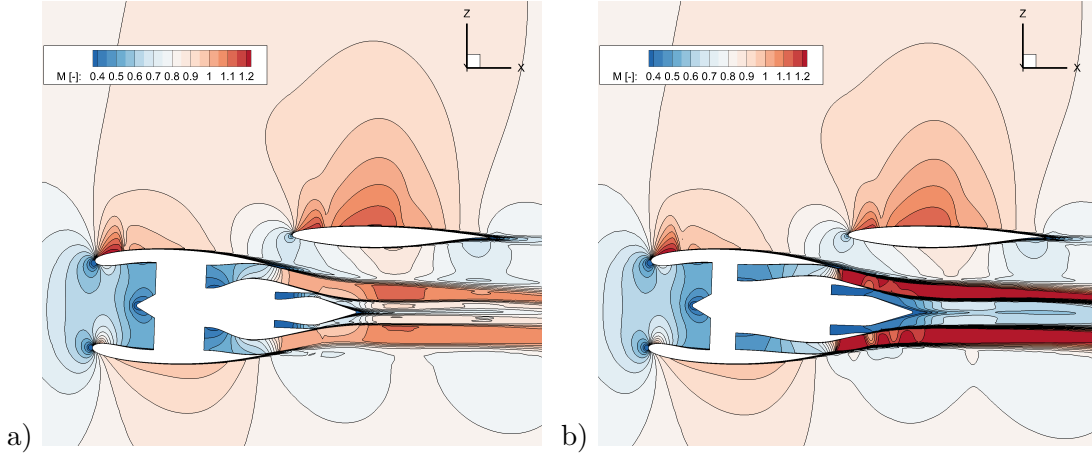


FIGURE E.9: Contours of Mach number for engine-aircraft configurations a) with E2 engine b) with 1.23xE1 engine; both in position C3 at $AoA_{A/C} = 2.5^\circ$; $Re = 45 \times 10^6$; $M_\infty = 0.82$

To quantify the effect of the exhaust design on the overall installation effect, the installation force diagram was created (Figure E.10). The two engine-airframe configurations were compared at the constant value of the overall vertical force ($C_{Z'} = 0.5$), which included the vertical component of thrust. To fully consider the performance of the isolated engine, the net propulsive force (NPF) of an engine was considered. The NPF consists of the effect of the standard net thrust (F_N), as well as the modified nacelle drag (D_{nac}^*) and the forces on the exhaust ($\theta_{exhaust}$). The standard net thrust of the engine depends predominantly on the stream forces at engine inlets and outlets. The notable difference between the engines was observed in the NPF (Figure E.10). As compared with -4.5% of $F_{N \text{ ref}}$ for the 1.23xE1 engine, the E2 engine exhibited -2.5% of $F_{N \text{ ref}}$ due to the re-design the engine cycle and the exhaust system to achieve a lower specific thrust design. The overall benefit of an isolated E2 engine comes from a $+1.2\%$ improvement in the forces on the exhaust ($\theta_{exhaust}$). Even though the projected

area of the core-cowl and the plug has decreased in comparison with the 1.23xE1 engine, the favourable fan nozzle pressure ratio (FNPR) results in a higher value of a beneficial pressure force exerted on the surface of the core-cowl. As a result, the $\theta_{exhaust}$ equaled +18.2%, +17.0% of $F_{N\ ref}$ for E2 and 1.23xE1, respectively. The beneficial forces on the exhaust were offset by a modified nacelle drag (D_{nac}^*). The penalty due to the D_{nac}^* was -20.7% and -21.6% of $F_{N\ ref}$ for E2 and 1.23xE1, respectively. As both engines were operated with a constant intake MFCR, the difference in the drag was a result of the different FNPR for the engines. The reduced FNPR for the E2 engine resulted in a change of the shape of the post-exit streamtube and due to the redirection of the flow, there was a gentle increase in the pressure force generated on the afterbody of the fan-cowl as compared with the 1.23xE1 engine.

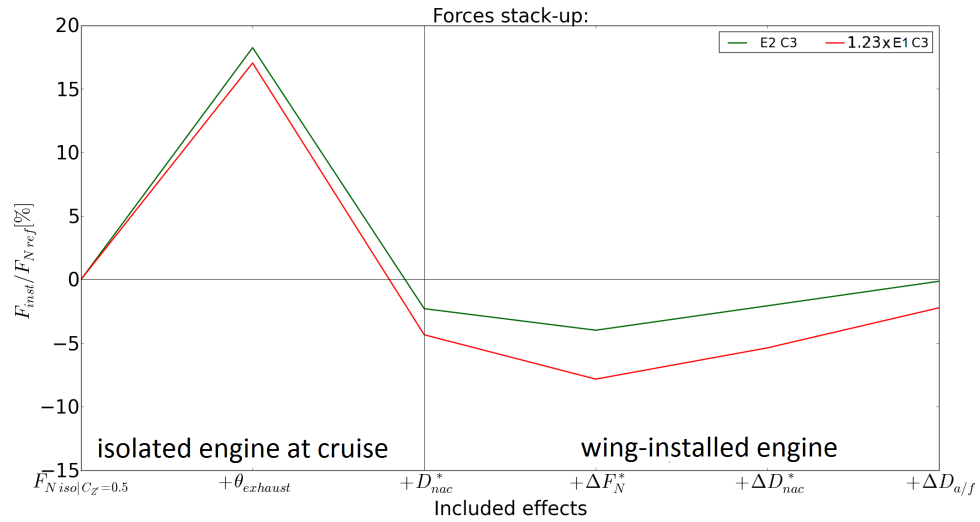


FIGURE E.10: Installation force diagram for E2 engine and 1.23xE1 engine in the position C3; standard net thrust of isolated engine at cruise ($F_{N\ iso|Cz'=0.5}$) plus the effect of exhaust forces ($+\theta_{exhaust}$), nacelle drag (D_{nac}^*), and detailed interference effects ($\Delta F_N^* / \Delta D_{nac}^* / \Delta D_{A/F}$); $Re = 45 \times 10^6$; $M = 0.82$; $C_{z'} = 0.5$

Due to the mutual interaction between the engines and the airframe, an overall interference effect (F_{interf}) was observed for E2 and 1.23xE1 configurations (Figure E.10). The E2 engine exhibits a beneficial overall interference of +2.7% as compared with +1.8% for 1.23xE1. The greatest part of the benefit came from the interference on the nacelle (ΔD_{nac}^*) as a result of the pressure force from the wing pressure field. The benefit was broadly constant for the two engines with +3.1% $F_{N\ ref}$ for E2 and +3.6% $F_{N\ ref}$ for 1.23xE1, as the engines had the same design of the nacelle. Furthermore, the beneficial effect of the airframe interference ($\Delta D_{A/F}$) contributed to the overall interference effect (F_{interf}) with +0.8% and +1.8% $F_{N\ ref}$ for E2 and 1.23xE1, respectively. The difference was attributed to interaction with the jet on the lower side of the wing, whereas the upper side of the wing remained broadly unaffected between the engine configurations. Eventually, the presence of the wing had an impact on the thrust (ΔF_N^*). Due to the overlap of the wing with the exhaust system, the acceleration of the flow in the created

channel between the core-cowl and the wing appears. The effect is more detrimental for the 1.23xE1 engine with -2.8% of $F_{N \text{ ref}}$ as compared with -1.2% $F_{N \text{ ref}}$ for the E2 engine. The higher FNPR for the 1.23xE1 engine results in a pattern of expansions and compressions. Due to this local acceleration of the flow in the jet and interaction with the wing due to the reduced flow area in an overlap position, a local backward-oriented pressure force was generated that caused the thrust loss. Overall (Figure E.10), the E2 engine performed +2% better than the 1.23xE1 engine as compared in isolation and also there was a +2% benefit for the E2 engine-airframe configuration in the overall installation force (F_{inst}). It can be stated that the main interference effects were captured correctly with the 1.23xE1 engine, even though the design of the exhaust was not optimised. The benefits for the E2 engine are explained by the favourable design of the exhaust system with a reduced FNPR to avoid strong shock interactions in the jet. The size of the nacelle was discovered to be the main factor that affected the interference terms, however, a modest effect of the FNPR was observed on the interference on thrust (ΔF_N^*) and the airframe interference ($\Delta D_{A/F}$). The changes in ΔF_N^* and $\Delta D_{A/F}$ were the function of FNPR and the two effects neutralised each other. Overall, approximately similar value of +2.7% $F_{N \text{ ref}}$ for the overall interference effect (F_{interf}) was observed for both E2 and 1.23xE1 engines.