

**CRANFIELD UNIVERSITY**



**J.M. Chang**

**A Flexible, Subsonic High Altitude Long Endurance**

**UAV Conceptual Design Methodology**

**COLLEGE OF AERONAUTICS**

**Ph.D. Thesis**

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**Ph.D. Thesis**

**J.M. Chang**

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**Supervisor: Dr. R.I. Jones**

**May 1997**

*To My Late Father*



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## *Abstract*

High Altitude Long Endurance (HALE) Unmanned Aerial Vehicles (UAVs) have been considered for use in both civil and military applications for some years. Their advantages, such as low cost, high survivability, long endurance, easy maintenance etc. relative to alternatives, make them useful to scientist and military personnel.

In this thesis, a flexible conceptual design methodology for HALE UAVs has been developed. This has been implemented as a FORTRAN computer code. However, it is unlike some commercially available general aircraft design software which appears to the user as a 'black box'. In this case, the code is broken-down into several sub-programs which deal with the different aspects such as parametric study, drag, performance etc. Each of these can be either used as it stands, tailored to the user needs in a particular case or by-passed if more accurate methods or known values are available in a particular area.

During production of the methodology problems were encountered in a number of areas due to the unusual operating regime and configurations of HALE UAVs.

Obtaining engine data for the high altitude of interest was a problem. This was addressed through use of an existing engine modeling code to generate data. The high altitude also leads to low Reynolds numbers and along with the high aspect ratios typical of HALE UAV configurations, these place such vehicles beyond the validity limits of data sheet methods for prediction of a number of important parameters. Improved methods for the prediction of Oswald efficiency and maximum lift coefficient, in particular, are recommended to be sought.

Accepting the above difficulties, an analysis of the Tier-II Plus, Global Hawk, was carried out using published data to provide some validation of the methodology and program. The results obtained provide confidence in the usefulness of the program in the analysis and investigation of HALE UAVs.

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## Notation

### General symboles

|              |                                                                           |                  |                                                               |
|--------------|---------------------------------------------------------------------------|------------------|---------------------------------------------------------------|
| AR           | Aspect ratio                                                              | $C_{D0emp}$      | Empennage zero-lift drag coefficient                          |
| $A_h$        | Horizontal tail aspect ratio                                              | $C_{Diemp}$      | Empennage coefficient due to lift                             |
| $A_T$        | Tailplane aspect ratio                                                    | $C_{Dfus}$       | Fuselage drag coefficient                                     |
| $\bar{a}$    | The average deceleration during the ground run                            | $C_{D0fus}$      | Fuselage zero-lift drag coefficient                           |
| b            | Wingspan                                                                  | $C_{Di}$         | Total airplane drag coefficient due to lift                   |
| $b_T$        | Tailplane span                                                            | $C_{Difus}$      | Fuselage lift drag coefficient                                |
| $C^3I$       | Command, control, communications, and intelligence                        | $C_{Dgear}$      | Landing gear drag coefficient                                 |
| $C^4I$       | $C^3I$ plus computers                                                     | $C_{Dmgear}$     | Main landing gear drag coefficient                            |
| C.G.         | Center of gravity                                                         | $C_{Dngear}$     | Nose landing gear drag coefficient                            |
| $CG_{eng}$   | C.G. location of engine                                                   | $C_{Dnac}$       | Engine nacelle drag coefficient                               |
| $CG_{fix}$   | C.G. location of fixed equipment                                          | $C_{Dnp}$        | Nacelle fuselage interference drag coefficient                |
| $CG_{fuel}$  | C.G. location of the fuselage; C.G. location of the fuel system           | $C_{D0nac}$      | Engine nacelle drag coefficient due to zero drag              |
| $CG_{htail}$ | C.G. location of the horizontal tailplane                                 | $C_{DEP}$        | Program cost associated with depots                           |
| $CG_{ml}$    | Nose landing gear C.G. location                                           | $C_{Dinac}$      | Engine nacelle drag coefficient due to lift                   |
| $CG_{nl}$    | Main landing gear C.G. location                                           | $C_{Dt}$         | Trim drag coefficient                                         |
| $CG_{pay}$   | C.G. location of payload                                                  | $C_{Dw}$         | Wing drag coefficient                                         |
| $CG_{vtail}$ | C.G. location of the vertical tailplane                                   | $C_{D0w}$        | Wing zero-lift drag coefficient                               |
| $CG_w$       | C.G. location of the wing                                                 | $C_{Diw}$        | Wing lift drag coefficient                                    |
| $C(y)$       | Local airfoil chord                                                       | $C_{Dn}$         | Nacelle fuselage interference drag coefficient                |
| $C_{ACQ}$    | Acquisition cost                                                          | $C_{Dtp}$        | Trim drag due to profile drag                                 |
| $C_{(a+e)m}$ | Cost of engine and avionics as acquired from vendors                      | $C_{em}$         | Cost per engine during the manufacturing phase                |
| $C_{CMAT}$   | Program cost of consumable materials used in conjunction with maintenance | $C_f/C$          | Proportion of control surface chord to wing chord             |
| $C_{crewpr}$ | Program cost for aircrews                                                 | $C_{ffus}$       | Turbulent flat plate skin friction coefficient of the fuselag |
| $C_D$        | Drag coefficient                                                          | $C_{fnac}$       | Turbulent flat plate skin friction coefficient of the nacelle |
| $C_{d0}$     | Total aircraft zero-lift drag coefficient                                 | $C_{fw}, C_{fw}$ | Turbulent flat plate friction coefficient of the wing         |
| $C_{dc}$     | Experimental steady state cross-flow drag coefficient a circular cylinder |                  |                                                               |
| $C_{Demp}$   | Empennage drag coefficient                                                |                  |                                                               |
| $C_{DISP}$   | Disposal cost                                                             |                  |                                                               |

|                           |                                                               |                 |                                                            |
|---------------------------|---------------------------------------------------------------|-----------------|------------------------------------------------------------|
| $C_h$                     | Horizontal tail volume coefficient                            | $C_{MAN}$       | Total airplane program manufacturing cost                  |
| $C_{intm}$                | Cost of interior                                              | $C_{manm}$      | Manufacturing labour cost                                  |
| $\frac{C_{IT}}{C_{LaT}}$  | Lift coefficient due to control surface deflection            | $C_{matm}$      | Manufacturing material cost                                |
| $C_{L0}$                  | Zero-angle-of-attack lift coefficient                         | $C_{mpdir}$     | Program cost for direct maintenance personnel              |
| $C_{LOWf}$                | Zero-angle-of-attack lift coeff. of wing-fuselage combination | $C_{m0b}$       | Increment in $C_{m0W}$ due to body of unswept wing         |
| $C_{Lh}$                  | Horizontal tail incremental lift coefficient                  | $C_{m0W}$       | Wing alone pitching moment coefficient at zero lift        |
| $C_{L\alpha}$             | Aircraft lift curve slope                                     | $C_{m0wb}$      | Wing body pitch moment coefficient at zero lift            |
| $C_{l\alpha}(y,t)$        | Local airfoil lift curve slope                                | $C_{MIS}$       | Program miscellaneous cost                                 |
| $C_{l\alpha OT}$          | Theoretical airfoil lift curve slope in incompressible flow   | $C_{OPS}$       | Operating cost                                             |
| $C_{Lah}$                 | h-tailplane lift curve slope                                  | $C_{PDIR}$      | Program cost of direct personnel                           |
| $C_{LaT}$                 | Tailplane lift curve slope                                    | $C_{PIND}$      | Program cost of indirect personnel                         |
| $(C_{l\alpha})_{atM=0}$   | Airfoil lift curve slope in incompressible flow               | $C_{POL}$       | Program fuel, oil and lubricants cost                      |
| $(C_{L\alpha w})_{atM=0}$ | Whole wing lift curve slope in incompressible flow            | $C_{PRO}$       | Profit                                                     |
| $(C_{l\alpha})_{atM}$     | Airfoil lift curve slope in compressible flow                 | $C_{qcm}$       | Quality control cost                                       |
| $(C_{L\alpha w})_{atM}$   | Whole wing lift curve slope in compressible flow              | $C_r$           | Wing root chord                                            |
| $C_{L\alpha w}$           | Whole wing lift curve slope                                   | $C_{RDTE}$      | Research, development, test and evaluation cost            |
| $C_{L\alpha wf}$          | Wing body lift curve slope                                    | $C_{SP}$        | Program cost of spares                                     |
| $C_{lmax}$                | Max. lift coeff. spanwise local airfoil max. lift coeff.      | $C_T$           | Tailplane root chord                                       |
| $C_{lmaxr}$               | Wing root maximum lift coeff.                                 | $C_t$           | Wing tip chord                                             |
| $C_{lmax t}$              | Wing tip maximum lift coeff.                                  | $C_{toolm}$     | Tooling cost                                               |
| $(C_{lmax})_w$            | Max. lift coefficient of the wing                             | CEF             | Cost escalation factor                                     |
| $C_l(y)$                  | Spanwise lift coefficient                                     | $\Delta C_{m0}$ | Body effect pitching moment coefficient at zero lift       |
| $C_l(y)_{basic}$          | Spanwise basic lift coefficient.                              | D               | Drag or the aircraft diameter                              |
| $C_l(y)_{add}$            | Spanwise additional lift coeff.                               | DC              | Distance to climb                                          |
| $C_v$                     | Vertical tail volume coefficient                              | $d_{nac}$       | Maximum engine nacelle dia.                                |
| $C_L$                     | Aircraft lift coefficient                                     | e               | Oswald efficiency                                          |
| $C_{Lcr}$                 | Cruise lift coefficient                                       | $e_h$           | Oswald efficiency for empennage                            |
| $C_{LH}$                  | Lift coefficient of tailplane                                 | E               | Endurance                                                  |
| $C_{lmax}$                | Maximum lift coefficient                                      | $E_p$           | The aircraft C.G. location of empty weight with payload    |
| $C_{IT}$                  | Rate of change of lift coefficient with flap deflection       | EP              | Engine price                                               |
| $C_{l\alpha T}$           | Tail lift curve slope                                         | $E_w$           | The aircraft C.G. location of empty weight                 |
| $C_{Lw}$                  | Wing lift coefficient                                         | F               | Empirical correction factor                                |
|                           |                                                               | $F_{a2}$        | Fuselage nacelle intersections without local area ruling   |
|                           |                                                               | $F_{OL}$        | A factor which accounts for the cost of oil and lubricants |
|                           |                                                               | f               | Equivalent parasite area                                   |

|                       |                                                                                                                              |               |                                                                                                 |
|-----------------------|------------------------------------------------------------------------------------------------------------------------------|---------------|-------------------------------------------------------------------------------------------------|
| FC                    | Fuel to climb                                                                                                                | $l$           | Characteristic length                                                                           |
| FD                    | Fuel density in pounds per gallon                                                                                            | $l_{Bn}$      | Distance from body nose to quarter chord point of MAC                                           |
| $F_{cad}$             | A judgement factor which accounts for the effect of computer aided design capability on airframe engineering and design cost | LCC           | Life cycle cost                                                                                 |
| $F_{diff}$            | A judgement factor which accounts for the difficulty of a new airplane program                                               | $l_T$         | Distance from aircraft most aft C.G. location to horizontal tail aerodynamic centre             |
| FP                    | Price of fuel in USD per gallon                                                                                              | $l_T^1$       | Distance from wing leading edge to tailplane aerodynamic centre                                 |
| $F_f$                 | Fuel flow                                                                                                                    | $L_{eng}$     | Engine length                                                                                   |
| $f_{dep}$             | Operating cost fraction about depot overhaul and maintenance                                                                 | $\Delta L$    | The change in aircraft lift                                                                     |
| $f_{pind}$            | Operating cost fraction which are not directly involved in flight operation or airplane maintenance                          | L/D           | Lift drag ratio                                                                                 |
| $F_{prom}$            | Profit factor                                                                                                                | $L_H$         | Moment arm, the distance from the horizontal tailplane quarter chord to the wing quarter chord. |
| $f_{sp}$              | Operating cost fraction about program spares                                                                                 | $L_m$         | Length of main landing gear                                                                     |
| H                     | Aircraft most aft C.G. location                                                                                              | $L_N$         | Length of nose landing gear                                                                     |
| HALE                  | High Altitude Long Endurance                                                                                                 | $l_{nac}$     | Engine nacelle length                                                                           |
| $H_0$                 | Wing body aerodynamic centre                                                                                                 | $L_t$         | Horizontal tail arm length                                                                      |
| h                     | Cruise altitude; Maximum height of body                                                                                      | $L_v$         | Moment arm, the distance from the vertical tailplane quarter chord to wing quarter chord.       |
| $h_{abs}$             | Absolute ceiling height                                                                                                      | MAC           | Mean Aerodynamic Chord                                                                          |
| $H_G$                 | Location of leading edge of MAC of the most after wheel position                                                             | $MAC_{htail}$ | Mean Aerodynamic Chord of h-tailplane                                                           |
| $h_h$                 | Height between wing and h-tailplane                                                                                          | $MAC_{vtail}$ | Mean Aerodynamic Chord of v-tailplane                                                           |
| $h_{TO}$              | Take-off obstacle height                                                                                                     | M             | Mach number; Mass flow rate                                                                     |
| $h_T$                 | Distance from thrust line above ground                                                                                       | $M_\infty$    | Air flow over the airfoil                                                                       |
| $H_i/H_v$             | For conventional tail is 0                                                                                                   | $M_{ff}$      | Total mission fuel fraction                                                                     |
| $i_h$                 | H-tailplane incidence angle                                                                                                  | $MHR_{acdp}$  | Total amount of engineering manhours required for the entire airplane program                   |
| $i_T$                 | Incidence angle of tailplane                                                                                                 | $MHR_{manp}$  | Total number of manhours required for the manufacturing of $N_{pro}$ airplanes                  |
| $i_w$                 | Wing body incidence                                                                                                          | $MHR_{flt}$   | No. of maintenance manhours required per flight hour                                            |
| K                     | Gust alleviation factor                                                                                                      | $\Delta n$    | The change in load factor                                                                       |
| $K_A, K_\lambda, K_h$ | Constant                                                                                                                     | $N_{acq}$     | Number of airplanes which are acquired                                                          |
| $K_B$                 | Pitch radius of gyration                                                                                                     | $N_{en}$      | Number of engines                                                                               |
| $K_n$                 | Stick fixed static margin                                                                                                    | $N_l$         | Limit landing load factor                                                                       |
| $K_{wf}$              | Wing-fuselage interference factor                                                                                            | $N_{loss}$    | Number of airplanes lost through accidents over the service life                                |
| L                     | Lift, Aircraft fuselage length                                                                                               | $N_m$         | Number of airplanes produced to production standard during an airplane program                  |
| $L'$                  | Airfoil thickness location parameter                                                                                         | $N_{mis}$     | Number of mission flown per                                                                     |
| $L'_h, L'_v$          | Airfoil thickness location parameter (empennage)                                                                             |               |                                                                                                 |

|                    |                                                                                  |                       |                                             |
|--------------------|----------------------------------------------------------------------------------|-----------------------|---------------------------------------------|
|                    | year                                                                             | $S_L$                 | Total landing distance                      |
| $N_{pro}$          | Total number of airplanes produced                                               | $S_{LG}$              | Landing ground run                          |
| $N_{rdte}$         | Number of airplanes produced during the RDTE phases                              | $S_n$                 | Maximum frontal area of the nacelle         |
| $N_{serv}$         | Number of airplanes of the type in actual service                                | $S_{net}$             | Wing net area                               |
| $N_t$              | Number of fuel tanks                                                             | $S_{mgear}$           | Main landing gear tire area                 |
| $N_z$              | Limit load factor                                                                | $S_{ngear}$           | Nose landing gear tire area                 |
| $N_{yr}$           | Service life                                                                     | $S_{p\ fus}$          | Fuselage platform area                      |
| $P_H$              | Susceptibility                                                                   | $S_{p\ nac}$          | Engine nacelle platform area                |
| $P_{K/H}$          | Vulnerability                                                                    | $S_T/S$               | Ratio of tailplane regarding to wing        |
| $P_s$              | Probability of survival                                                          | $S_{TO}$              | Take-off field length                       |
| $P_t$              | Total pressure ratio                                                             | $S_{TOG}$             | Take-off ground distance                    |
| $q$                | Dynamic pressure                                                                 | $S_V$                 | The area of vertical tailplane              |
| $R$                | Cruise distance (to target area); Reynolds number                                | $S_W$                 | Planform area of wing                       |
| $R_{cmat}$         | Average cost (per maintenance hour) for consumable materials                     | $S_{wb}$              | Wing /body adjacent area                    |
| $R_{em}$           | Engineering manhour rate at manufacturing phase                                  | $S_{wet}$             | Wetted area                                 |
| $R_{mm}$           | Manufacturing labour rate per hour for the entire program                        | $S_{wet\ fus}$        | Wetted area of fuselage                     |
| $R_{er}$           | Engineering manhour rate at RDT&E phase                                          | $S_{wetnac}$          | Wetted area of nacelle                      |
| $R_{mml}$          | Military maintenance labour rate                                                 | $S_{wetw}$            | Wetted area of the wing                     |
| $R_{Nfus}$         | Fuselage Reynolds number                                                         | $T$                   | Thrust                                      |
| $R_{Nwing}$        | Wing Reynolds number                                                             | $TET$                 | Turbine inlet temperature                   |
| $R_{L\ s}$         | Lifting surface correction factor                                                | $t_{cl}$              | Time to climb                               |
| $R_{wf}$           | Wing/fuselage interference factor                                                | $(\frac{1}{c})_{max}$ | Maximum thickness ratio wing                |
| $R_{wfh}, R_{wfv}$ | Empennage/fuselage interference factor                                           |                       | chord sweep angle                           |
| $RC$               | Rate of climb                                                                    | $(t/c)_r$             | Thickness ratio of root section             |
| $RCS$              | Radar cross section                                                              | $(t/c)_t$             | Thickness ratio for tip section             |
| $RDT\&E$           | Research, development, test and evaluation                                       | $(\frac{1}{c})_T$     | Tailplane thickness ratio                   |
| $RSTA$             | Reconnaissance, surveillance and target acquisition                              | $T_{net}$             | Net thrust                                  |
| $S$                | Wing gross area                                                                  | $TOP$                 | Aircraft take-off parameter                 |
| $S_B$              | Platform area of body                                                            | $T_{SSL}$             | Standard atmosphere sea level thrust        |
| $S_{air}$          | The distance from the obstacle height to the point of touchdown                  | $T_{TO}$              | Sea level static net thrust                 |
| $S_{HT}$           | Horizontal tailplane area                                                        | $T/W$                 | Aircraft thrust weight ratio                |
| $sfc$              | Specific fuel consumption                                                        | $T_P$                 | The aircraft C.G. of the take-off weight    |
| $S_{bfus}$         | Fuselage base area                                                               | $T_w$                 | The aircraft C.G. of empty weight with fuel |
| $S_{Bn}$           | Platform area of body forward of lateral line through quarter chord point of MAC | $U$                   | Gust velocity                               |
| $S_f$              | Fuselage wetted area                                                             | $UAVs$                | Unmanned Aerial Vehicles                    |
| $S_H$              | The area of horizontal tailplane                                                 | $U_{anu}$             | Annual utilisation in flight hours          |
|                    |                                                                                  | $U_{de}$              | The standard vertical gust velocity         |
|                    |                                                                                  | $\bar{V}$             | Tail volume coefficient                     |
|                    |                                                                                  | $V_{app}$             | Aircraft approach speed                     |
|                    |                                                                                  | $V_{cl}$              | Climb speed                                 |
|                    |                                                                                  | $V_o$                 | Equivalent aircraft velocity                |
|                    |                                                                                  | $V_i$                 | Integral tanks volume                       |
|                    |                                                                                  | $V_{dmin}$            | Minimum drag speed                          |

|                    |                                                                                                    |
|--------------------|----------------------------------------------------------------------------------------------------|
| $V_{\max}$         | Maximum speed                                                                                      |
| $V_s$              | Aircraft stall speed                                                                               |
| $V_t$              | Total mission fuel volume                                                                          |
| $V_T$              | Aircraft true velocity                                                                             |
| $V_{TD}$           | Touchdown speed                                                                                    |
| $V_{LOF}$          | Take-off speed                                                                                     |
| $W_{\text{ampr}}$  | Aeronautical manufacturers<br>planning report weight                                               |
| $W_{\text{emp}}$   | The empty weight of the aircraft<br>includes structure, propulsion,<br>avionics, instruments, etc. |
| $W_{\text{en}}$    | Engine weight                                                                                      |
| $W_{\text{fli}}$   | Flight control system weight                                                                       |
| $W_{\text{fuel}}$  | Mission fuel weight or fuel<br>system weight                                                       |
| $W_{\text{Fused}}$ | Mission fuel used in pounds per<br>mission                                                         |
| $W_{\text{fuse}}$  | Aircraft fuselage group weight                                                                     |
| $W_{\text{HT}}$    | Horizontal tailplane group<br>weight                                                               |
| $W_{\text{hyd}}$   | Hydraulic system weight                                                                            |
| $W_i$              | Weight during different phase, $i$<br>$= 1 \dots n$                                                |
| $W_l$              | Landing design gross weight                                                                        |
| $W_{\text{mis}}$   | Miscellaneous weight                                                                               |
| $W_{\text{ml}}$    | Main landing gear weight                                                                           |
| $W_{\text{nl}}$    | Nose landing gear weight                                                                           |
| $W_{\text{pay}}$   | Payload weight                                                                                     |
| $W_{\text{VT}}$    | Vertical tailplane group weight                                                                    |
| $W_{\text{wing}}$  | Wing group weight                                                                                  |
| $W/S$              | Aircraft wing loading                                                                              |
| $W_{\text{to}}$    | Aircraft take-off gross weight                                                                     |
| $x/c$              | Chordwise location of boundary<br>layer transition point                                           |
| $Z$                | Vertical displacement of quarter<br>chord point of wing root above<br>mid body position            |
| $Z_T$              | Distance of thrust line below<br>centre of gravity                                                 |

## Greek Symbols

|                             |                                                                                                                                                                      |
|-----------------------------|----------------------------------------------------------------------------------------------------------------------------------------------------------------------|
| $\sigma$                    | Density ratio                                                                                                                                                        |
| $\Lambda$                   | Sweep angle of wing leading<br>edge                                                                                                                                  |
| $\Lambda_{c/4}$             | Sweep angle of wing quarter<br>chord                                                                                                                                 |
| $\Lambda_{\% \max}$         | Maximum thickness ratio of<br>tailplane sweep angle                                                                                                                  |
| $\Lambda_H$                 | Horizontal tailplane leading<br>edge sweep angle                                                                                                                     |
| $\Lambda_V$                 | Vertical tailplane leading edge<br>sweep angle                                                                                                                       |
| $\lambda$                   | Taper ratio                                                                                                                                                          |
| $\lambda_h$                 | Horizontal tailplane taper ratio                                                                                                                                     |
| $\lambda_v$                 | Vertical tailplane taper ratio,                                                                                                                                      |
| $\rho$                      | Air density                                                                                                                                                          |
| $\rho_{\text{SSL}}$         | Standard sea level air density                                                                                                                                       |
| $\Gamma$                    | Wing dihedral                                                                                                                                                        |
| $\epsilon_t$                | Wing twist angle                                                                                                                                                     |
| $\epsilon(y)$               | Twist angle of local section at $y$                                                                                                                                  |
| $\epsilon_{0T}$             | Tailplane downwash angle for<br>zero aircraft angle of attack                                                                                                        |
| $\tau$                      | Ratio of thickness ratio at tip<br>and root                                                                                                                          |
| $\tau$                      | Inner trailing edge angle                                                                                                                                            |
| $\tau_a$                    | Outer trailing edge angle                                                                                                                                            |
| $\eta$                      | Ratio of the drag of a finite<br>cylinder to the drag of an<br>infinite cylinder;<br>Elevator deflection; Spanwise<br>distance from root as fraction of<br>semi-span |
| $\eta_h$                    | Dynamic pressure ratio                                                                                                                                               |
| $\eta_i$                    | Aileron inner position, from<br>wing root                                                                                                                            |
| $\eta_o$                    | Aileron outer position, close to<br>wing tip                                                                                                                         |
| $\mu$                       | Mass ratio; dynamic viscosity                                                                                                                                        |
| $\mu_g$                     | Ground friction coefficient                                                                                                                                          |
| $\lambda_f$                 | Fineness ratio                                                                                                                                                       |
| $\lambda_{\text{nac}}$      | Engine nacelle fineness ratio                                                                                                                                        |
| $\beta$                     | Prandtl-Glauert compressibility<br>factor                                                                                                                            |
| $\frac{d\epsilon}{d\alpha}$ | Down wash gradient at the                                                                                                                                            |



|                                     |                                                                                                                               |
|-------------------------------------|-------------------------------------------------------------------------------------------------------------------------------|
|                                     | horizontal tail                                                                                                               |
| $\alpha$                            | Fuselage angle of attack                                                                                                      |
| $\alpha_{CL\ max}$                  | Average angle of attack of wing airfoil                                                                                       |
| $\alpha_h$                          | Horizontal tailplane                                                                                                          |
| $\alpha_{OLh}$                      | Zero-lift angle of attack of h-tailplane                                                                                      |
| $\alpha_{0l}$                       | Change in zero-lift angle of the wing per degree of positive twist at the tip                                                 |
| $\alpha_{OL}$                       | Airfoil zero-lift angle of attack                                                                                             |
| $\Delta\alpha_{0r}$                 | Body effect on zero lift angle of attack                                                                                      |
| $\alpha_{0lr}$                      | Zero lift angle of attack of the root section                                                                                 |
| $\alpha_{OLw}$                      | Zero-lift angle of attack of wing                                                                                             |
| $\alpha_{0rWB}$                     | Wing body zero lift angle of attack                                                                                           |
| $\alpha_{nac}$                      | Engine nacelle angle of attack                                                                                                |
| $\alpha_T$                          | Angle of incidence of horizontal surface; trim in cruise                                                                      |
| $\alpha_W$                          | Wing body incidence                                                                                                           |
| $\delta$                            | Increment of plane wing vortex induced drag coeff. due to additional lift; Local camber dependent twist relative to wing root |
| $v$                                 | Induced drag factor due to linear twist                                                                                       |
| $\omega$                            | Zero-lift drag factor due to linear twist                                                                                     |
| $\omega_s$                          | Short period frequency                                                                                                        |
| $\zeta_s$                           | Short period damping ratio                                                                                                    |
| $\zeta_L$                           | Damping ratio in phugoid motion                                                                                               |
| $\frac{\Delta\alpha_0}{\epsilon_i}$ | Change in wing zero-lift angle of attack per degree of linear wing twist                                                      |
| $\psi$                              | Angle between zero lift plane of wing and airflow direction for zero body alone pitch moment                                  |
| $\phi_{le}$                         | Tailplane leading edge sweep angle                                                                                            |
| $\phi_{te}$                         | Tailplane trailing edge sweep angle                                                                                           |

# Chapter 1

## Introduction

During the 1970's, the US military successfully applied UAV in the Vietnam war for effective utilization of battlefield surveillance and target designation. Such techniques have also been employed in more recent times, i.e. the Gulf war. The commanders of US Army, Navy and Marine are well aware of UAVs operational effectiveness. During the Gulf war, the Pioneer UAVs flew over 300 combat missions over the course of operation. In 1,641 hours of flight only one was shot down. They were extremely successful in target selection, damage assessment, near real time reconnaissance and target confirmations and locations. Their success established the utility and importance of UAVs in combat.

Nowadays, the UAVs applications not only for military, but for civilian operations, includes high altitude endurance atmospheric, storm research, meteorology and mapping, weather information, ocean fishing law enforcement, relay of radio, television and communication, border patrol and surveillance etc. All of these missions, ideally, require continuous coverage.

Following two decades of dispute over the controversial topics of the importance of UAVs, their usefulness has now become clear, and everybody wants them. Reference 55 and 75 stated that the unmanned air vehicle market is set to grow to around USD 1 billion per year by the turn of the century. The reason is the understanding of UAVs in both military and civilian sector is increasing, and the reliability and safety will be

improved by using modern state of the art technology. Therefore, we foresee in the next decades the UAVs market will continue to expand rapidly.

## **1.2 Why we need UAVs**

In general UAVs can carry out both military and civilian missions such as communication relay, reconnaissance, surveillance command, control, electronic warfare, electronic Support Measurement, electronic Counter Measurement, missions need loiter for a long time (not suitable for man) etc.

On the basis of above applications, some of the important factors are summarized to emphasize why we need UAVs.

### **1.2.1 Command, control, communications, computer and intelligence**

As the technology improves, and user needs increase, reference 78 states that the capabilities of UAVs from the previous C<sup>3</sup>I, command, control, communications, and intelligence upgrade to C<sup>4</sup>I, command, control, communications, computer and intelligence. As enemy forces become more mobile and weapon system technology advances, the gathering of tactical reconnaissance data by manned aircraft becomes difficult and more hazardous. The UAVs intent is to provide real time multimission support to the C<sup>4</sup>I efforts required to conduct joint operations.

### **1.2.2 Without loss of human life**

In terms of “without loss of human life”, this is not only good for military but for civil applications;

- a. Military: During a war, high risk missions occurs as a result of enemy heavy air defense weapons. If we use UAVs to collect intelligence, the worst case scenario is the UAV being shot down, without loss of personnel.
- b. Civilian: Some missions need a long loiter time, i.e. more than 20 hours, therefore hard for human beings to cope with. If we choose unmanned vehicles, crew fatigue is not a problem.

### **1.2.3 Extremely flexible, capable of doing dynamic tasks**

Without human beings, strike, salary, passion, and affection are no longer a problem, day and night, rain or sleet, as long as the mission needs, the UAVs will be there. Therefore, once a UAV is ready, it is extremely flexible, and there should be less possibilities of human related problems compromising the mission.

### **1.2.4 Reliable and Safe**

Past records show that the biggest problem for UAVs is the reliability and safety. Without man on board, real time correction can only be provided for by a very sophisticated autonomous control system. As the technology improves, the requirement of reliability and safety of UAVs is similar to that of manned aircraft. Therefore, in the future the UAVs will be trustworthy and indispensable for battle commanders or scientists.

### **1.3 Current trend of UAVs**

Reference 78 stated that since 1989 the United States has organized the Unmanned Aerial Vehicles (UAV) Joint Project Office. It is the single Department of Defense organization charged with management responsibility for UAVs. From them an annual UAV Master Plan was produced. In this plan, they divided UAVs into four categories: close-range ( 30-50 km), short-range ( up to 150 km beyond the forward line of own troops ), medium-range ( up to 650 km ) and endurance ( 24 hours on-station capability, preprogrammed and remote control ).

The recent announcement of Department of Defense's major UAV programs were Tier-II Plus Global Hawk and Tier-III Minus Dark Star. In May 1995, Teledyne Ryan Aeronautical was awarded a contract to develop the Tier-II Plus program. The low-observable Dark Star, Tier-3 UAV, was awarded to Boeing/Lockheed Martin.

Within a US\$ 10 million cost cap for both air vehicles. Tier-II Plus and Tier-III Minus are designed to meet similar needs but use radically different approaches. Tier-III Minus is a very survivable aircraft. While Tier-II Plus goes for the maximum reconnaissance capability. The Tier-II Plus will depend on its all-weather, day/night sensors and high altitude (up to 65,000 ft) to stay away from most threats or in high threats to wait until very-high-altitude surface-to-air missiles have been suppressed. The Tier-III Minus Dark Star is intended to penetrate air defense without too much difficulty before they are suppressed. Tier-III Minus is a much lower endurance aircraft, it carries only one sensor (either radar or TV) at much shorter range and duration and transmits a lower data rate.

From above description, it is not difficult to picture that a quick response, fast deployed, high altitude, long endurance and even hard to detect UAVs are on the horizon. It appears possible that in the near future, UAVs will carry out the missions the fighters do at present.

The usage of UAVs has been confirmed by many countries, but in terms of propulsion system, especially for high altitude and endurance, it is not clear between turbine, prop, or solar engines which one is best.

## 1.4 Scope of this thesis

Figure 1.4.1 is the flowchart of this thesis. Chapter 2 covers the main conceptual design program, and in chapter 3, we chose Teledyne Tier-II Plus and new HALE UAV to perform validation and application. Chapter 4 addresses the general cost analysis and discussion, and in chapter 5, the conclusions and future work recommendations are presented.

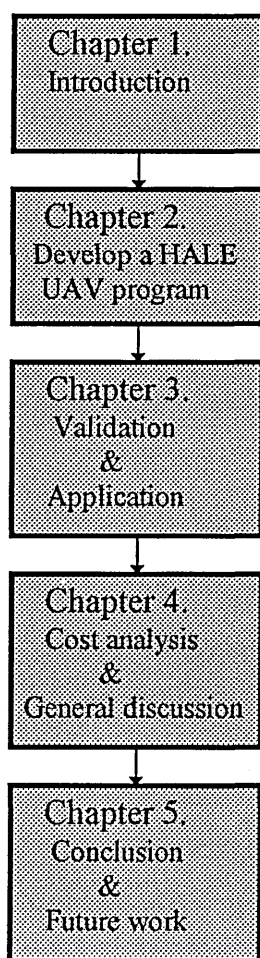


Fig 1.4.1 Flowchart of thesis

To provide a quick response vehicle, we are concentrating on the choice of a turbofan engine as our propulsion system. Through careful assumptions and calculation, the results shows that with a turbofan engine as a propulsion system it is quite feasible to meet the requirement of fast deployment to high altitude and long endurance (more than 20 hours).

In this thesis, the main objective is to produce a conceptual design methodology of subsonic high altitude long endurance UAV (not for a canard configuration). This objective is realized through the production of a computer code which allows the user to either produce an outline design for an aircraft beginning with only a basic mission specification or perform analysis on an existing vehicle. To allow for variations in the types of vehicle considered or types of analysis performed, the programme is written in a form where the user can easily alter sections, equations or logic to meet their particular needs. Though the propulsion system, in this program, is a turbofan engine, as long as the flight speed is within subsonic, it is feasible for any other kind propulsion systems.

This program is written in the Fortran language. At present it is compiled and run under the Cranfield university computer center Vulcan system. With minor modifications, it could be compiled with other existing Fortran compilers and run on other computer system.

## **Chapter 2**

### **Develop a High Altitude Long Endurance UAV Program**

In accordance with reference 1, chap. 1, for design of an airplane, it is necessary that a mission specification for the aeroplane is available. In general, the mission specification will come from the customer.

Following the specification, the airplane designer begins the aeroplane preliminary design, such as preliminary sizing, configuration, and layout design, etc. As a result of the preliminary design studies, a specific need can be met, then full scale development of the aeroplane can follow.

The purpose of this chapter is to describe the basis of a program for a high altitude long endurance UAV. Program is written to fulfil requirement, it follows preliminary design procedures, step by step, to finish the preliminary design job.

#### **2.1 Mission profile initialisation**

High altitude long endurance UAV, for military usage, should incorporate new technology, and the battle field requirements. From reference 61, the mission should included; Reconnaissance, Surveillance, Target Acquisition (RSTA).

RSTA Mission includes availability, accuracy and timeliness of cuing data, target mobility and proximity to the UAV launch platform, threat condition and the size of the area to be searched with the sensor available.

In order to meet an air defence requirement, the following requirements are needed. Long endurance on-station loiter, surveillance covering over a greater range, and real time transmitting of enemy activity imagery to the battle commander. A high altitude

long endurance UAV can achieve these requirements. If we require quick deployment of UAVs to a high altitude, then an UAV with turbofan engine should be employed. After analysis and comparisons of general UAV missions, the following mission is chosen during the preliminary design phase, fig. 2.1.1.

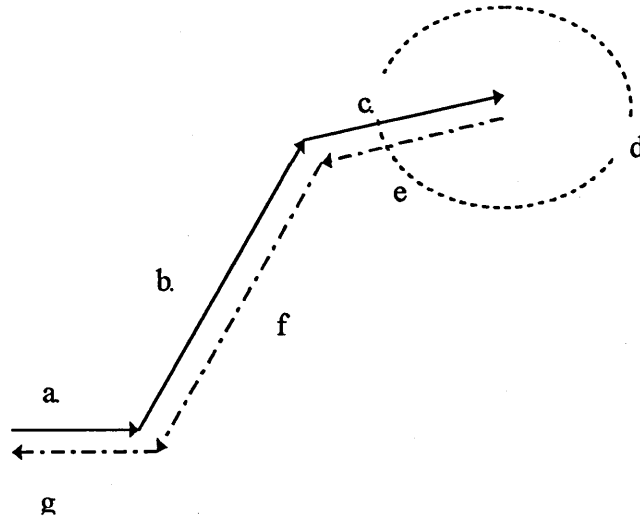


Fig. 2.1.1 Mission profile

In fig. 2.1.1, section a. is the taxi and take-off segment, section b. is the climb segment, section c. is cruise altitude, then cruise to target area, and section d. is on-station loiter for 24 hours.

Sections e. and f. are cruise back and then descent to sea level respectively. Finally section g. is landing and engine shut-off.

## 2.2 The initial specification of HALE

Following the mission requirement in the previous section, the next step is to build an initial specification for preliminary design.

This UAV is designed for military use, and the purpose of it is to operate at high altitude with long endurance. In order to meet the requirement, the initial specifications are listed in table 2.2.1.



|                      |                                          |
|----------------------|------------------------------------------|
| Altitude             | 50,000 ~ 65,000 ft                       |
| Endurance on station | 24 hrs                                   |
| Cruise Speed         | 0.5 ~ 0.6 M                              |
| Payload              | MSAR, FLIR<br>ESM, AN/SRQ-4<br>> 200 lbs |
| Weight               | ~ 10,000 lbs                             |

Table 2.2.1 Initial Specification

As we mentioned in chap. 2.1, Reconnaissance, Surveillance and Target Acquisition (RSTA) missions should be included. From reference 61, there are many different kind of payload combination can meet our requirement. But if we put weight and power consumption into consideration, then the payload combination listed in table 2.2.1 is a good selection. Following is the usage of each payload.

- a. Radar: Miniature Synthetic Aperture Radar (MSAR) provide an all weather, day and night sensor that could produce high quality reconnaissance imagery in adverse weather and visibility conditions.
- b. Forward Looking Infrared (FLIR) is capable of detecting large contacts at long ranges provided there is a sufficient difference between the contact and background temperatures.
- c. Electronic Support Measures (ESM) provides information in a timely fashion and in forms that are readily usable by the users it supports.
- d. Electronic data link system: data links of future UAV's must be capable of performing in increasing hostile electromagnetic threat environments. AN/SRQ-4 data link system, both in size and weight, is specifically developed for use onboard.

As well as the specification in table 2.2.1, due to the military usage of HALE UAV, rapid deployment is another very important factor to be considered in conceptual design.

Quick deployment means the flying speed should be fast. However, based on the mission requirement, the HALE UAV is designed for on station, loiter of around 24 hours. For such a long period of loiter, in terms of fuel consumption, the subsonic speed is of economic vantage. Therefore, if we take both factors, fast speed and fuel consumption, into consideration, then the turbofan engine is the best choice for the propulsion system of the HALE UAV.

## 2.3 Flowchart of the HALE UAV program

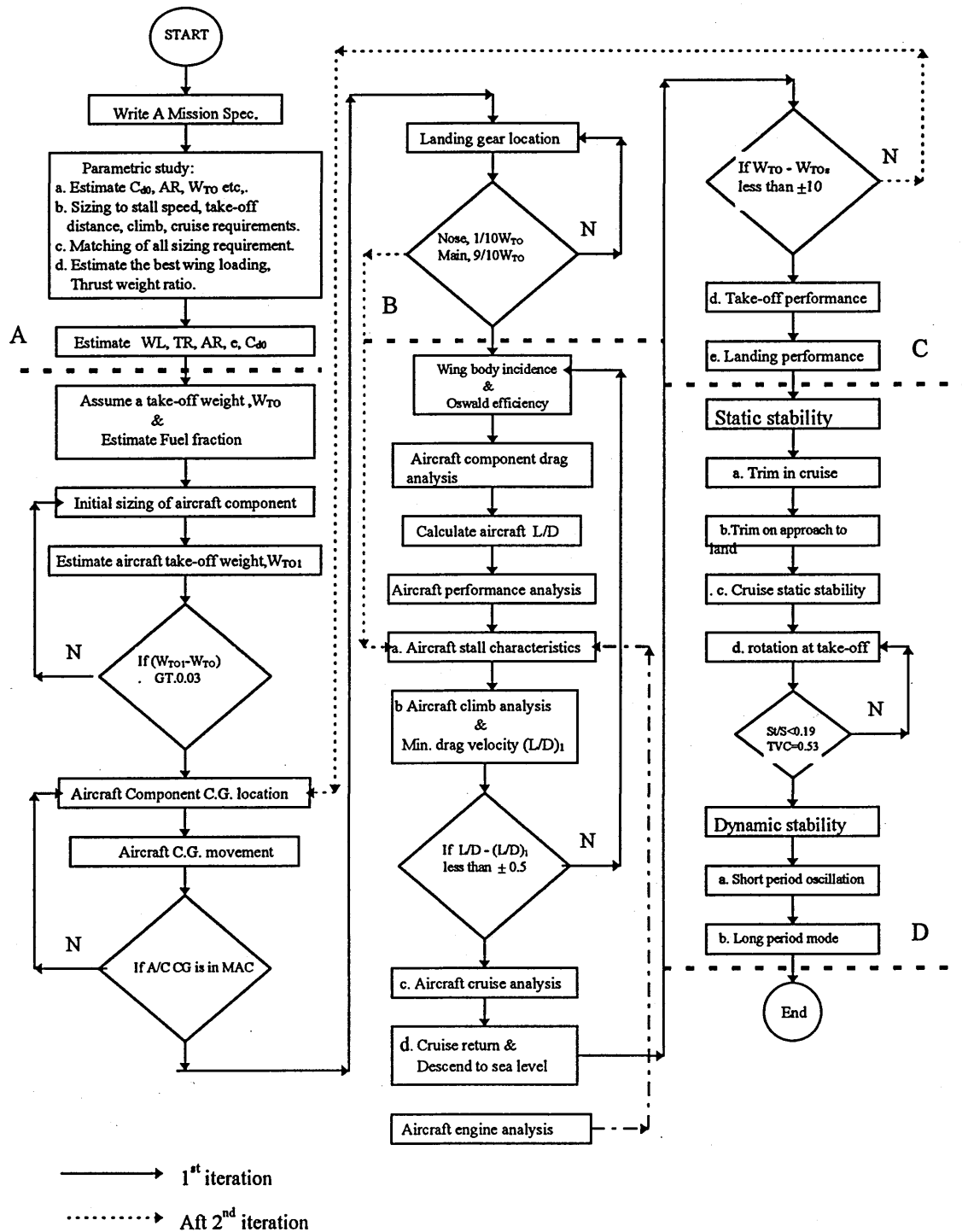


Fig 2.3 The program flowchart

We divided fig. 2.3 into four different blocks. In block A, following the parametric study, the aircraft wing loading, aspect ratio and zero drag coefficient may be estimated. In block B, first we assume a take-off weight, based on this weight to estimate mission fuel fraction and the aircraft component weight. Then the new aircraft take-off weight is equal to the mission fuel weight plus the aircraft component weight. If the difference between new weight and assumed weight is over  $\pm 3$  percent, then we recalculate the take-off weight. In block C, the aircraft performance is being estimated. In this block, after the cruise mission is finished, we compare the left fuel weight with what is needed during cruising. If the difference is over  $\pm 10$  pounds, then the more or insufficient fuel will be added or reduced from the previous take-off weight and go back to recalculate again. In block D, the aircraft static and dynamic stability is estimated.

## 2.4 Parametric study and fuel fraction analysis

When an aircraft is being designed, relevant aircraft data is a good reference to use for parametric study. After analysing high altitude reconnaissance aircraft from references 62, 63, 64 and 65, information on manned and unmanned high altitude reconnaissance aircraft was collected, and this data is presented in appendix A, table A.1.1.

### 2.4.1 Parametric study

In order for parametric study analysis, the similar aircraft data is a good source of reference. From appendix A.2, we consider four similar aircraft.

- TR-1,  $W_{to}$  is 40,000 lbs,  $S$  is 1,000  $\text{ft}^2$ , aspect ratio is 10.6.
- YQM-94A,  $W_{to}$  is 17,220 lbs,  $S$  is 485  $\text{ft}^2$ , aspect ratio is 16.7.
- Ryan QM-98A,  $W_{to}$  is 14,310 lbs,  $S$  is 347  $\text{ft}^2$ , aspect ratio is 16.5.
- Canberra,  $W_{to}$  is 40,000 lbs,  $S$  is 960  $\text{ft}^2$ , aspect ratio is 4.2.

The detailed calculation of similar aircraft lift drag ratio analysis is listed in appendix A.2, the results are listed in table 2.4.1.

| Aircraft | $C_{d0}$ | $C_l$ | $C_d$   | L/D   | Oswald effi. |
|----------|----------|-------|---------|-------|--------------|
| TR-1     | 0.01     | 1.439 | 0.081   | 17.76 | 0.87         |
| YQM-94A  | 0.015    | 1.11  | 0.04435 | 25    | 0.8          |
| QM-98A   | 0.0184   | 1.289 | 0.0585  | 22    | 0.8          |
| Canberra | 0.0133   | 0.338 | 0.0227  | 14.9  | 0.92         |

Table 2.4.1 Similar aircraft lift drag ratio estimation

From these results we select the airplane performance aspects that are similar to the new designed one, and take their  $C_{d0}$ , aspect ratio, Oswald efficiency, and lift drag data as references.

Whether the results from parametric study is close to the requirement mainly rely on the information which comes from the similar aircraft. Therefore, a first guess of aircraft parameters will be an important factor worth to be considered. From table 2.4.1, comparing with the similar aircraft data, YQM-94A and QM-98A, the value of  $C_{d0}$  as 0.0175, aspect ratio as 15 and Oswald efficiency as 0.8 have been chosen for the parametric study of the HALE UAV.

Following reference 1, based on the requirement of stall speed, takeoff, climb, and cruise, the initial aircraft sizing, including calculating the wing loading, and thrust weight ratio of the new aircraft is performed. Detailed calculation is listed in appendix A.3, and the results are in fig. 2.4.1.

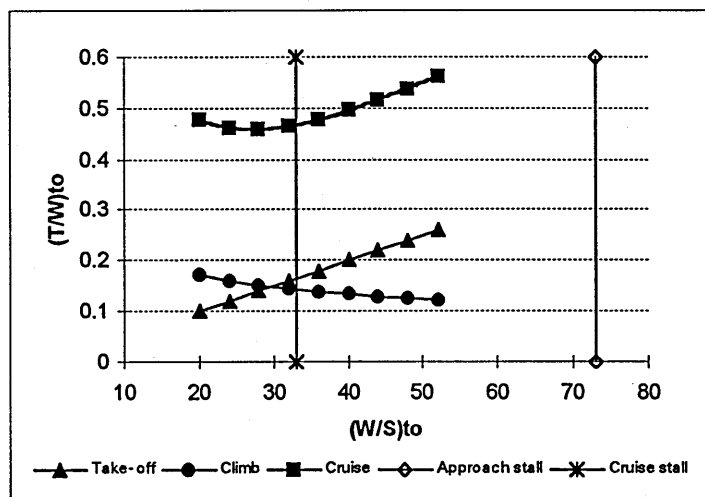


Fig. 2.4.1 Matching results of sizing a HALE UAV

In order to estimate the optimum matching point in fig. 2.4.1, following steps are taken.

- The wing loading ratio should be not greater than 33 psf (above this value, during cruise the aircraft would be susceptible to stall).
- Since HALE UAV is cruising at high altitude, the equivalent thrust requirement for cruise is higher than the thrust requirement for take-off and climb. Therefore, if thrust requirement and wing loading is satisfied for cruise, then it will automatically be satisfied for the thrust required during take-off and climb.
- Based on the assumption, aspect ratio is 15, zero drag coefficient is 0.0175, and Oswald efficiency is 0.8, we obtain the wing loading is 30 psf, the thrust weight ratio is 0.4589 (detailed calculation is in appendix A.3).

## 2.4.2 Fuel fraction analysis

From table 2.4.1, the lift drag ratio of YQM-94A and QM-98A is 25 and 22 respectively. In the beginning of our calculation, we assume the lift drag ratio, as a conservative estimate of 22.

There are six different segments, fig. 2.1.1, in the mission profile. Following the assumptions listed in reference 8, the fuel fraction can be calculated (the figure of fuel fraction, in each segment, is based on assumption, it may be slightly different from real situation. However, during calculation it will be rechecked later).

### Phase 1. Engine start and take-off.

This phase includes the start of the engine, taxi, take-off etc. A reasonable figure of the fuel consumption is around 2.5 to 3.0 percent to take-off weight. So, if we assume this phase begins at  $W_1$  and ends at  $W_2$ , then the ratio of weight is

$$\frac{W_2}{W_1} = 0.975$$

### Phase 2. Climb and accelerate to cruise condition

After take-off the aeroplane is climbing to the mission cruise altitude. During this phase the fuel consumption is assumed as stated in reference 8, fig. 5.5. The fuel consumption is about 3 to 3.5 percent of the weight. So, if the beginning weight is  $W_2$ , the end is  $W_3$ , then the ratio of weight is

$$\frac{W_3}{W_2} = 0.97$$

### Phase 3. Cruise out

According to reference 8, the weight ratio is determined by Breguet range equation.

$$R = \left( \frac{V}{sfc} \right) \left( \frac{L}{D} \right) \ln \left( \frac{W_3}{W_4} \right) \quad (2.4.1)$$

$W_3$  is the beginning weight, and  $W_4$  is at the end of this phase. Assume during cruise-climb that the specific fuel consumption, sfc, is 0.7 lb/lb hr, cruising distance, R, is 500 nautical miles, and cruise-climb speed is 0.55 M, 532.44 fps. Substitute these values into eq. 2.4.1, then the ratio of weight is

$$\frac{W_4}{W_3} = 0.97$$

### Phase 4. Loiter

As mentioned in reference 8, the fuel weight fraction during this phase is determined by using the endurance equation for jet aircraft :

$$E = \left(\frac{L}{D}\right) \left(\frac{1}{sfc}\right) \ln\left(\frac{W_4}{W_5}\right) \quad (2.4.2)$$

$W_4$  is beginning weight,  $W_5$  is at the end of this phase. During the loiter phase the specific fuel consumption,  $sfc$ , is assumed as 0.65 lb/lb hr, and loiter time is 24 hr. Substitute these data into eq. 2.4.2, then the ratio of weight is

$$\frac{W_5}{W_4} = 0.492$$

#### Phase 5. Return cruise

The same as in phase 3,  $W_5$  is the beginning weight, and  $W_6$  is at the end of this phase. Thus the ratio of weight is

$$\frac{W_6}{W_5} = \frac{W_4}{W_3} = 0.97$$

#### Phase 6. Descent and landing

From cruising altitude descent to sea-level. Assume the fuel consumption is about 1 to 1.5 percent of the weight.  $W_6$  is the beginning weight, and  $W_7$  is at the end of descent, then the ratio of weight is

$$\frac{W_7}{W_6} = 0.99$$

According to reference 1, pp:16, the total mission ratio of weight can be calculated as:

$$M_f = \prod \left( \frac{W_{i+1}}{W_i} \right) \quad (2.4.3)$$

Thus, the total mission ratio of weight is

$$\begin{aligned} M_f &= \left(\frac{W_2}{W_1}\right) \left(\frac{W_3}{W_2}\right) \left(\frac{W_4}{W_3}\right) \left(\frac{W_5}{W_4}\right) \left(\frac{W_6}{W_5}\right) \left(\frac{W_7}{W_6}\right) \\ &= 0.434 \end{aligned}$$

From reference 1, eq. 2.14. The mission fuel weight is :

$$\begin{aligned} W_{fuel} &= (1 - M_f) W_{TO} \\ &= 0.566 W_{TO} \end{aligned} \quad (2.4.4)$$

If we add three percent mission fuel as reserve, then the mission fuel weight will become:

$$W_{fuel} = 1.03(1 - M_f) W_{TO} \quad (2.4.5)$$

Thus, if the take off weight,  $W_{TO}$ , of HALE is around 10,000 lbs, then the mission fuel weight,  $W_{fuel}$ , is 5,829.8 lbs. This fuel weight is the first iteration result, and its weight is a function of aircraft take-off weight. Therefore, any further aircraft take-off weight changes will cause the fuel weight to change as well.

## 2.5 The take-off weight estimation

From the previous analysis, the take-off wing loading and mission fuel weight have been estimated. From this, the aircraft take-off weight,  $W_{TO}$ , can be estimated.

The take-off gross weight,  $W_{TO}$ , is a very important design parameter for sizing the entire aircraft. Since it's the first iteration, we don't know much detail about the aircraft except what the missions are. In this chapter we use the method listed in reference 8, chap. 5.1, to calculate the weight of each aircraft component.

We should bear in mind, that this method of calculating component weights is determined through the use of empirical formulations which are functions of many different geometric properties. Thus, it is unlikely to give an accurate answer for weight estimation until the design is finalised.

Therefore, in this section we use an iteration process to do the weight estimation.

### 2.5.1 Initial sizing of aircraft component

Before calculating the take-off weight, the aircraft configuration, such as the wing geometry, fuselage shape and empennage etc., should be defined first.

From chap. 2.3, the program flowchart, we know that the aircraft take-off weight and aircraft component sizing are not fixed until the final aircraft design is finished. Therefore, following the aircraft take-off weight and wing area, fuselage length etc., are the results of the first iteration. The final results require more iterations.

#### 2.5.1.1 Wing layout

According to references 8 and 9, the general design requirement of the airfoil section and planform are selected as follows:

- High lift coefficient
- High lift curve slope
- High wing fuel volume
- High aspect ratio
- Low weight
- Low zero lift drag coefficient

The above factors are the ideal case for a selected wing, in real situation some of these items contradict, so it is impossible to incorporate every good aspect into a certain wing airfoil. Thus, we have to consider the mission requirement of our own aircraft and tailor these factors into a wing airfoil.

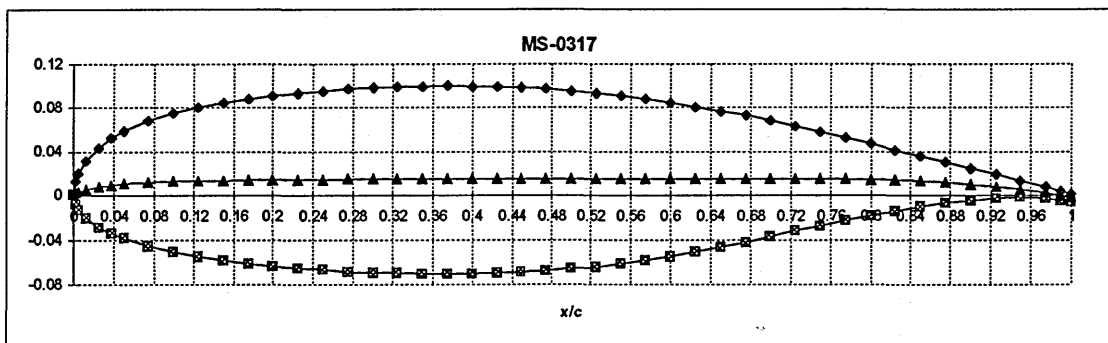
As in the mission requirement, fig. 2.1, the aircraft is designed to fly at subsonic

speed, high altitude and loiter for a long time. Therefore, from previous aircraft with similar mission requirement, the wing should be designed with a high aspect ratio that can reduce the induced drag, with high lift coefficient and low drag coefficient this can reduce the thrust required and, therefore, the fuel consumption will be lower. Apart from this, the high wing fuel volume is also an important factor for long endurance flight.

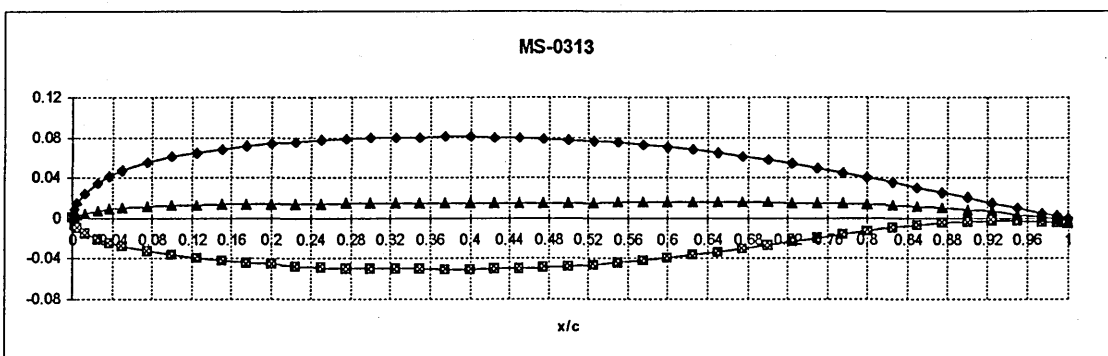
When these factors are taking into consideration, from reference 28, at subsonic speed the maximum lift coefficient of the MS(1)- series airfoils ranging from 1.6 to 2.0 with Reynolds number ranging from  $2 \times 10^6$  to  $12 \times 10^6$  has been selected to avoid potential buffet problem.

Therefore, if flying speed is subsonic, then the MS(1)- series airfoils should be a good candidate for generating high lift coefficient. According to reference 9, p:236, the thickness ratio of wing root section,  $(t/c)_r$ , is from 15 percent to 20 percent. It can provide adequate room for fuel and undercarriage retraction. The thickness ratio for tip section,  $(t/c)_t$ , is from 10 percent to 15 percent, this reduction proportion is good for lower structural weight. Thus, we take MS(1)-0317, and MS(1)-0313 airfoils for wing root and wing tip respectively.

The profiles of MS(1)-0317, and MS(1)-0313 airfoils are listed in fig. 2.5.1(a), (b).



(a)



(b)

Fig 2.5.1 Airfoil profile (a) MS(1)-0317, (b) MS(1)-0313



According to chap. 2.4.1, the parametric study, the wing loading is 30 psf, aspect ratio is 15, and the other wing airfoil information can be calculated. The detailed calculation of wing planform is discussed in appendix A.4.1.1 and A.4.1.2, the results are listed as:

- Wing gross area is 340 ft<sup>2</sup>.
- Wingspan is 71.41 ft.
- Sweep angle of leading edge,  $\Lambda$ , is 2 degrees.
- Sweep angle of quarter chord line,  $\Lambda_{c/4}$ , is 0 degrees.
- The wing taper ratio,  $\lambda$ , is 0.5.
- The value of wing root chord,  $C_r$ , is 6.35 ft.
- The value of wing tip chord,  $C_t$ , is 3.175 ft.
- The thickness ratio of root section,  $(t/c)_r$ , is 17 percent.
- The thickness ratio for tip section,  $(t/c)_t$ , is 13 percent.
- The Mean Aerodynamic Chord, MAC, is 4.94 ft.
- The fuel weight which can be accommodated in wing fuel tank is 5164.4 lbs.

#### **2.5.1.2 Estimate wing dihedral, $\Gamma$ , and twist, $\epsilon$**

Reference 9, p: 258, and reference 10, table 4.2, suggests for low-wing aircraft with straight wings, a dihedral of 5 to 7 degrees. The reason for more dihedral for low wing aircraft is that the flow around the wing-fuselage intersection in a sideslip induces a non-symmetric wing lift distribution, resulting in a rolling moment which counteracts the effect of the dihedral. For HALE, because the wing span is quite large, we choose wing dihedral,  $\Gamma$ , as 5 degrees.

The wing twist is used to prevent tip stall and to revise the lift distribution to approximate to an ellipse. Reference 10, p: 58, suggests typically the wing twist is ranging from 0 to 5 degrees nose down at tip. For initial design phase, 3 degrees of wing twist would provide adequate stall characteristics. Thus, for HALE, the wing twist,  $\epsilon$ , is 3 degrees in preliminary design phase.

#### **2.5.1.3 The aileron sizing**

The aileron is the primary roll control device, and operates by increasing lift on one wing and reducing it on the other. Reference 10, fig. 6.3, mentions, that typically the ailerons extend from about 50 percent to about 90 percent of the span.

For HALE, by comparing with similar mission aircraft (in appendix A.4.1.3), the aileron extends from 50 percent to 80 percent of the wing semispan is assumed.

#### 2.5.1.4 The fuselage length

Reference 9, chap. 3, mentions that there is no standard fuselage design solution to different purpose aircraft. However, in general a cylindrical mid-section is used for the following reasons.

- Structural design and manufacture are considerably simplified.
- With efficient internal layout.
- With flexibility to adjust arrangement.
- Further development by increasing or decreasing the length of fuselage is easy.

Apart from above reasons, when comparing with the fuselage configuration of similar mission aircraft, YQM-98A and YQM-94A, we decide to select a cylindrical mid-section as our fuselage configuration. From the first iteration calculation, the fuselage length is 35.7 ft.

The detailed calculations of the fuselage presented in appendix A.4.1.4, and the fuselage configuration is shown in fig. 2.5.2.

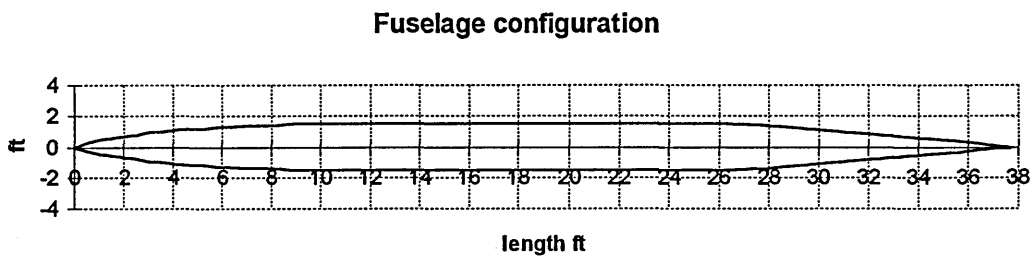


Fig. 2.5.2 Fuselage configuration

#### 2.5.1.5 The horizontal and vertical tailplane sizing

The primary purpose of a tail is to counter the moments produced by the wing. Thus the size of tailplane will be in some way related to the wing size. According to reference 9, chap. 3.1.2, as the length of the fuselage increases, the areas of the tail surfaces will be reduced. From the previous section, the proportion of fuselage length and width is similar to sailplane. Thus the tail area should be small.

Reference 9, chap. 9.5.1, the basic requirement of tailplane shape is that the airfoil section should have a high lift curve slope, and a large range of usable angles of attack. Frequently a symmetrical airfoil is used, and the thickness ratio is from 9 to 12 percent. According to this, in the initial design, we decide to take NACA 0012 airfoil for the tailplane configuration.

From reference 11, the basic information of NACA 0012 airfoil is acquired, and the airfoil profile is shown in fig. 2.5.3.

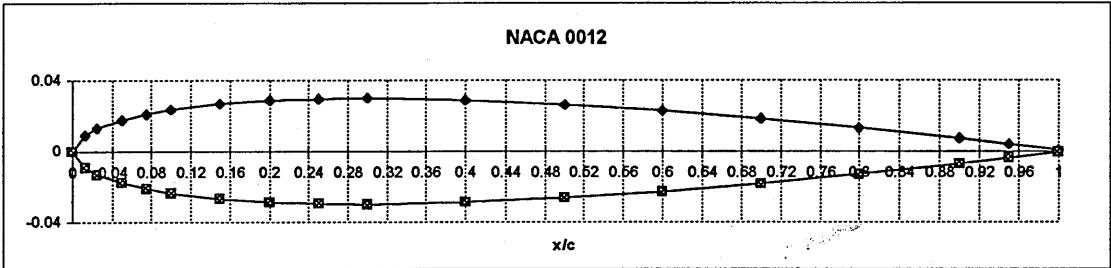


Fig. 2.5.3 NACA-0012 airfoil

Following is the disposition of horizontal and vertical tailplane of HALE. When comparing with YQM-98A and YQM-94A, both of them have two vertical tails, and the vertical tails are mounted on the horizontal plane, and the horizontal tailplane is mounted on the fuselage.

Therefore, for HALE, like YQM-98A and YQM-94A, we assume the number of vertical tails is two, mounted on either side of the horizontal tailplane, and the horizontal tailplane is mounted on the fuselage.

The detailed calculation of the horizontal and vertical tailplane sizing is listed in appendix A.4.1.5, and the results summarised in table 2.5.1.

|                                | Horizontal tail | Vertical tail |
|--------------------------------|-----------------|---------------|
| Area           ft <sup>2</sup> | 64.642          | 42.095        |
| Tail span      ft              | 18.311          |               |
| Root chord    ft               | 3.53            | 4.41          |
| Tip chord      ft              | 3.53            | 2.205         |
| Taper ratio                    | 1               | 0.5           |
| Aspect Ratio                   | 5.187           | 1.92          |
| L.E. sweep angle (degrees)     | 0               | 30            |
| MAC           ft               | 3.53            | 3.432         |
| Airfoil                        | NACA 0012       | NACA 0012     |

Table 2.5.1 Tail sizing

### 2.5.1.6 Three-view diagram of HALE

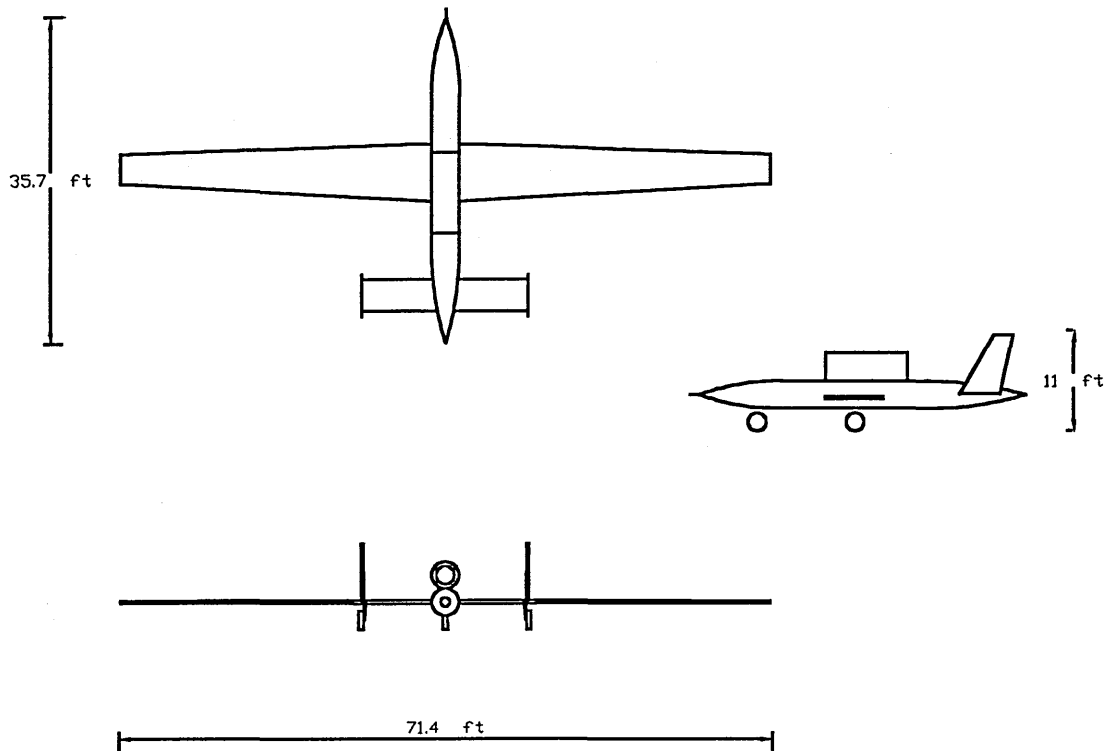


Fig. 2.5.4 Three-view diagram of HALE

### 2.5.2 Aircraft take-off weight estimation

According to reference 10, chap. 15, there are three categories, fighter, cargo and general aviation, available for estimating aircraft take-off weight. Because the HALE is categorised neither fighter nor cargo aircraft, in preliminary design phase, the General-aviation take-off weight method has been chosen to do weight estimation.

As previously stated, the following aircraft component weights are not the final solution, they are the results after first iteration calculation.

The detailed calculation is listed in appendix A.5, and each component weight is listed in table 2.5.2.

| Aircraft Component         | Weight lbs | Percent |
|----------------------------|------------|---------|
| Wing                       | 1309.35    |         |
| Horizontal tail            | 69.998     |         |
| Vertical tail              | 54.21      |         |
| Fuselage                   | 266.22     |         |
| Main landing gear          | 202.15     |         |
| Nose landing gear          | 61.63      |         |
| a. Structure               | 1963.558   | 0.187   |
| Engine install             | 1502.37    |         |
| Fuel system                | 386.18     |         |
| b. Propulsion              | 1888.55    | 0.18    |
| Hydraulic system           | 10.2       |         |
| Flight control             | 153.33     |         |
| inst.+avion.+elect.        | 327.82     |         |
| c. Total fixed equipment   | 491.35     | 0.047   |
| A. Empty weight (a.+b.+c.) | 4343.466   | 0.414   |
| B. Mission payload         | 200        | 0.019   |
| C. Fuel                    | 5946.244   | 0.567   |
| Gross weight (A.+B.+C.)    | 10489.702  |         |

Table 2.5.2 Aircraft component weight

## 2.6 Balance calculation

After calculating the aircraft component weight, it is followed by the aircraft balance calculation. This section, covers aircraft CG movement estimation and landing gear balance calculation.

### 2.6.1 Aircraft C.G. movement

In order to calculate the aircraft CG movement, the aircraft component C.G. location has to be estimated first. First we assume the reference axis origin is 62 inches (50 inches from reference axis, 12 inches is the length of pitot tube) ahead of aircraft nose, then according to the method in reference 2, chap. 10, each component C.G. location is calculated. The detailed calculation is listed in appendix A.6.1, and the results summarised in table 2.6.1.

| TYPE OF COMPONENT    | $X_i$ , inch |
|----------------------|--------------|
| Fuselage group       | 238.67       |
| Wing group           | 268.4        |
| Horizontal tailplane | 448.752      |
| Vertical tailplane   | 450.035      |
| Engine group         | 267.427      |
| Nose landing gear    | 175.25       |
| Main landing gear    | 311.14       |
| Fixed equipment      | 179.78       |
| Fuel                 | 268.4        |
| Fuel tank            | 268.4        |
| Payload              | 252.5        |

Table 2.6.1 Aircraft component C.G. location

From chap. 2.5.2, table 2.5.2 and table 2.6.1, the results of aircraft component weight and component C.G. location are given. Thus, from reference 2, chap. 10, the C.G. location must be calculated for all feasible loading scenarios. Although these loading scenarios are dependent on the mission requirements, typically, for HALE, the loading combinations are empty weight, empty weight with payload, empty weight with fuel, and take-off weight respectively.

Based on the above assumption, the calculation of aircraft C.G. movement for these four scenarios is performed. The detailed calculation is in appendix A.6.1, and the results are put in fig. 2.6.1.

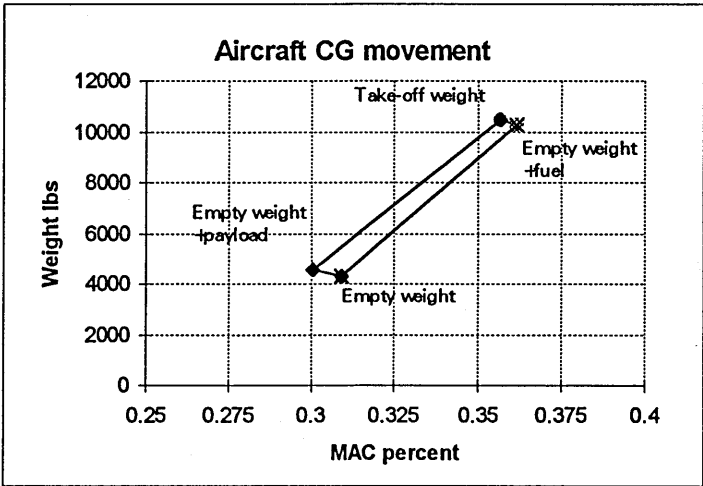


Fig. 2.6.1 Aircraft C.G. movement

From fig. 2.6.1, the aircraft C.G. movement is from 30 to 36 percent of MAC. According to reference 9, table 8.16, for jet engine airplanes, the C.G. movement should around 10 to 40 percent of wing MAC (for the type of configuration considered here). The C.G. movement of HALE is between 30 to 36 percent of wing MAC. Obviously, the aircraft C.G. movement is within limit.

## 2.6.2 Landing gear balance calculation

From table 2.6.1, the aircraft landing gear location and C.G. has been calculated. But whether the location of aircraft landing gear is correct, still needs to be verified.

Reference 66, suggests that in the preliminary design, the nose landing gear and main landing gear are designed to take 10 percent and 90 percent of aircraft take-off weight respectively. Therefore, if we want to know whether the aircraft landing gear is in it's proper location, the following calculation should be done.

First, in fig. 2.6.2 is an illustration of aircraft landing gear arrangement.

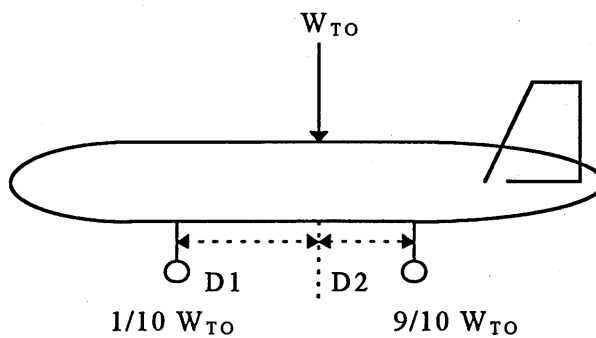


Fig. 2.6.2 Aircraft landing gear location

According to fig. 2.6.2, the following formula may be used.

$$D2 * W_{TO} = (D1 + D2) * \frac{1}{10} W_{TO} \quad (2.6.1)$$

$D1$  and  $D2$  is the distance from aircraft C.G. to nose landing gear and main landing gear respectively. During calculation the sum of  $D1$  and  $D2$  assumed to be a constant, and the aircraft take-off weight,  $W_{TO}$ , is at the aircraft most aft C.G. location.

The detailed calculation is in appendix A.6.2. After calculation, the nose and main landing gear moved forward from previous 169.1 and 297.67 inches to a new location 138.12 and 266.67 inches. Since the landing gears weight compared to the take-off weight is small, and their effect to the aircraft C.G. movement is not significant. Therefore, we did not go back to recheck the aircraft C.G. movement.

## 2.7 Wing body incidence and Oswald efficiency

Before calculating the drag and performance of the aircraft, two factors, wing body incidence,  $i_w$ , and Oswald efficiency,  $e$ , need to be determined.

When calculating wing body incidence, the basic aircraft data such as what kind airfoil and tail has been chosen, the value of the lift curve slope and Reynolds number etc., is needed first.

### 2.7.1 Wing body incidence

The HALE aircraft is designed to cruise for more than 24 hours. So the fuel consumption should be kept as low as possible. In order to have a low fuel consumption, the lift/drag ratio must be as high as possible. That means, during cruise, the aircraft should generate the required lift with as little drag as possible.

From chap. 2.5.1.1, we take MS(1)-0317, -0313 airfoil for wing root and tip, and according to reference 28, these airfoils can generate more lift in subsonic flight. From chap. 2.5.1.5, we select NACA-0012 for horizontal and vertical tailplane airfoil.

According to references 3 and 9, wing body incidence is the wing setting relative to the fuselage, and usually chosen during cruising flight with a representative all-up weight at a representative altitude so that the fuselage axis will be horizontal. If we ignore the effects of wing/fuselage interference, body lift and trim load, then for preliminary design phase, eq. A.7.17 may be used.

After calculation (the detailed calculation is in appendix A.7.3), the value of the wing body incidence is 6.8 degrees.

At this point, the value of the wing body incidence is based on some assumptions, such as the lift coefficient during cruise is 1. Therefore, the wing body incidence will vary when the real lift coefficient obtained. Thus the real wing body incidence, will be recalculated later.

### 2.7.2 Oswald efficiency

According to reference 9, eq. 5.10, may be used to calculate the aircraft drag coefficient due to lift coefficient, for a wing has elliptic spanwise lift distribution.

$$C_{Di} = \frac{C_L^2}{\pi AR} \quad (2.7.1)$$

But unfortunately, the spanwise lift distribution of most wings is not elliptic. Thus we need a factor, Oswald efficiency,  $e$ , to account for non-elliptic lift distribution.



Many textbooks introduce how to estimate Oswald efficiency, but most of them are only valid when the aspect ratio is less than a certain value, such as reference 6, eq. 4.12, is only valid when the aspect ratio is between 2 to 10.5. It's hard to find a method which can really estimate Oswald efficiency when the aspect ratio is higher than 20.

Due to the difficulty in finding a proper method to estimate Oswald efficiency for high aspect ratio wing. Reference 66, reference 9, eq. F.18, and reference 27, p:174, has been used to estimate the value of Oswald efficiency. After calculation (the detailed calculation is listed in appendix A.7.4), a value of 0.8 was chosen as Oswald efficiency to perform further conceptual design.

## 2.8 Performance Calculation

The subject of aircraft performance is highly interesting to both designer and customers. Customers may not be interested in either complicated aerodynamics formula or fancy structure composition. But if you mention how fast the aircraft can fly, how high and how many hours the aircraft can stay and how long the runway the aircraft need for take-off and landing, to most customers that is what they want.

The following, based on reference 7, is an attempt to estimate aircraft performance.

### 2.8.1 Aircraft component drag analysis

According to reference 6, chap. 4, the total aircraft drag is:

$$D = \frac{1}{2} \rho V^2 S C_D \quad (2.8.1)$$

In eq. 2.8.1,  $C_D$  is total aircraft drag coefficient,  $\rho$  is the air density,  $V$  is the free stream velocity and  $S$  is the wing area.

In general, the total aircraft drag coefficient can be simply calculated by eq. 2.8.2.

$$C_D = C_{D0} + \frac{C_L^2}{\pi A Re} \quad (2.8.2)$$

But, when doing performance analysis, in order to acquire accurate aircraft performance data, use of eq. 2.8.2 to calculate aircraft total drag coefficient is not enough. In another words, we need to know the value of each aircraft component drag.

From eq. 2.8.2, we know that the lift induced drag coefficient can be reduced by using a higher aspect ratio. From chapter 2.5.1.1, the aspect ratio of HALE is 15, which is higher than ordinary commercial and military aircraft. According to reference 31, such

a high aspect ratio is like a high performance glider. High aspect ratio usually generates high lift drag ratio, and thus, the total drag will decrease with higher aspect ratio.

According to reference 30, the amount of lift generated depends on the circulatory strength of the bound vortex, and on its length, which in turn depends on the span of the wing. Therefore, basically the longer span, the higher aspect ratio you have the less total drag you get. But a longer span will generate structural problems. Therefore, how to make a compromise between light structure weight and high aspect ratio is an important question for aircraft designers.

In this paragraph, we want to calculate each component of the zero lift drag and lift induced drag then sum all together to obtain the total aircraft drag. Since the total aircraft drag will vary with aircraft altitude, velocity and weight, so, it is difficult to list the detailed drag value in each different phase. Thus, in this paragraph, we only select a particular case, which is during cruise, as an example to explain, in this program, how to calculate the total aircraft drag coefficient.

First we assume the aircraft weight is 10,400 lbs, flight speed is 0.55 M, and the altitude is 60,000 ft.

For HALE, in preliminary design, we are only dealing with drag coefficient of wing, fuselage, empennage, trim drag, engine, engine nacelle and landing gear. Miscellaneous drag coefficient is very small and this is not important to the overall result. Thus, in this example, it is ignored.

After calculation (the detailed calculation is in appendix A.8), each aircraft component drag is obtained.

Following summation, during cruise, the total aircraft drag coefficient independent of lift , $C_{D0}$ , is 0.0155, and the total aircraft drag coefficient due to lift,  $C_{di}$ , is 0.0296.

Finally the total aircraft drag is 0.045 (in table 2.8.1 is the detailed component drag coefficient).

|          | Wing   | Fuselage | Empennage | Trim   | Engine nacelle |
|----------|--------|----------|-----------|--------|----------------|
| $C_{D0}$ | 0.008  | 0.00264  | 0.0026    | 0      | 0.00222        |
| $C_{Di}$ | 0.0293 | 7E-07    | 0.000035  | 0.0002 | 9E-07          |

Table 2.8.1 Aircraft component drag coefficient

Each aircraft component drag proportion is illustrated in fig. 2.8.1.

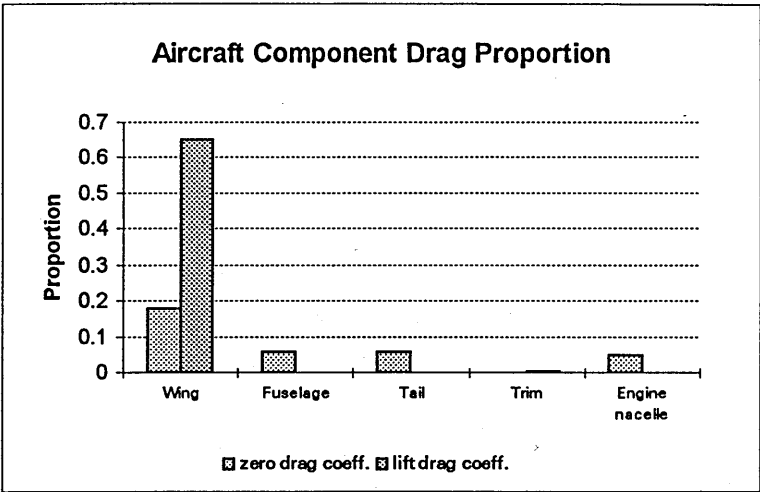


Fig. 2.8.1 Aircraft component drag illustration

From fig. 2.8.1, whether for zero lift drag coefficient or drag coefficient due to lift, the wing drag coefficient is a dominant factor in total drag coefficient. In other words, for HALE, in terms of aircraft performance, the wing drag will become the most important factor to deal with.

2.8.2 Stall analysis

Reference 33, p:28, states that when the aircraft angle of attack increases, the aircraft lift will increase, but this increase can not go on indefinitely. Intuitively, one might reason that there would be a practical limit to angle of attack, and indeed there is such a limit, and this limit is known as the stalling point.

Reference 9, p:227, the purpose of specifying certain acceptable stall characteristics is to minimise the chance of an inadvertent entry into a stall and ensure recovery from it without loss of life or property.

The maximum lift characteristics of an airfoil as well as the associated stall behaviour are of great importance in the aircraft performance. In this paragraph we use the methodology mentioned in reference 7, chap. 4 and reference 9, appendix E, to calculate the stall characteristics of HALE.

According to the specification of the HALE, on reaching 60,000 ft, it starts cruise-climb to 65,000 ft and then loiters for more than 24 hours on station.

Thus, before calculating the stall speed, we should assume some condition to do the calculation. The condition chosen is, the aircraft altitude is 60,000 ft, weight 10,400 lbs and flight velocity is 0.55 M.

Since the aircraft stall speed varies with aircraft weight and altitude, we may generate a stall speed chart, based on different aircraft weights and altitudes. This is given in fig. 2.8.2.

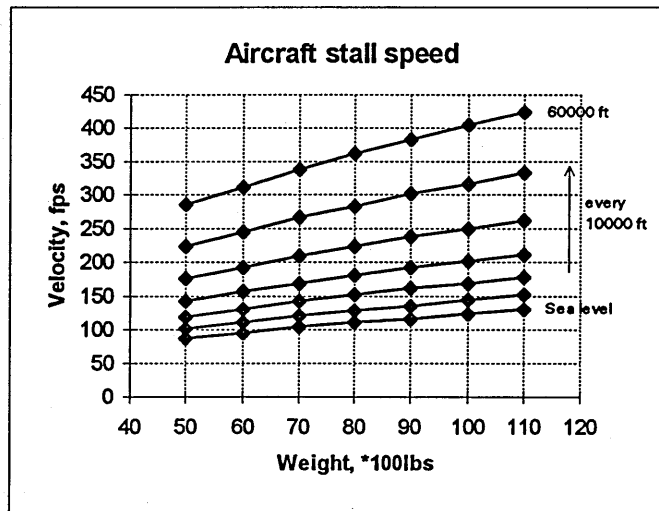


Fig. 2.8.2 Stall speed

In fig. 2.8.2, it is very easy to see that if the altitude is fixed, the aircraft stall velocity increases with aircraft weight. And if the aircraft weight is fixed, then the stall velocity increases with altitude.

The detailed calculation is listed in appendix A.9.

### 2.8.3 Engine performance

Before analysing the aircraft performance, the detailed engine data map is required.

The aircraft performance is very dependent on the real engine data, such as specific fuel consumption, net thrust etc. Many different sources were consulted, but no real engine data was obtained.

Due to lacking of real engine data, it was decided to collect the similar engine data from published magazines, such as references 12 and 26. From these, the basic engine data such as engine diameter, length, bypass ratio, pressure ratio, mass flow rate, and engine weight was obtained.

On the basis of these, some assumptions for our engine were made, and an engine simulation program, reference 36, used to generate a “simulated” engine data map.

In the beginning, based on the assumed engine basic parameters, many possible sensible arrangements, such as evenly distribute engine total pressure ratio 21 to fan and compressor and adjust turbine inlet temperature, were used to generate different results. From these, the best engine results to do aircraft performance analysis were chosen.

The detailed simulation analysis is in appendix A.10, here only one value of total pressure, 21, engine inlet temperature, 1480<sup>0</sup>F, mass flow rate, 140 lb/s, and the static net thrust, 5,459 lbs from the simulation results is presented in fig. 2.8.3.

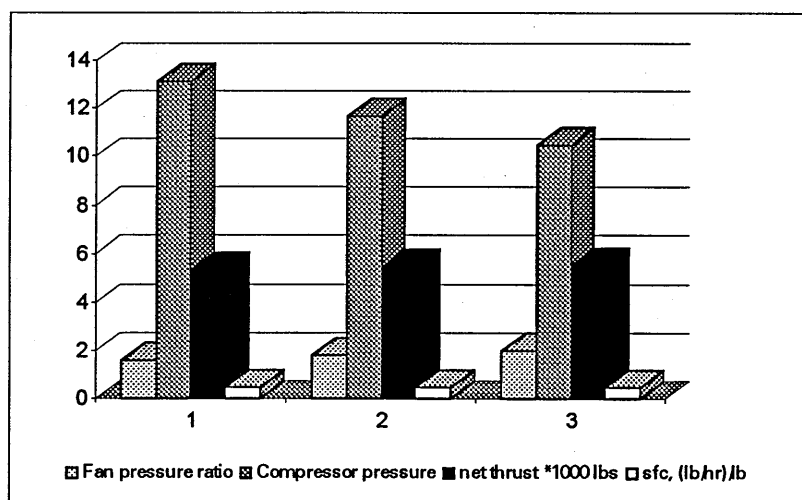


Fig. 2.8.3 Engine simulation design point results

In fig. 2.8.3, the number 1, 2, 3, denote three different pressure ratio combinations, detailed information is listed in appendix A.10.

Finally one single result, the design point data, which is closest to what is required is selected, These engine parameters were then used in the engine simulation program, for a range of different Mach numbers, from sea level design point to 66,000 ft, to generate the engine data for off design condition.

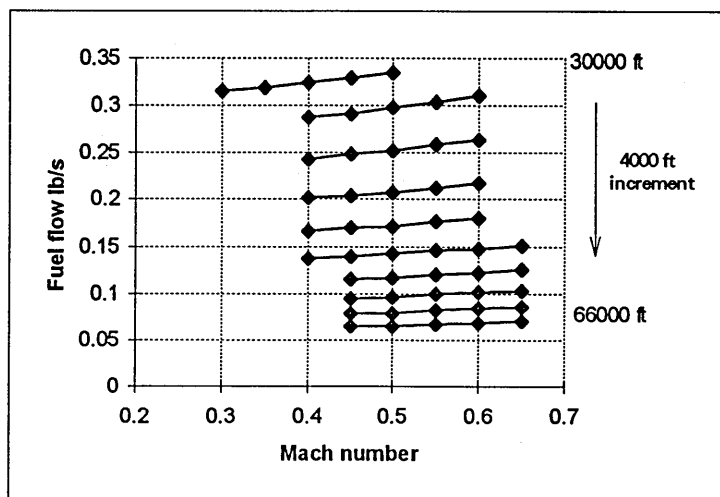


Fig. 2.8.4 Engine Mach number VS Fuel flow

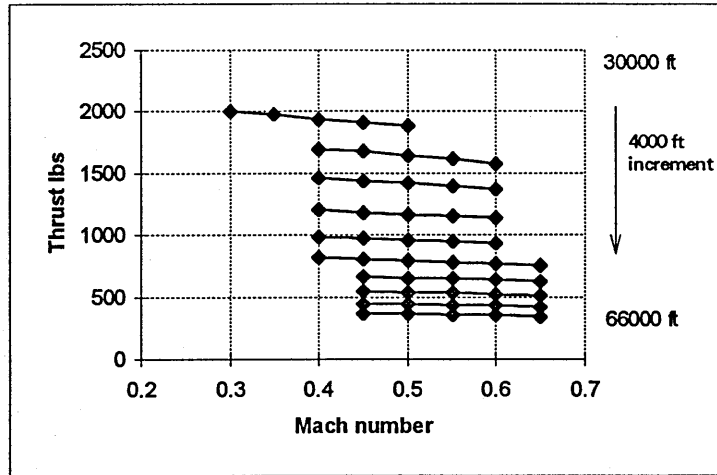


Fig. 2.8.5 Mach number VS Thrust

In order to make sure that the results of engine data is what the aircraft needs, the following analysis are performed.

From chap. 2.4.1, the thrust weight ratio of the HALE, during cruise, compare to sea-level, is 0.46, and from table 2.5.2, the aircraft take-off weight is around 10,490 lbs. Then the design point static net thrust at sea level should no less than 4825.4 lbs.

In addition, the HALE is designed to cruise at 65,000 ft. Reference 34, p:135, states that the thrust characteristics of turbofan engine varies with the altitude. The reason is that the density of the air directly entering the turbofan engine will decrease as the ambient air density becomes less. The following equation, from reference 34, eq. 4.7, may be simply used to represent the thrust decreasing rate with altitude.

$$\frac{T}{T_{SSL}} = \frac{\rho}{\rho_{SSL}} = \sigma \quad (2.8.3)$$

In eq. 2.8.3, the subscript SSL represents standard atmosphere sea level condition, the capital letter T represents the thrust, and the Greek letter  $\rho$  stands for density.

From eq. 2.8.3, if we take the sea level and 65,000 ft as an example, the density is 0.002377 lb-s<sup>2</sup>/ft<sup>4</sup> and 0.000177 lb-s<sup>2</sup>/ft<sup>4</sup> respectively. Then the density ratio of 65,000 ft to sea level is 0.074. That's means if the turbofan engine generate one pound thrust at sea level, when the altitude reaches to 65,000 ft, the thrust will reduce to 0.074 pound. Thus, if aircraft weight is 8200 lbs, cruise velocity is 0.55M at 65,000 ft, from reference 15, eq. 15.2, the thrust needed to maintain a level flight may be estimated.

$$T = D = C_D \frac{\rho}{2} V^2 S \quad (2.8.4)$$

From previous section, the same calculation can be used to obtain the drag coefficient, after calculation, the drag coefficient is 0.0439. With the wing area as 380 ft<sup>2</sup>,

substituting these variables into eq. 2.8.4, the thrust, to maintain level flight is 418.5 lbf. When convert to sea level static thrust, it is equal to 5656 lbf at sea level.

From above calculation, the net thrust at sea level should be 5,656 lbf. The simulated engine net thrust at sea-level is 5,459 lbf. This is close to the requirement estimated and thus us used for the subsequent calculation.

## **2.8.4 Climb performance**

Fast deployment of the HALE to high altitude is one major requirement in its specification. However, the ability to climb fast is mainly dependent on the engine net thrust. Thus, if the aircraft has a powerful engine, then the climb performance will be better.

From Chap. 2.8.3, the engine thrust is big enough to meet our requirement. The next step is to choose an efficient climbing speed. According to reference 66, in terms of safety, the climbing speed should be higher than stall and the minimum drag speed.

Reference 15, fig. 15.9, shows the DC-10 climb from sea level to 42,000 ft. During climbing, it is easy to tell that the climb speed is not constant. The discontinuities in the curves are due to changing the climb speed from one constant indicated airspeed to another. Since the HALE is climbing from sea level to 52,000 ft, it is decided to divide the climb into four segments. First, from sea-level climb to 22,000 ft, with constant indicated airspeed 338 fps. Second, from 22,000 to 34,000 ft, with constant indicated airspeed 270.4 fps. Third, from 34,000 to 44,000 ft, with constant indicated airspeed 236.6 fps. Fourth, from 44,000 to 52,000 ft, with constant indicated airspeed 210 fps.

In order to ensure that the climb speed has been chosen correctly, we have to obtain the stall speed and minimum drag speed. From appendix A.9, table A.9.2 and appendix A.11, table A.11.1, the stall speed and the minimum drag speed are given. The results are presented at fig. 2.8.6.

From fig. 2.8.6, it is obvious that the climb speed is higher than aircraft minimum drag speed and stall speed. Thus, the climb speed meets the requirement.

Since the detailed engine data such as engine net thrust, fuel flow and specific fuel has been calculated in appendix A.10, we may calculate the time, fuel, and distance to climb during climb.

From appendix A.11, the detailed calculation of the rate of climb, time of climb, fuel flow of climb, and distance to climb has been estimated respectively.

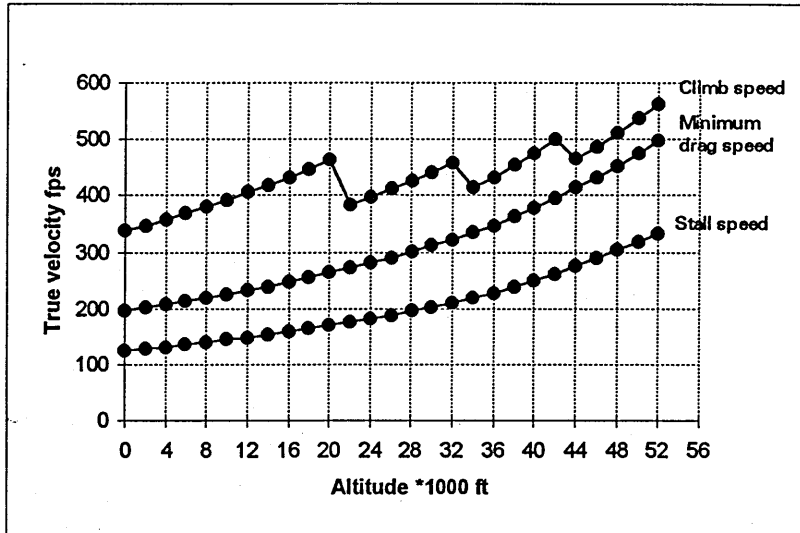


Fig. 2.8.6 Climb speed VS Stall and  $V_{min}$  speed

After calculation, the results of climbing from sea level to 52,000 ft are presented in fig. 2.8.7. The total climbing time (from sea level to 52,000 ft) is around 19 minutes, the total fuel consumption is around 337 lbs, and the total climb distance is 86 nautical miles.

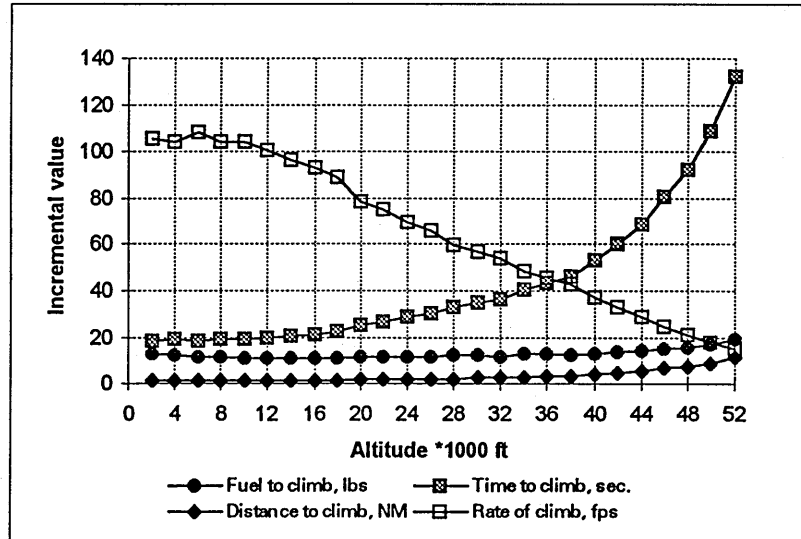


Fig. 2.8.7 Climb performance



### **2.8.5 Minimum drag velocity**

As the aircraft reaches 52,000 ft, from previous calculation, including the fuel consumption, the actual weight may be obtained. From this point on, the HALE is going to cruise at high altitude for 24 hours. For such a long period of cruise, we need to reduce the fuel consumption to as much as possible. Reference 15, eq. 15.5, may be used to calculate the minimum drag speed. For jet-powered aircraft, the fuel flow depends on the thrust and is expressed as specific fuel consumption, and minimum thrust will give the least fuel flow and the longest endurance. Therefore, minimum drag speed is the speed for best endurance for jets.

From appendix A.11.2, the minimum drag speed, at 52,000 ft, steady-state level flight, is 508.3 fps.

Since the speed and aircraft weight are known, then these can be substituted into the drag coefficient subroutine (in appendix A.8), after calculation the lift drag ratio is 23.7, and the lift coefficient, at this moment, is 0.7.

### **2.8.6 Modification of the wing body incidence angle**

From section 2.7.1, according to the assumption, the altitude is at 60,000 ft, the aircraft velocity is 0.55M, the lift coefficient during cruise is 1, and the weight at take-off is 10,489.7 lbs. After calculation, we obtained that the value of the wing body incidence is 6.796 degree and the lift drag ratio is 22.3.

From section 2.8.5, the value of lift drag ratio is 23.7. The difference of these two lift drag ratios is 1.4. From fig. 2.3, the program flowchart, if the difference is more than 0.5, we have to do the wing body incidence angle modification.

Therefore, we substitute the aircraft weight as 10,152.7 lbs, the altitude as 52,000 ft, the aircraft speed as 508.3 fps, and the lift coefficient as 0.7 into appendix A.7 to recalculate the wing body incidence angle. After calculation, the wing body incidence angle is 4.266 degrees, and the lift drag ratio is 23.8.

Now the difference of these two lift drag ratio is 0.1, according to our requirement, this is acceptable. Therefore, the value of 4.266 degrees has been taken as wing body incidence angle.

### 2.8.7 Cruise performance

According to chap. 2.1, fig. 2.1.1, the total cruise time, from the mission specification, is 24 hours, and the cruise speed of the HALE is within the 0.5 to 0.6 M range, the altitude is from 50,000 to 65,000 ft.

There are 24 hours in the cruise segment, and the aircraft will cruise within 50,000 to 65,000 ft. Also, by the end of climb, the altitude is 52,000 ft. Reference 39 suggests three methods to calculate the cruise range and endurance. From these, we selected the method for cruise air range at constant lift drag ratio, the cruise-climb method. This method is the most economical one in terms of fuel consumption.

Initially, we begin to cruise-climb from the altitude 52,000 ft, and the aircraft weight is around 10,158 lbs. Since the cruise requirement is 24 hours, and the altitude is within 50,000 to 65,000 ft. Thus, we use cruise-climb method and let the aircraft keep climbing until it reaches 65,000 ft. However, if before reaching 65,000 ft, the flight time is 24 hours, then we say the cruise performance calculation is finished. Otherwise, if the altitude reaches 65,000 ft, and the flight time is not 24 hours, then for the remaining hours we will use the method, of cruise at constant V or M, to do the cruise performance calculation.

The detailed calculations on cruise performance are presented in appendix A.12, and after 21 hours climb, the aircraft has already reached the altitude of 65,000 ft. According to the requirement, it still needs another three hours steady state level flight. The results of cruise performance and the wing lift, zero drag coefficient are listed in fig. 2.8.8 and fig. 2.8.9 respectively.

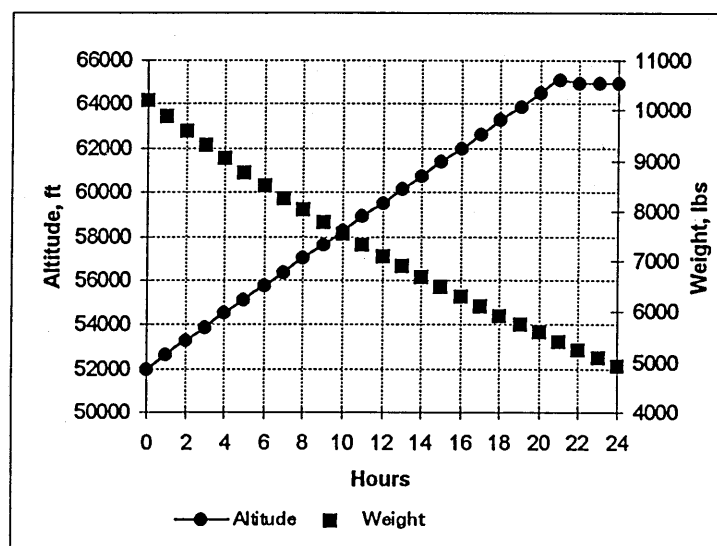


Fig. 2.8.8 Cruise performance

The reason for only listing the wing lift and zero drag coefficient in fig. 2.8.9, is that the proportion of this two wing drag coefficient during cruise is from 71.6 to 74.5 percent of the total drag coefficient. Therefore, in order to emphasise its importance, compared with other components drag coefficient, we only showed the wing drag coefficient in this figure.

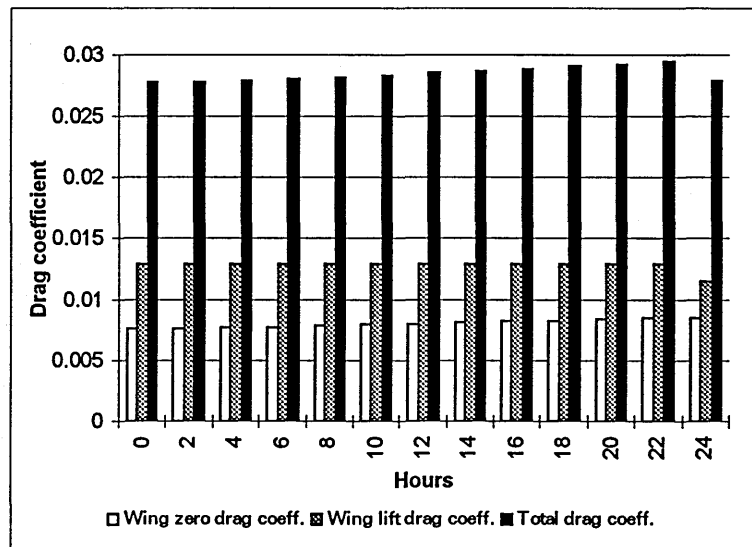


Fig. 2.8.9 Variation in wing zero, induced and total drag coefficient during cruise

## 2.8.8 Cruise return and descend to sea level

At the end of cruise, there is a cruise return from 65,000 ft to 52,000 ft. According to the mission requirement, during this segment the return cruise distance is 500 nautical miles. Assuming the aircraft descends 13,000 ft in 2 hours, equivalent to the descend rate as 1.8 fps. Assuming for the first hour the aircraft speed is 0.54M (the same speed as in cruise) and the second hour this reduces to 0.5 M, with the descend rate as 1.8 fps, after 2 hours descent, the aircraft weight becomes 4,663.55 lbs, the altitude is 51,608 ft and the travel distance is 606 nautical miles.

Then there is a descent from 51,608 ft to sea level. Assuming the descent time and number of segments are the same as in climb, first, from 51,608 to 44,000 ft, with constant airspeed 0.5 M. Second, from 44,000 to 34,000 ft, with constant airspeed 0.4 M. Third, from 34,000 to 22,000, with constant airspeed 0.3 M. Fourth, from 22,000 ft to sea-level with constant airspeed 0.25 M.

According to engine source data, at the same speed and same altitude, the fuel flow at throttle idle is approximately one seventh the fuel flow at maximum throttle.

Therefore, by taking the average fuel flow at each segment and divided by 7, and multiplying by the descent time, the total fuel consumption during descent is 44.28 lbs, and the aircraft weight becomes 4619.27 lbs (the detailed calculation is presented at appendix A.13)

### 2.8.9 Take-off weight comparison

According to fig. 2.3, the flowchart diagram, after carrying out the mission, a weight comparison is performed with the final aircraft weight compared with the empty weight of the aircraft plus the weight of payload.

This comparison, shows a difference of above weights of the aircraft is still more than 75.8 lbs of fuel. Thus, we may go back and reduce the fuel quantity available at the beginning, then go through the same process again.

According to our requirement, the difference of the aircraft final weight and the empty weight of the aircraft plus the weight of payload, should less than 10 lbs. In this case, as shown in the detailed calculation listed in appendix A.14. After five iterations, finally the take-off aircraft weight is reduced to 10,337.6 lbs and it still can meet our requirement.

In order to emphasise the difference between take-off weight and the excess fuel weight, we present the vehicle weight predicted and the excess fuel weight, during each iteration, in fig. 2.8.10 and fig. 2.8.11 respectively.

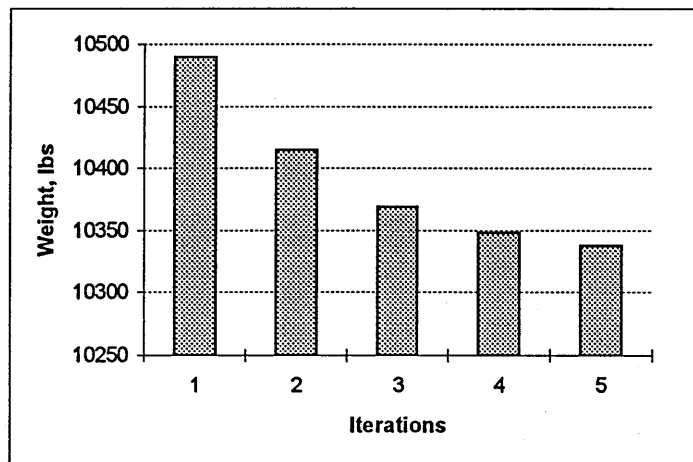


Fig. 2.8.10 Weight difference in each iteration

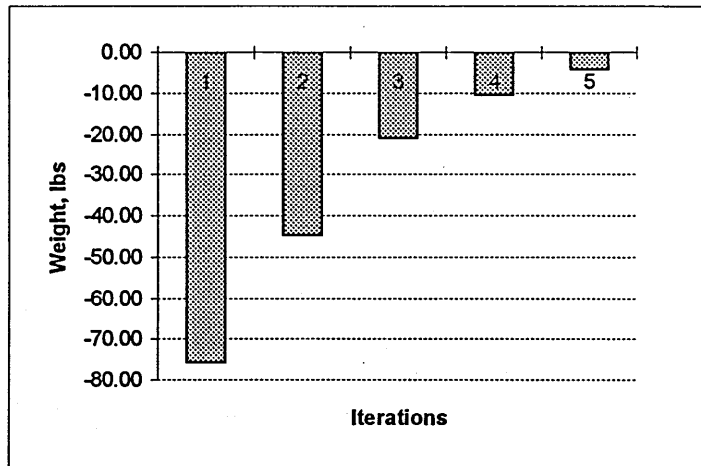


Fig. 2.8.11 Excess fuel in each iteration

There are five iterations in fig. 2.8.10, and fig. 2.8.11. During each iteration, if the weight in fig. 2.8.11 is negative, which means after the mission, the aircraft mission fuel is more than the fuel required. Therefore, at next iteration the excess fuel will be deducted. Finally after five iterations, the excess fuel, in fig. 2.8.11, has reduced to 4.3 lbs, which meets the requirement of the weight difference is within 10 lbs and iteration stopped.

## 2.8.10 Take-off performance

From reference 33, p:138 to 142, and reference 15, p:290 to 295, take-off is the transition from ground operation to flight operation. The take-off performance problem is basically an acceleration to the required speed plus a climb segment to a certain height (turbine-powered this is 35 ft). The required runway length is defined as the distance from the start of take-off to the point where this obstacle heights is reached.

Reference 7, chap. 5.2, the take-off distance may be calculated by eq. 5.6. After calculation, the aircraft take-off ground distance,  $S_{TOG}$ , is 772 ft, and the total take-off distance,  $S_{TO}$ , is 1,205 ft. The detailed calculation is listed in appendix A.14, and the take-off chart is shown in fig. 2.8.12.

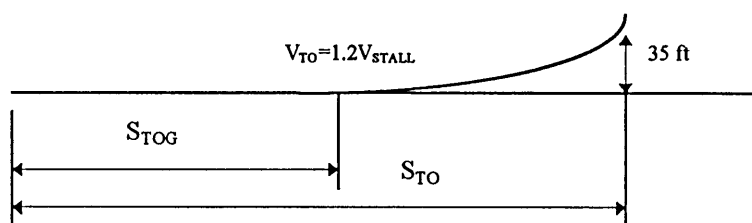


Fig. 2.8.12 Take-off distance

## 2.8.10 Landing performance

From reference 33, p:142 to 145, and reference 15, p:299 to 302, landing performance is basically take-off in reverse. It is a transition from flight to ground operation. The landing distance consists of two segments, the air run from a height of 50 ft to the surface accompanied by a slight deceleration and flare, and the ground deceleration from the touchdown speed to a total stop.

Using reference 7, eq. 5.81, the landing distance may be calculated. After calculation, the landing ground run distance,  $S_{LG}$ , is 457.6 ft, and the total landing distance,  $S_L$ , is 1,161 ft. The detailed calculation is listed in appendix A.15, and the landing chart is shown in fig. 2.8.13.

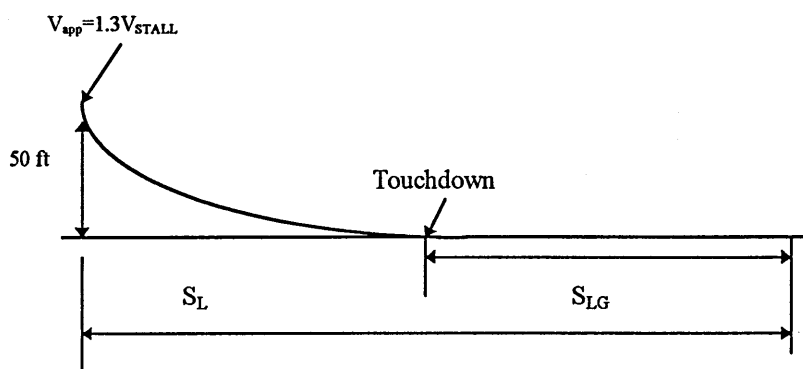


Fig. 2.8.13 Landing distance

## 2.9 Sensitivity analysis

According to chap. 2.4.1, from the parametric analysis, the aircraft aspect ratio as 15, the wing loading as 30 psf and the thrust weight ratio as 0.4589 were obtained. Based on these assumption, and following the program produced, we performed an aircraft conceptual design.

However, whether these assumption are the best solution or whether there are other possibilities, still needs to be verified.

This section considers if the aspect ratio or the wing loading changes will affect the aircraft sizing obtained.

### 2.9.1 Wing loading changes

Initially keeping the aspect ratio the same as in the previous section, and changing the wing loading range from 28 to 38. The effect on the design are explored.

After calculation, the wing loading to aircraft take-off weight relationship is shown in appendix A.16.1, table A.16.1 and A.16.2. Following figures provide a summary.

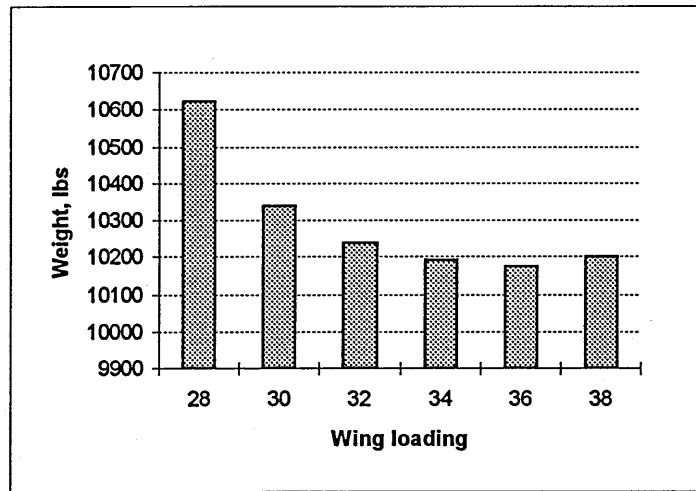


Fig. 2.9.1 Wing loading VS take-off gross weight

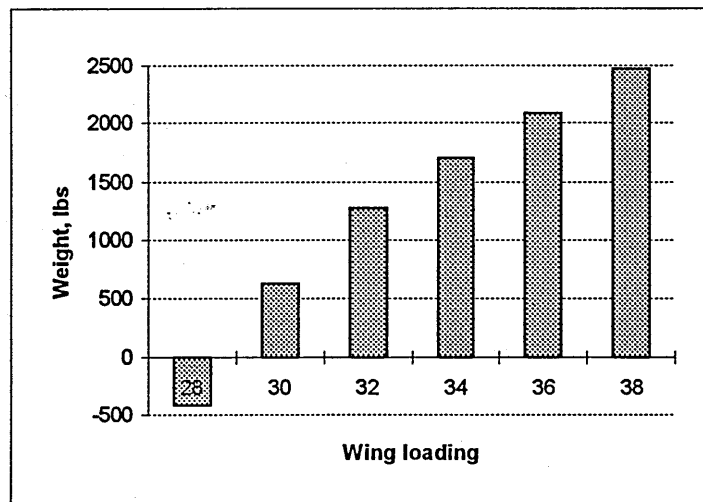


Fig. 2.9.2 Wing loading VS fuel weight discrepancy

From fig. 2.9.1, we find that the take-off gross weight decreases with increase in wing loading. This trend continues until the wing loading reaches a certain value, 36, then the aircraft take-off gross weight slightly increases again.

However, the benefit of higher wing loading and reduced take-off gross weight is not the only consideration. From fig. 2.9.2, we can easily see that as the wing loading increases, the wing area decreases and the wing fuel tank becomes smaller, and finally more and more fuel will need to be put into the fuselage.

For a high altitude UAV, we need to install every possible detection sensor in the fuselage. Therefore, if most of the mission fuel can be put into the wing fuel tank, then most of the fuselage can be used for surveillance equipment. From fig. 2.9.2, when wing loading varies from 30 to 38, we need to put from 630 to 2469 lbs mission fuel into fuselage. Therefore, which value of wing loading is the best, is dependent on both the mission and payload requirement.

Alternatively, we could make the fuselage slightly longer to have more space for sensors, but the more structure will cause weight and drag penalty. Therefore, whether it is a good idea of having longer fuselage still needs further analysis.

### 2.9.2 Aspect ratio

In this section the effect of changing the aspect ratio on the aircraft take-off gross weight is explored.

From previous section, the aircraft wing loading as 30 psf has been selected. We substitute the aspect ratio as 14, 15 and 16 into this conceptual design program to do more calculation (the detailed calculation is in appendix A.16.2), and summarised into figure 2.9.3.

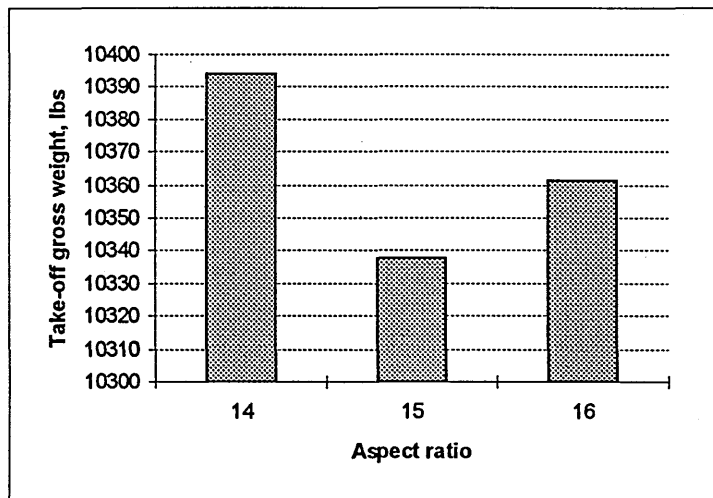


Fig. 2.9.3 Aspect ratio to take-off gross weight

From fig. 2.9.3, we find that at different aspect ratio, the take-off weight change is within 0.5 percent. Therefore, in this case, the aspect ratio changes did not have too much effect to the aircraft take-off weight, despite the wing mass increases as aspect ratio increases.

### 2.9.3 Discussion

Regarding to the results of previous section, in this paragraph, we are going to explore why the wing loading changes having some effect on the take-off gross weight.

As the wing drag is the dominant factor in the total drag, in following figures, we will discuss its effect to the aircraft take-off gross weight (the detailed aircraft cruise drag coefficients is listed in appendix A.16.3).



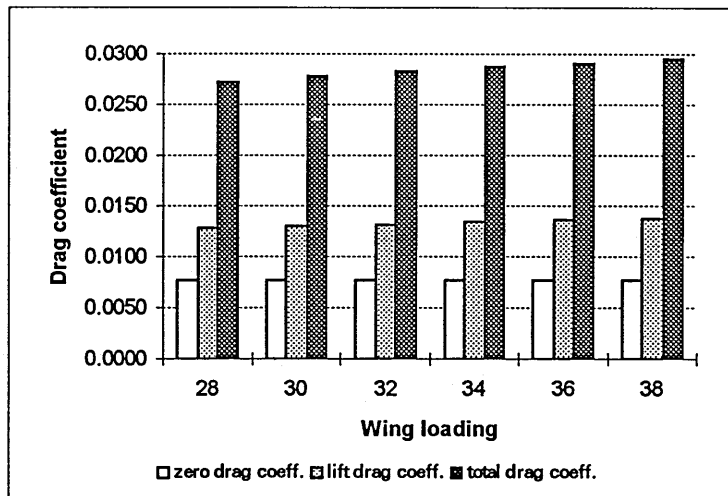


Fig. 2.9.4 Wing loading VS Drag coefficient

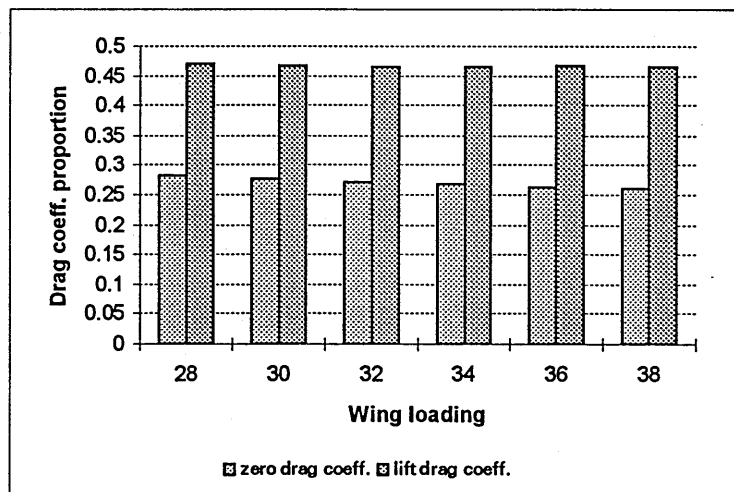


Fig. 2.9.5 Wing loading VS Drag coefficient

Fig. 2.9.4 shows that during cruise as the wing loading increases, the total drag coefficient, wing zero drag coefficient and wing lift drag coefficient are slightly increase.

Fig. 2.9.5 shows that as the wing loading increases, proportion of the lift drag coefficient to the total drag coefficient is about the same, but the proportion of the zero drag coefficient to the total drag coefficient will decrease.

However, in level and constant speed flight the drag is the same as the thrust. Reference 15, eq. 3.9, states that the drag is the function of density, velocity, wing area and drag coefficient. Therefore, the drag coefficient increase does not necessary mean the drag force will increase as well. In fig. 2.9.6 (the results of paragraph

A.16.3.2, table A.16.7), as the wing loading increases, the drag first decreases and then increases. As stated before, during level and constant flight, the drag is equal to the thrust. Therefore, the thrust varies like the drag, and thrust is related to fuel consumption. The more thrust needs to be generated, the more fuel needs to be consumed.

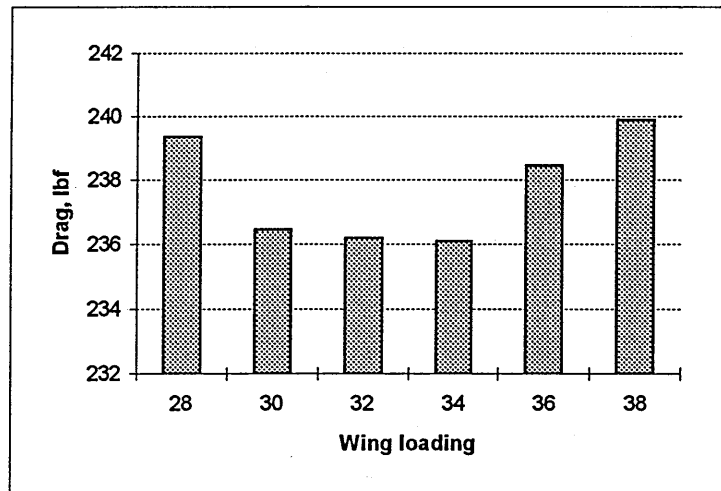


Fig. 2.9.6 Wing loading VS Drag

In fig. 2.9.6, the highest drag happens at wing loading as 38, and from wing loading as 28 the drag decrease until wing loading as 34, and slightly increases again. This proves the results in fig. 2.9.1, as the wing loading increases, the trend of take-off gross weight first decreases then slightly increases.

Due to the wing drag having the largest influence on the total drag, the lift drag coefficient plus the zero drag coefficient is more than 70 percent of the total drag coefficient. Therefore, if the wing drag can be reduced, such as by using a laminar flow airfoil, then the total drag can be reduced as well. That means the thrust required for aircraft to maintain level flight can be brought down, and fuel consumption may be decreased, and finally the take-off gross weight may be further cut down.

## 2.10 Stability analysis

In previous sections, we discussed aircraft drag, rate of climb, cruise altitude and engine type. Such parameters are of paramount importance in aircraft design. Analysing such parameters and incorporating the results into the design, we can determine the fuel requirements for conducting a certain mission, and also the thrust requirements. However, there is another equally important parameter which should be considered, safety, the aircraft needs to fly in a stable situation.

According to reference 15, chap. 16, the stability is defined as the ability of a vehicle to return to its original equilibrium position, without other assistance, after a disturbance has pitched, yawed or rolled it to a different angle or has caused a speed change.

### 2.10.1 Static stability

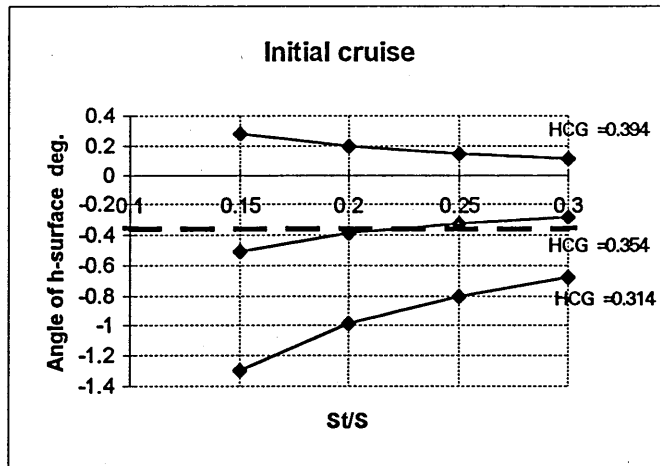
Static stability refers to the initial tendency of a system to return to or move away from an equilibrium position after a disturbance. According to reference 45, the static stability of the HALE is estimated as follows :

#### 2.10.1.1 Trim in cruise

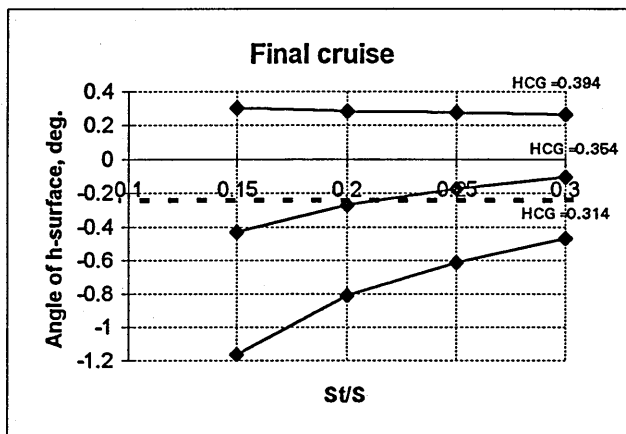
For an airfoil with trailing edge control surfaces, it is usual to establish a value of angle of incidence of horizontal surface,  $\alpha_T$ , so that in cruise the variation of elevator deflection,  $\eta$ , through the centre of gravity range is as near zero as possible.

The mission of the HALE, according to its specification, is 24 hours on station loiter. After such a long time loiter, the aircraft weight will have halved. In order to obtain a reasonable figure of  $\alpha_T$  during cruise, it was decided to take both extreme conditions during cruise, i.e. initial cruise and final cruise, as a special case to calculate the value of angle of incidence of horizontal surface,  $\alpha_T$ .

Following reference 45, the angle of incidence of horizontal surface,  $\alpha_T$ , during cruise for trim may be calculated. The detailed calculation is listed in appendix A.18.1, in this section, we only put the results into fig. 2.10.1 (a), (b).



( a )



( b )

Fig. 2.10.1 Angle of incidence of horizontal surface,  $\alpha_T$ , for trim in cruise

Since the range of aircraft C.G. movement is very small, it was decided to take the C.G. as the 0.354 line as aircraft C.G., and take the average value of  $\alpha_T$ , among  $s_T/s$ , finally we obtain the angle of incidence of horizontal surface at the initial of cruise, for trim, as -0.376 degrees.

Again at final cruise, we obtain the angle of incidence of horizontal surface, for trim, as -0.24 degrees.

### 2.10.1.2 Trim on approach to land

From section 2.10.1.1, we obtain the angle of incidence of horizontal surface,  $\alpha_T$ , for trim during cruise, as -0.376 degrees at initial cruise and as -0.24 degrees at final

cruise. By substituting this value into eq. A.18.1, the elevator deflection,  $\eta$ , for landing approach trim may be estimated.

As we did in section 2.10.1.1, two different angles of elevator deflection,  $\eta$ , are obtained, the detailed calculation is listed in appendix A.18.2.

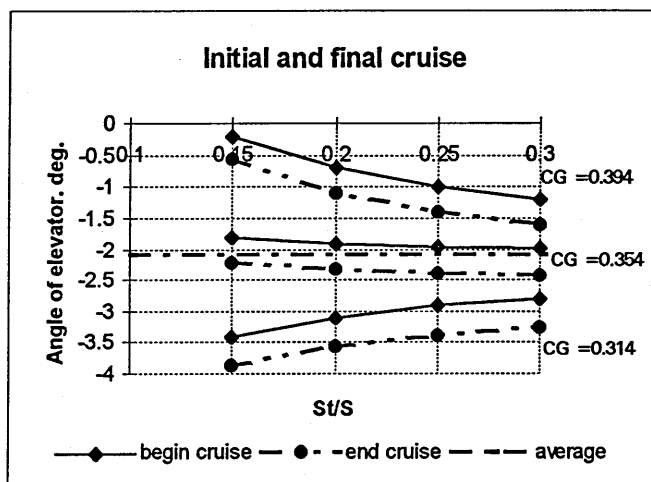


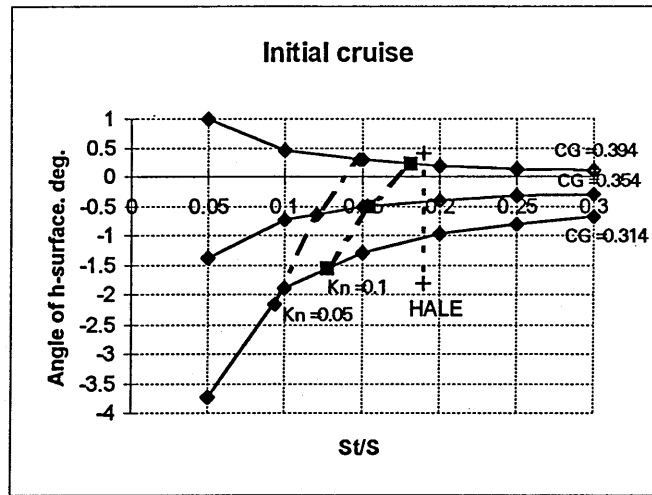
Fig. 2.10.2 Angles of elevator deflection,  $\eta$ , trim for approaching

According to reference 45, the maximum positive and negative elevator angles available may be  $\pm 20$  degrees. Unless the elevator angles to trim significantly exceed 67 percent of the available movement, there is no need to recalculate the value of the angle of incidence of horizontal surface. From fig. 2.10.2, the trim angle of elevator deflection, at initial and final case, do not exceed of -2.3 degrees. Therefore, in accordance with reference 45, there is no need to adjust the horizontal surface angle.

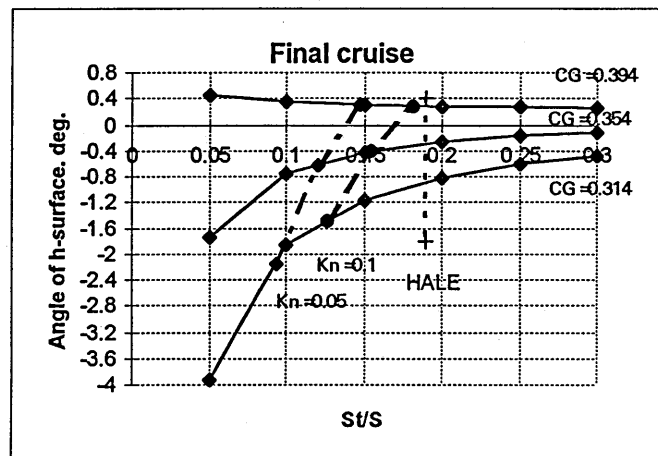
### 2.10.1.3 Cruise static stability

Assuming a static margin,  $K_n$ , of 0.05 or 0.1 is required, from reference 45, eq. 3, the necessary ratio of tail to wing area may be calculated.

As before, we take two extreme points in cruise, initial and final as an example for calculation. The detailed calculation is listed in appendix A.18.3, the results are listed in fig. 2.10.3.



(a)



(b)

Fig. 2.10.3 Cruise static stability

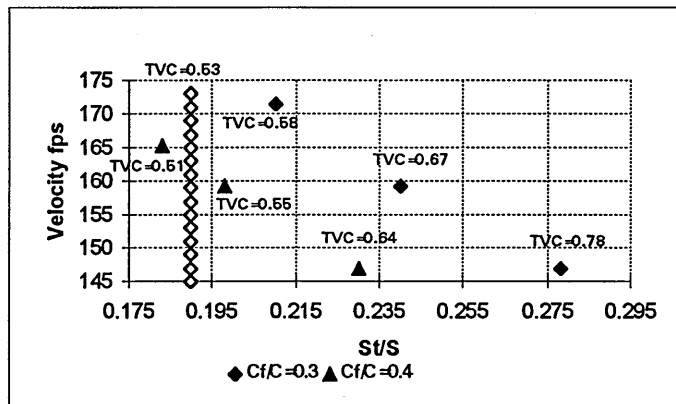
In both cases, if the location of aircraft C.G. is at 0.354, regarding the wing MAC. No matter whether the static margin,  $K_n$ , is 0.05 or 0.1, the value of  $s_r/s$ , during cruise, does not exceed 0.16.

For HALE, the value of  $s_r/s$ , is 0.19. Therefore, the cruise static stability meets the recommended requirements.

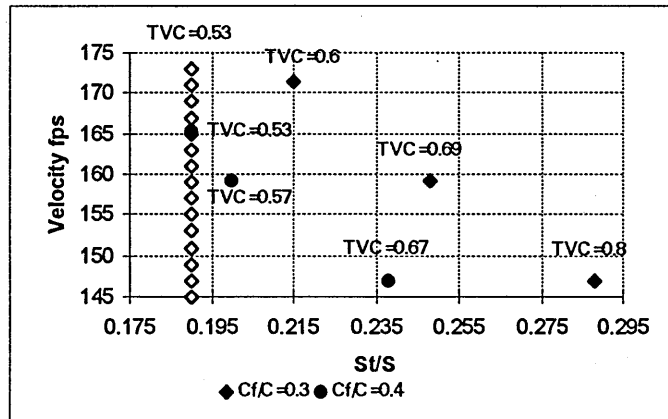
#### 2.10.1.4 Rotation at take off

Assume the elevator deflection angle,  $\eta$ , during take off is -25 degrees. From reference 45, eq. 1, the value of  $s_r/s$ , at take off rotation, may be estimated.

The proportion of control surface chord to wing chord has a big impact to the lift coefficient due to control surface deflection. Therefore, the following calculation is to ensure that the two different proportions of control surface chord with respect to tail chord, 0.3 and 0.4, the value of tail volume coefficient and  $s_r/s$  may meet the requirements during take off ( detailed calculation is listed in appendix A.18.4). For HALE, the value of  $s_r/s$ , is 0.19. Thus, from appendix A.18.4, eq. A.18.31, the value of tail volume coefficient,  $\bar{V}$  (TVC), is 0.53.



( a )  $\alpha_T$  is -0.376 degrees



( b )  $\alpha_T$  is -0.24 degrees

Fig. 2.10.4 Rotation at take-off

From Fig. 2.10.4 (a), (b), if the proportion of control surface chord to wing chord,  $c_f/c$  is 0.3, irrespective of whether the rotation speed is 1.2 ,1.3 or 1.4 multiplied by the stall velocity, the value of tail volume coefficient,  $\bar{V}$  and the value of  $s_r/s$ , do not meet our requirement.

But if we change the proportion of control surface chord to wing chord,  $c_f/c$  as 0.4, when the rotation velocity is 1.35 multiplied by the stall velocity, the value of tail volume coefficient,  $\bar{V}$  and the value of  $s_r/s$ , meet our requirement.

## 2.10.2 Dynamic stability

According to reference 15, chap. 16, dynamic stability concerns the entire history of motion, and in particular the rate at which the motion damps out.

It has been stated that the longitudinal and lateral motions are usually quite independent of each other and, therefore, can be considered separately. But for conceptual design, it was decided to only take longitudinal motion into consideration.

The longitudinal motion consists of two distinct oscillations, a short period oscillation called short period mode, and a long period motion called the phugoid mode.

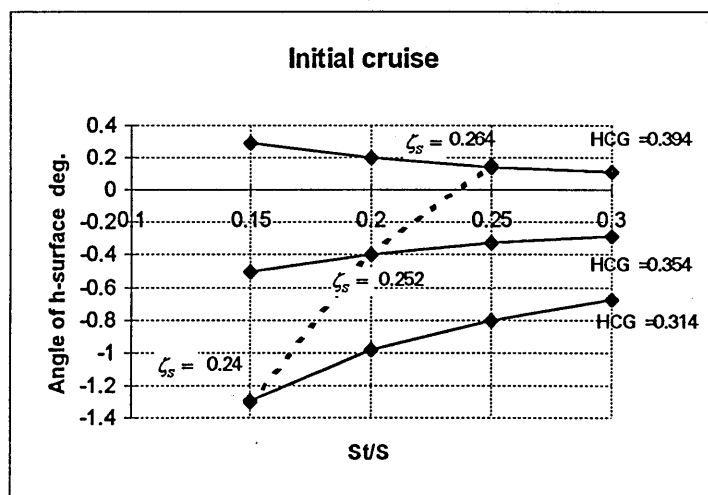
In the following paragraph we will discuss these two modes.

### 2.10.2.1 Short period oscillation

From reference 45, satisfactory handling qualities are determined by a combination of both frequency and damping in the short period mode. The short period mode is very fast oscillation, usually heavily damped. Such oscillations have a period of a few seconds.

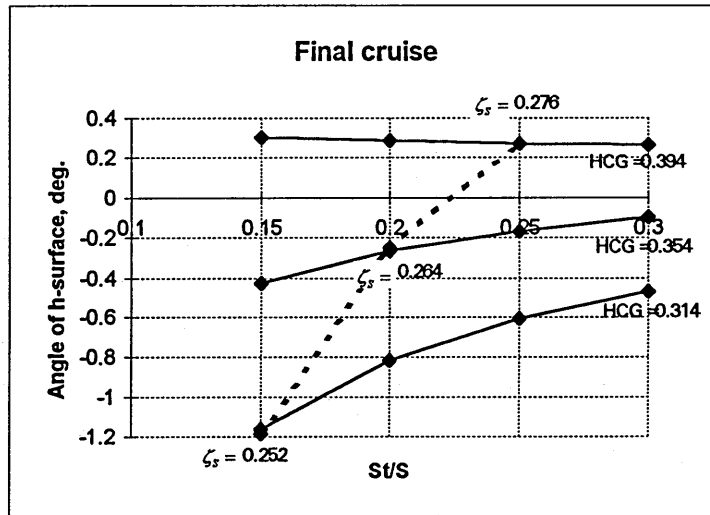
a. Short period damping ratio,  $\zeta_s$

The detailed calculation is listed in appendix A.18.5, the results of both initial and final cruise, the values of damping ratio are listed in fig. 2.10.5 (a), (b).



( a )





(b)

Fig. 2.10.5 Short period damping ratio,  $\zeta_s$

b. Undamped short period frequency,  $\omega_s$

From reference 45, eq. 5, at the value of  $s_r/s$  as 0.2, the undamped short period frequency,  $\omega_s$ , at initial and final cruise may be estimated. The results are listed in fig. 2.10.6.

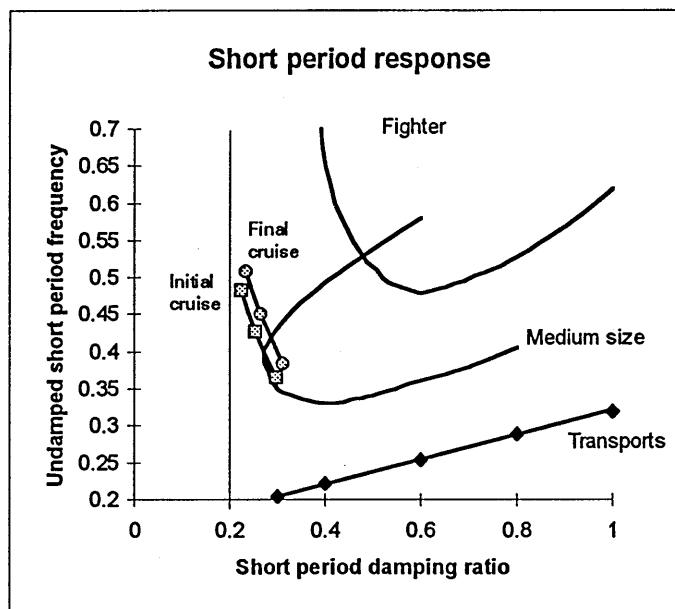


Fig. 2.10.6 Undamped short period frequency,  $\omega_s$

Fig. 2.10.6 is based on handling quality for manned aircraft and has been used here to give an indication of the likely behaviour of the unmanned aircraft in terms of control and stability. However, for unmanned aircraft, it is easily to use artificial control system to compensate the difficulties of control and stability, the impacts will be the technology risk and cost increases.

### 2.10.2.2 Long period mode

The phugoid, long period mode, is a slow interchange of kinetic energy and potential energy about some equilibrium energy level. The motion is slow, and the damping is very weak. The period is usually 20 to 60 seconds.

Evaluation of phugoid characteristics is difficult. In practice the frequency of the motion is of little consequence, the critical issue is the damping ratio. From reference 45, appendix A, eq. A.16, may be used to estimate the damping ratio in the phugoid mode.

The detailed calculation is in appendix A.18.5.2, and the results are listed below.

The value of  $\zeta_L$ , of HALE is 0.19. As previously, for the initial and final cruise, the damping ratio in the phugoid mode is:

a. Begin cruise

a.1 For aircraft C.G. is at 0.314,  $\zeta_L$  is 0.019.

a.2 For aircraft C.G. is at 0.356,  $\zeta_L$  is 0.0186.

b. End cruise

a.1 For aircraft C.G. is at 0.314,  $\zeta_L$  is 0.0189.

a.2 For aircraft C.G. is at 0.354,  $\zeta_L$  is 0.0184.

According to reference 45, the damping ratio should, ideally, be about 0.2. However, this value may be difficult to achieve. Instead, a significant lower damping ratio assisted by a flight control system can provide adequate stability.

## **Chapter 3**

### **Validation and application**

In chapter 2, a preliminary conceptual High Altitude Long Endurance UAV design program has been constructed. In this chapter, Tier-II Plus, a HALE UAV with turbofan engine, and a new HALE UAV are used to perform validation and application. The former is not only an existing, latest UAV, but independent of this program, the latter is an example application of this program. From their results information on whether this program is feasible for conceptual design or need further modification can be gained.

#### **3.1 Introduction**

In May 1995, a team headed by Teledyne Ryan Aeronautical was awarded a contract by US air force to develop Tier-II Plus program. According to its specification, Tier-II Plus is intended to operate in low and moderate threat levels. Cruising at 65,000 ft, outside the envelope of most air defence systems. Loiter for more than 24 hrs, 3,000 nm from the take-off and landing point. Protected by decoys and jammers, the aircraft should have at least some capability of penetrating enemy air defence. This makes the system capable of near-global coverage from US-controlled base.

The first of two Teledyne Ryan Tier-II Plus vehicles is due to fly at the end of 1996. And the Tier-II Plus UAV is designed to have a unit flyaway cost of US\$ 10 million in Fiscal 1994 dollars.

### 3.2 Mission profile of Tier-II Plus

From reference 38, the Tier-II Plus air vehicle can take-off automatically, transit 3,000 nautical miles, loiter on station for 24 hours and then return to base. Fig. 3.2.1 is the mission profile of Tier-II Plus.

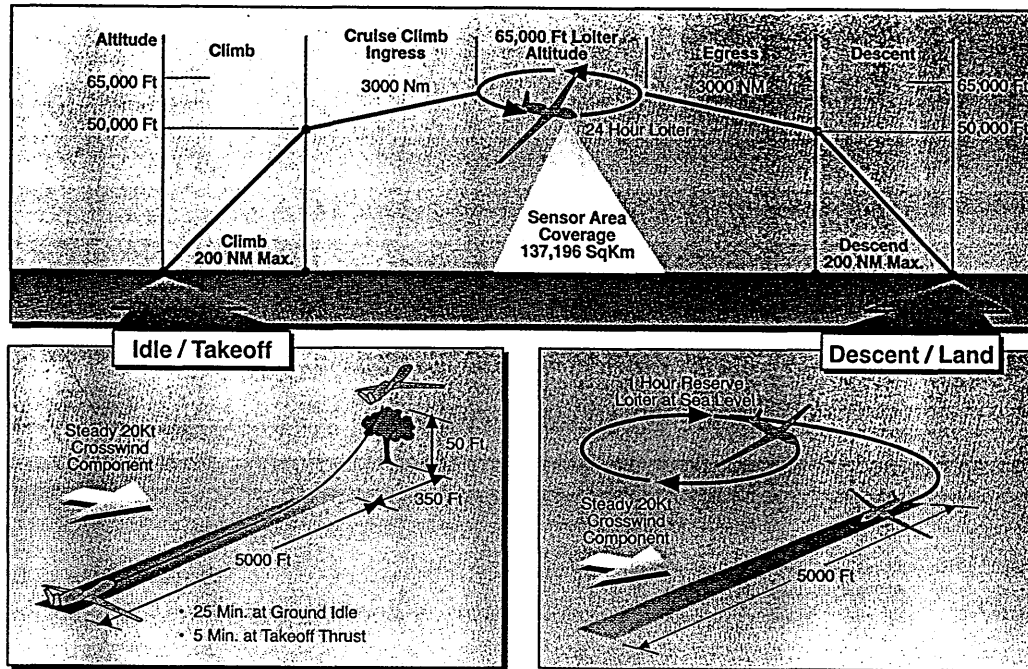


Fig. 3.2.1 Tier-II Plus mission profile (from ref. 38)

According to the objective of Tier-II Plus, it is to develop and demonstrate a “High Altitude Endurance UAV system capable of affordable, continuous, all-weather, wide-area surveillance in support of military operations.”

Based on above requirement, the specific Tier-II Plus UAV mission goals and requirement include:

- Take-off from a 5,000 ft military runway and climb rapidly to a 50,000 ft.
- Climbing from 50,000 ft to 65,000 ft, at 300 to 400 kt., flying 1,000 to 3,000 nautical mile to a desired surveillance area.
- Loiter, flying speed at 0.6 M, for at least 24 hours at an altitude up to 65,000 ft. Searching 40,000 sq. naut. mi. each day.
- Returning to base over a 1,000 to 3,000 naut. mi. egress segment, descending from 65,000 ft to 50,000 ft.
- Descending from 50,000 ft to Base, 1 hour reserve loiter at sea level, then landing at a 5,000 ft field.

### 3.3 Flowchart of Tier - II Plus

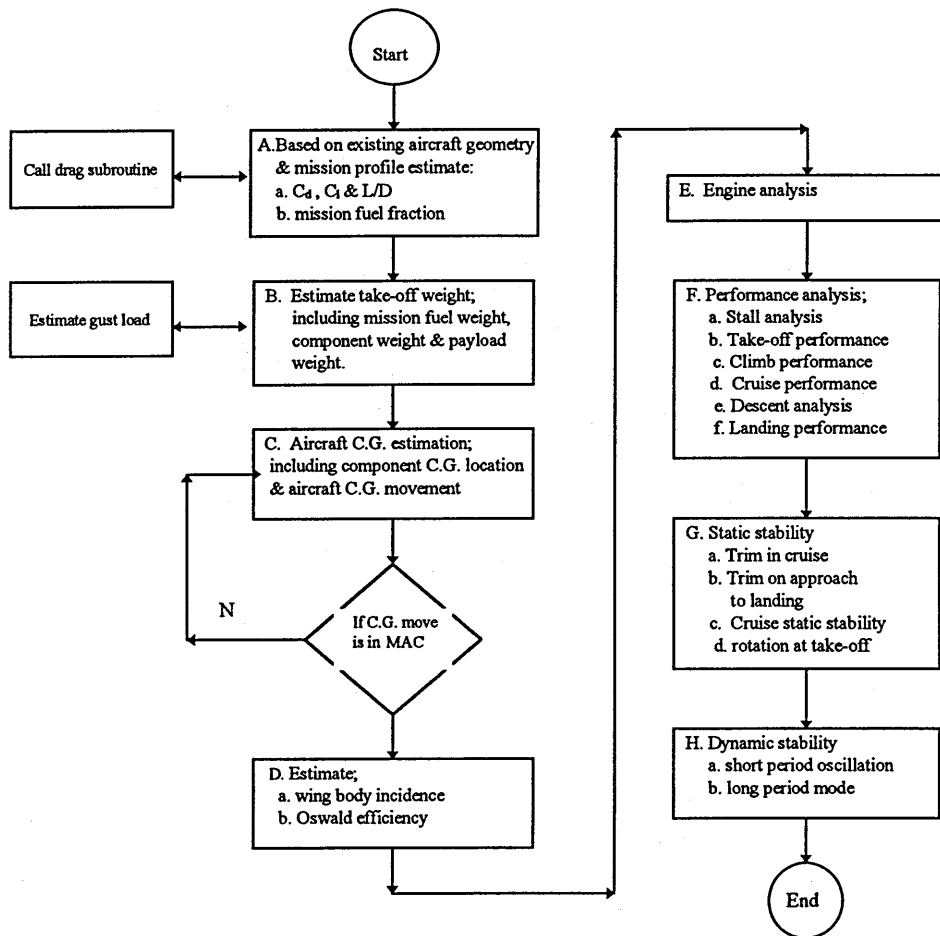


Fig. 3.3.1 Tier-II Plus flow chart

Due to the fact that the configuration of Tier-II Plus is fixed, the program flow chart is not exactly the same as in chap. 2.3. The differences are that all the items listed in block 1, in fig. 2.3, are removed, and with block 2, fig. 2.3, if the estimated weight and aircraft weight from specification are different, we do not iteration to alter the estimated weight. Finally, using the predicted values for Tier-II Plus to complete loiter in between 50,000 and 65,000 ft for 48 hours, we only discuss whether it is feasible for Tier-II Plus to fly such a long period of time, there is no iteration to reduce or add mission fuel to fulfil mission requirement.

In chap. 2, there is no existing aircraft available, therefore, we have to go through all the conceptual design process, finally, a preliminary conceptual design aircraft is

created. However, in this paragraph, the aircraft configuration is already being fixed. What us required is use of the previous program to evaluate the specification of Tier-II Plus. If it is found that Tier-II Plus is predicted to meet its requirement, then this program can to some extent be taken to be validated.

3.4 Basic aircraft information

Aircraft basic geometry data and other information is listed in table 3.4.1:

|                                                 |                           |
|-------------------------------------------------|---------------------------|
| Wing span                                       | 116.2 ft                  |
| Fuselage length                                 | 44.4 ft                   |
| Aircraft height                                 | 15.2 ft                   |
| Aircraft weight                                 | 22,914 lbs                |
| Payload                                         | 2,000 lbs                 |
| Endurance (ingress, egress & loiter on station) | 42 hrs                    |
| Engine ( Allison AE 3007)                       | 7,100 lbs (static thrust) |

Table 3.4.1 The basic geometric data of Tier-II Plus

The Tier-II Plus three view diagram is in fig. 3.4.1

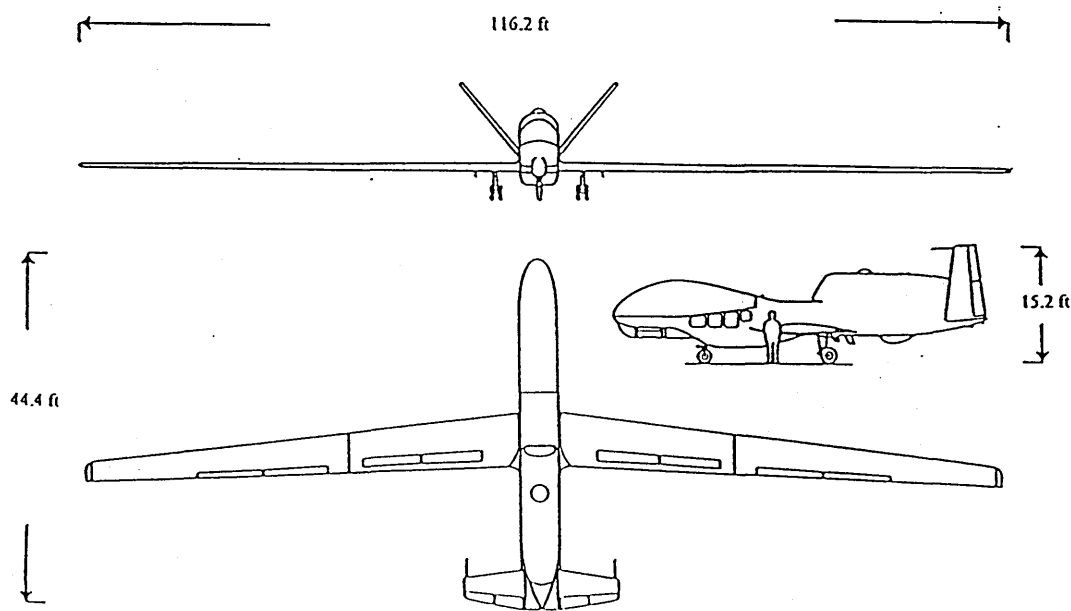


Fig 3.4.1 Tier-II Plus 3-view diagram (from ref. 26)

### 3.5 Initial lift/drag, fuel fraction analysis

Following the aircraft basic data listed in table 3.4.1 and aircraft three view dimension presented in fig. 3.4.1, we measure the size of wing, fuselage and tailplane of Tier-II Plus, and put the aircraft dimension in table 3.5.1(approximately).

|                         |                       |
|-------------------------|-----------------------|
| Wing                    |                       |
| Wing span               | 116.2 ft              |
| Gross area              | 523.8 ft <sup>2</sup> |
| Aspect ratio            | 25.78                 |
| Wing loading (T.O.)     | 43.7                  |
| Mean aero. chord        | 4.99 ft               |
| Root chord              | 6.875 ft              |
| Tip chord               | 2.38 ft               |
| Taper ratio             | 0.35                  |
| Root thickness          | 17 % (assumption)     |
| Tip thickness           | 13 % (assumption)     |
| Leading edge angle      | 7 deg.                |
| Trailing edge angle     | 2.365 deg.            |
| Fuselage                |                       |
| Length                  | 44.4 ft               |
| Diameter (Max.)         | 4.72 ft               |
| Tailplane (V - tail)    |                       |
| Tail-span (total)       | 31 ft                 |
| * Horizontal(projected) | 19.5 ft               |
| Tail area (total)       | 127.7 ft <sup>2</sup> |
| * Horizontal(projected) | 80.38 ft <sup>2</sup> |
| Aspect ratio            | 7.52                  |
| * Horizontal(projected) | 4.735                 |
| Tail mean aero. chord   | 4.325                 |
| Tilt angle ( vertical ) | 39 deg.               |
| Leading edge angle      | 7 deg.                |
| Trailing edge angle     | 5.37 deg.             |
| Root chord              | 5.8 ft                |
| Tip chord               | 2.38 ft               |
| Taper ratio             | 0.4                   |
| Thickness               | 12 %                  |

Table 3.5.1 Dimensions of Tier-II Plus

The first block in flow chart fig. 3.3.1, is to estimate the mission fuel fraction. Before estimating, the aircraft lift drag ratio has to be given. As stated before, the lift drag ratio is equal to the lift coefficient divided by the drag coefficient. Therefore, if the value of the lift coefficient and the drag coefficient can be solved, then the lift drag ratio is obtained.

In chap. 2, a subroutine of estimating aircraft component drag coefficient is given. Therefore, it can be used to estimate the drag coefficient of Tier-II Plus. For Tier-II Plus, the most important mission segment is loiter, 42 hours. So we decide to estimate the drag coefficient at loiter. In order to estimate, during cruise, the average aircraft drag coefficient, we decide to take the initial and final cruise point.

In using the drag subroutine to estimate aircraft component drag coefficient, there are some variables which are first required, such as Oswald coefficient and wing body incidence. However, at this moment, these variables are unknown. The aspect ratio of Tier-II Plus is 25.78, it is higher than conventional aircraft. Reference 10 stated that typically the value of Oswald efficiency is between 0.7 and 0.85. For Tier-II Plus, the value of 0.78 is chosen. Since there is no positive answer for the value of wing body incidence, the value as 4 degrees has been chosen (these values will be corrected during later calculation). Following is the estimation;

a. Initial cruise:

Initial cruise, the altitude of Tier-II is 50,000 ft. Assume its weight is around 22,000 lbs, flight speed is 0.6 M, then by calling drag subroutine.

b. Final cruise:

At final cruise, the altitude of Tier-II is at 65,000 ft, assuming its weight is around 12,000 lbs, flight speed is 0.6 M, then by calling drag subroutine.

Results are presented in table 3.5.2.

| Cruise  | Lift drag<br>coefficient | Zero drag<br>coefficient | Lift<br>coefficient | Drag<br>coefficient | Lift drag<br>ratio |
|---------|--------------------------|--------------------------|---------------------|---------------------|--------------------|
| Initial | 0.00877                  | 0.0128                   | 0.69                | 0.0216              | 31.6               |
| Final   | 0.011                    | 0.0145                   | 0.772               | 0.0255              | 30.3               |

Table 3.5.2 Lift drag ratio during cruise

From above calculation, the lift drag ratio is around 31.6 and 30.3, at the beginning and the end of cruise respectively.

As the above results are based on assumptions, the lift drag ratio predicted might be higher than in practice. To be conservative, the value of 31 and 30 are taken as the lift drag coefficient to estimate the mission fuel fraction at ingress and loiter respectively.



From fig. 3.2.1, there are seven different segments in the mission profile (As in chap. 2.4.2, the figure of the ratio of weight, in each segment, is based on assumption, it may be recalculated later).

#### **Phase 1. Engine start and take-off**

The same as in chap. 2.4.2, we take the fuel consumption as around 2.5 to 3.0 percent of take-off weight. Therefore, if this phase begins at  $W_1$  and ends at  $W_2$ , then the ratio of weight is 0.975.

#### **Phase 2. Climb and accelerate to 50,000 ft**

After take-off the airplane is climbing to 50,000 ft. During this phase the fuel consumption is about 3 to 3.5 percent of the weight. If the beginning weight is  $W_2$ , the end is  $W_3$ , then the ratio of weight is 0.965.

#### **Phase 3. Fixed speed cruise-climb from 50,000 ft to 65,000 ft**

Using the same equation as in chap. 2.4.2, eq. 2.4.1, if the  $W_3$  is the beginning weight, the end is  $W_4$ . Assume during cruise-climb the specific fuel consumption,  $sfc$ , is 0.65 lb/lb hr, cruising distance is 3,000 nautical miles, and the speed of cruise-climb is from 0.54M up to 0.6 M, then the ratio of weight is 0.897.

#### **Phase 4. Fixed speed loiter at 65,000 ft (E is 24 hrs, M is 0.6)**

Using the same equation as in chap. 2.4.2, eq. 2.4.2, if the  $W_4$  is the beginning weight, the end is  $W_5$ . During loiter the specific fuel consumption,  $sfc$ , is assumed as 0.67 lb/lb hr, loiter time is 24 hours, and loiter velocity is 0.6M, the ratio of weight is 0.585.

#### **Phase 5. Fixed speed cruise-descent from 65,000 ft to 50,000 ft**

The same as phase 3,  $W_5$  is the beginning weight, the end is  $W_6$ . During this phase the specific fuel consumption,  $sfc$ , is assumed as 0.65 lb/lb hr, cruising distance is 3,000 nautical miles, and cruise velocity is from 0.6M down to 0.5M, the ratio of weight is 0.897.

#### **Phase 6. Descent from 50,000 ft to Sea level**

From 50,000 ft descent to sea-level. Assume the fuel consumption is about 2 to 2.5 percent of the weight.  $W_6$  is the beginning weight, the end is  $W_7$  and the ratio of weight is 0.97.

#### **Phase 7. Landing and engine shut-off**

$W_7$  is the beginning weight, the end is  $W_8$ , assuming fuel consumption is 0.005 of the weight, the ratio of weight is 0.995.

In accordance with chap. 2.4.2, eq 2.4.3, the value of the total mission ratio of weight,  $M_{ff}$ , is 0.4278. From chap. 2.4.2, eq. 2.4.4, the mission fuel weight compared with the take-off weight is 0.5722.

Reference 26 stated that the take-off weight of Tier-II ( $W_{to}$ ) is around 22,914 lb. Therefore, the mission fuel weight ( $W_f$ ) is 13,110.5 lb.

As we did in chap. 2.4.2, three percent mission fuel is added as reserve fuel, then the mission fuel weight finally becomes 13503.8 lb.

## 3.6 Weight and Balance calculation

The basic aircraft dimensions, mission profile and fuel fraction have been obtained. According to fig. 3.3.1, Tier-II Plus flowchart, following is aircraft weight calculation and aircraft C.G. estimation.

### 3.6.1 Weight estimation

Before calculating the weight, the limit load factor ( $N_z$ ) should be defined. By using the equation in appendix A, eq. A.5.1 to A.5.7, when climbing through 20,000 ft, the limit load is 2.182, when descending through 20,000 ft, the limit load is 3.225. Since during climb the aircraft weight is much higher than when descending, a limit load factor of 3 is chosen as reasonable.

Reference 10, chap. 15, suggests three categories, fighter, cargo and general aviation, available for estimating aircraft weight. For the same reason as discussed in chap. 2, in preliminary design, the General-aviation weight method was selected to perform weight estimation. The detailed calculation is showed in appendix B.1 and each component weight is listed in table 3.6.1.

| Component                | Weight, lbs | Percent total weight |
|--------------------------|-------------|----------------------|
| wing                     | 3080.162    |                      |
| tailplane                | 184.4       |                      |
| fuselage                 | 635.71      |                      |
| main landing gear        | 336.15      |                      |
| nose landing gear        | 80.292      |                      |
| a. total structure       | 4316.714    | 0.186                |
| engine install           | 2016.86     |                      |
| fuel system              | 653.6       |                      |
| b. total propulsion      | 2670.46     | 0.114                |
| hydraulic system         | 22.9        |                      |
| flight control           | 343.34      |                      |
| inst.+avion.+elect.      | 530.95      |                      |
| c. total fixed equipment | 897.19      | 0.038                |
| A. weight empty (a+b+c)  | 7884.38     | 0.337                |
| B. mission payload       | 2000        | 0.085                |
| C. fuel                  | 13503.8     | 0.577                |
| gross weight ( A+B+C )   | 23388.2     |                      |

Table 3.6.1 Component weight of Tier-II Plus

Now, from above results, the difference is 474.2 lbs (2 percent) from specification weight (22,914 lbs). Thus, the results is acceptable.

### 3.6.2 Balance calculation

After calculating the component weight, in this paragraph aircraft C.G. operating range is calculated. Before calculation, we assume the reference axis is 50 inch ahead of the aeroplane.

#### 3.6.2.1 Component C.G. Locations

As in appendix A.6.1, each component C.G. can be estimated. First the reference axis is assumed to be 50 inches ahead of aircraft nose, then by following appendix A.6.1, each component C.G. location can be calculated. The detailed calculation is listed in appendix B.2.1, and the results summarised in table 3.6.2.

| TYPE OF COMPONENT | Xi ,    inch |
|-------------------|--------------|
| Fuselage group    | 257.79       |
| Wing group        | 337.32       |
| Tailplane         | 563.32       |
| Engine group      | 413.6        |
| Nose landing gear | 180          |
| Main landing gear | 357.5        |
| Fixed equipment   | 134          |
| Fuel              | 337.32       |
| Fuel tank         | 337.32       |
| Forward payload   | 219.2        |
| Aft. payload      | 280.4        |

Table 3.6.2 Aircraft component C.G. location

#### 3.6.2.2 Aircraft C.G. movement

In paragraph 3.6.1, the aircraft component weight has been given, and from the previous paragraph, each component C.G. has been calculated. By assuming the loading scenarios are the same as in chap. 2.6.1, and using the same calculation method as in chap. 2.6.1, the aircraft C.G. location at empty weight, empty weight with payload, empty weight with fuel and take-off weight can be calculated.

The detailed calculation is in appendix B.6.1, and the results are presented in fig. 3.6.1.

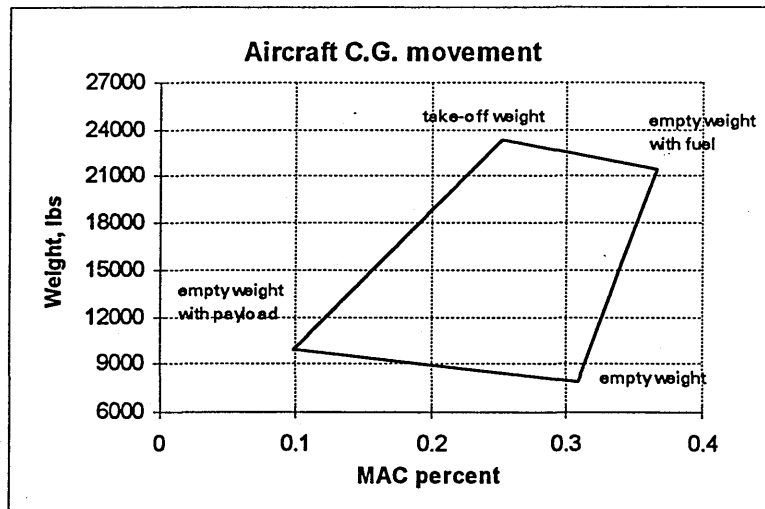


Fig. 3.6.1 Aircraft C.G. movement

From fig. 3.6.1, the aircraft C.G. movement is between 10 to 36.6 percent of wing mean aerodynamic chord. According to reference 9, table 8.16, for jet engine airplanes, the C.G. movement should around 10 to 40 percent of wing MAC (for the type of configuration considered here). The C.G. movement of Tier-II Plus is between 10 to 36.6 percent of wing MAC. Thus, the aircraft C.G. movement is within limit.

### 3.7 Wing body incidence and Oswald efficiency

As in chap. 2.7, before calculating the drag and performance of aircraft, the value of wing body incidence,  $i_w$  and Oswald efficiency,  $e$ , should be estimate.

#### 3.7.1 Wing body incidence

As in chap. 2.7.1, eq. A.7.17 may be used to calculate the wing body incidence. After calculation, two different wing body incidence at 65,000 ft cruise are obtained. First one, at the beginning of cruise, the wing body incidence is 7.948 degrees. Second, at the end of cruise, the wing body incidence is 4.1 degrees.

Reference 66 suggests taking the average value from these two extreme condition. Therefore, the wing body incidence as 6 degrees has been chosen.

### 3.7.2 Oswald efficiency

As in chap. 2.7.2, it is hard to find a proper method to estimate the value of Oswald efficiency of high aspect ratio wing.

After calculation (detailed calculation is in appendix B.3.4), 0.78 is the figure not too far from estimation. Therefore, 0.78 has been chosen as Oswald efficiency to perform conceptual design.

## 3.8 Performance Analysis

As in chap. 2.8, in this paragraph, we are trying to estimate aircraft performance.

### 3.8.1 Aircraft component drag analysis

As in chap. 2.8.1, before doing performance analysis, the aircraft total drag coefficient should be given. The total aircraft drag will vary with aircraft altitude, velocity and weight. Therefore, in this paragraph, we only chose one particular case, in the cruise phase as an example to calculate the aircraft drag coefficient.

In this example, we assume aircraft weight is 19,000 lbs, flight speed is 0.6 M, and altitude is 65,000 ft.

As in chap. 2.8.1, we only consider the drag coefficient of wing, fuselage, empennage, trim, nacelle and landing gear.

After calculation, in this example, the total airplane drag coefficient is 0.0423 (the detailed calculation is listed in appendix B.4).

In table 3.8.1 is the component drag coefficient (if the value lower than 0.0001, is not in this table).

| $C_{Dow}$ | $C_{Diw}$ | $C_{Dof}$ | $C_{Dif}$ | $C_{D0h}$ | $C_{Dih}$ | $C_{D0n}$ | $C_{Din}$ | $C_D$   |
|-----------|-----------|-----------|-----------|-----------|-----------|-----------|-----------|---------|
| 0.00832   | 0.02553   | 0.00277   | 0.0002    | 0.00236   | 0.0014    | 0.0011    | 0.00063   | 0.04232 |

Table 3.8.1 Aircraft component drag coefficient

Compared with the total drag coefficient, we put the proportion of each component drag into fig. 3.8.1.

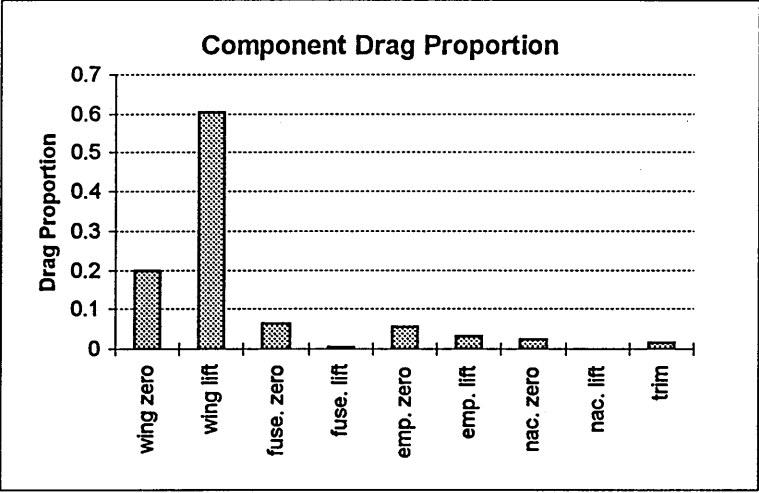


Fig. 3.8.1 Component drag proportion

From fig. 3.8.1, the wing drag coefficient, zero drag and lift drag coefficient, is equal to around 80 percent of total drag coefficient. That means, the wing drag is a dominant factor for the drag of Tier-II Plus.

3.8.2 Stall analysis

As in chap. 2.8.2, we assume the condition to calculate aircraft stall speed is at altitude 50,000 ft, weight 22,000 lbs and speed 0.6 M.

The aircraft stall speed is varied with aircraft weight and altitude. Thus, after calculation, a stall speed chart, based on different weight and altitude, is presented in fig. 3.8.2.

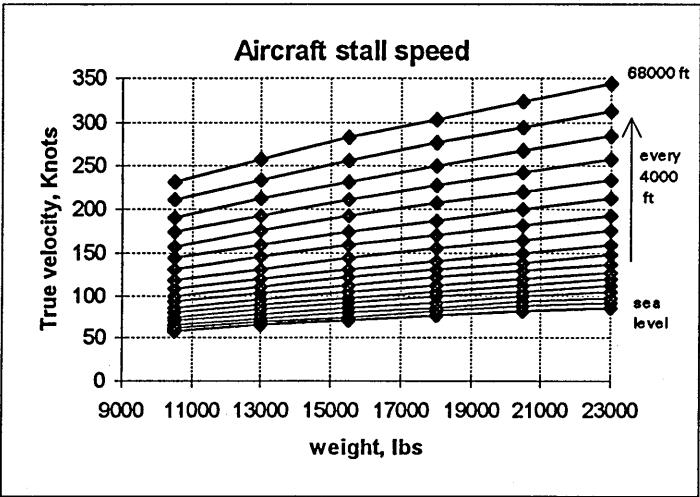


Fig. 3.8.2 Aircraft stall chart

In fig. 3.8.2, it is very easy to find that if the altitude is fixed, the stall speed is increased as the aircraft weight increases. And if the weight is fixed, the stall speed is increased as the altitude goes up. Detailed calculation is listed in appendix B.5.

### 3.8.3 Engine performance

From reference 26, it is mentioned that the propulsion system in Tier-II Plus is the Allison AE3007 two shaft subsonic turbofan engine, the net static thrust at sea level is 7150 lbf and specification fuel consumption at sea level is 0.39 lb/lb-h.

From reference 35, we only acquire the basic engine data such as engine diameter, length, bypass ratio, pressure ratio, mass flow rate, and engine weight. As for other detailed engine data, such as the specific fuel consumption and engine net thrust when flying at different altitude and different speed, this is not available.

As in chap. 2.8.3, due to lack of real engine data, we decide to chose the engine simulation program, reference 36, to generate “simulated” engine data.

With different combinations of fan and compressor pressure ratio, and different turbine inlet temperature to run the program, finally we obtain the net thrust at sea level, 7,175 lbf, which is close to Allison AE3007 turbofan engine.

According to reference 35, the gas exhausted from the nozzle of Allison AE3007 turbofan engine should be mixed. By running TURBOMATCH program, we find that if the altitude is below 50,000 ft, the engine performance of mixed exhaust gas is very good. But when the altitude above 50,000 ft, the engine specific fuel consumption will increase dramatically. That means this kind engine is good for operating at 50,000 ft or below. Alternatively, we use non-mixed nozzle to simulate Allison AE3007 turbofan engine, and obtain more believable engine data.

The engine map, from sea level to 66,000 ft is very complicated. In following figures, we only list, from 42,000 ft to 66,000 ft, the engine data map, such as specific fuel consumption versus Mach number in fig. 3.8.3 and thrust versus Mach number in fig. 3.8.4. The detailed engine simulation process is listed in appendix B.6.

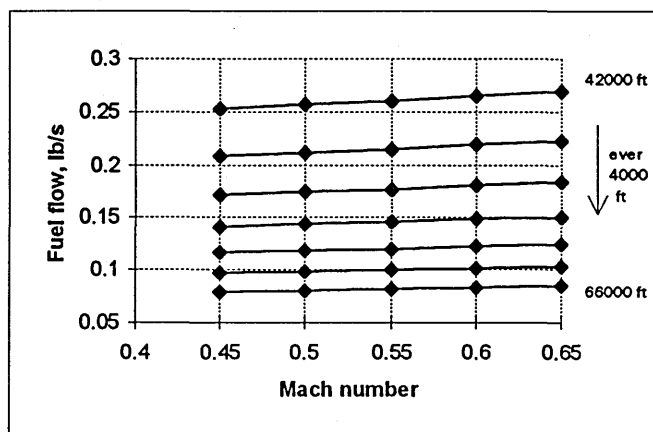


Fig. 3.8.3 Mach number VS Fuel flow

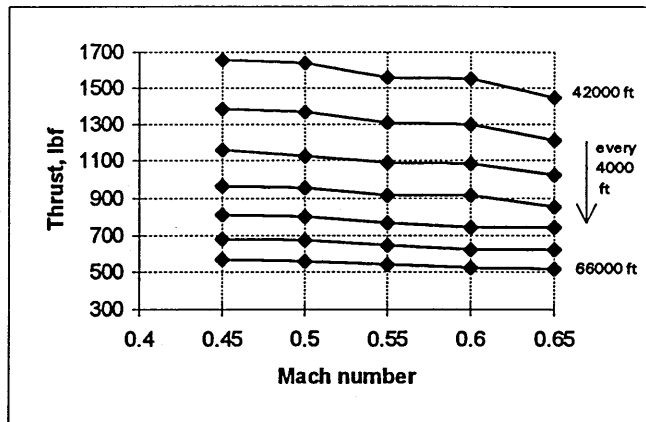


Fig. 3.8.4 Mach number VS Thrust

### 3.8.4 Climb performance

According to the Tier-II Plus spec., after take-off, it should climb to 50,000 ft.

As in chap. 2.8.4, we divide the climb segment into three parts. First part, from sea-level climb to 22,000 ft, with constant indicated airspeed 338 fps. Second part, from 22,000 to 36,000 ft, with constant indicated airspeed 270.4 fps. Third part, from 36,000 to 50,000 ft, with constant indicated airspeed 236.6 fps. If the climbing speed is higher than stall speed and minimum drag speed, then the climbing speed has been chosen correctly.

As in chap. 2.8.4, fig. 2.8.11, the stall speed and minimum drag speed are calculated respectively. In this example, the aircraft weight is assumed to be 22,000 lbs, the altitude is assumed from 2,000 to 50,000 ft, after calculation (the detailed calculation is in appendix B.7), the results are presented at fig. 3.8.5.

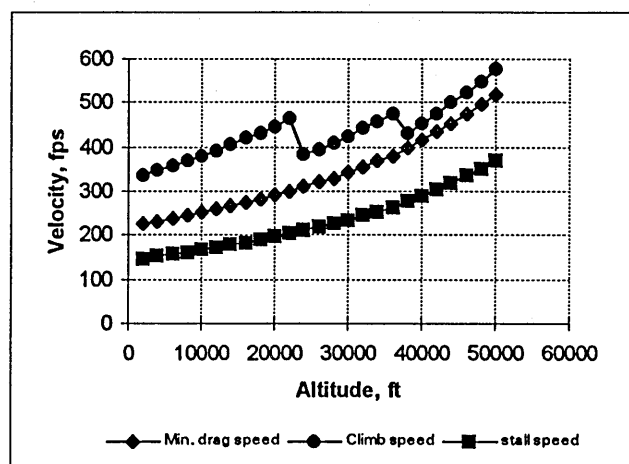


Fig. 3.8.5 Climb speed VS Stall & minimum drag speed



In fig. 3.8.5, the climb speed is higher than aircraft minimum drag velocity and stall speed. Thus, the climb speed is correct.

As in chap. 2.8.4, the engine data such as engine net thrust, fuel flow and specific fuel consumption has been given in appendix B.6. Therefore, the time, fuel, and distance to climb in each interval, every 2,000 ft, can be obtained.

After computation, the total climbing time (from sea level to 50,000 ft) is around 26 minutes, the total fuel consumes around 603.4 lbs, and the total climb distance is 119.46 nautical miles. The detailed calculation is listed in appendix B.7.1, and the results are presented at fig. 3.8.6.

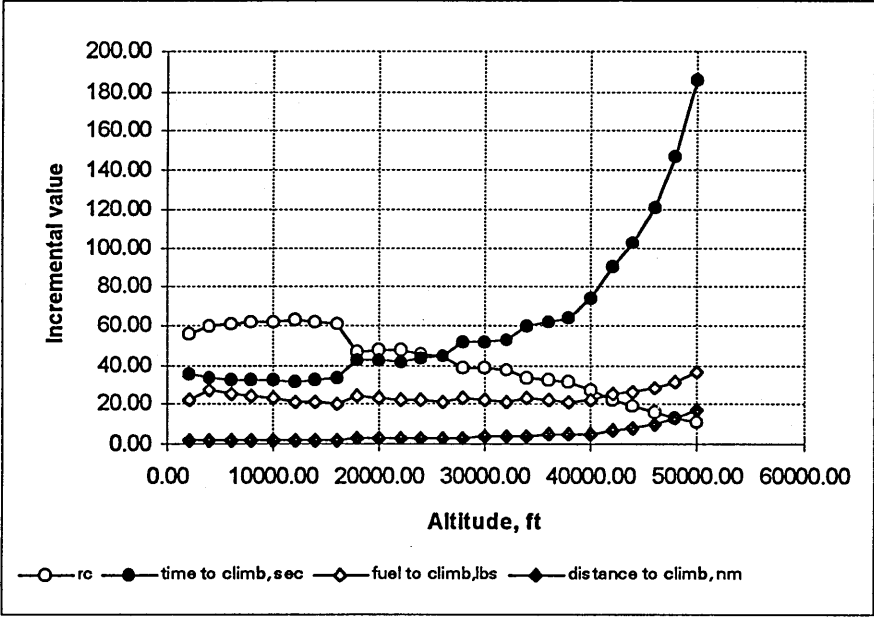


Fig. 3.8.6 Climb performance

### 3.8.5 Cruise performance

Reference 37 states that the total range of aircraft is the distance traversed in climb, cruise and descent. For most aircraft, the distance of cruise segment is the longest. Especially for Tier-II Plus, from its mission specification, in fig. 3.2.1, the total cruise time, including cruise-climb, loiter and egress segment, is more than 40 hours. Thus, for Tier-II Plus, the cruise segment is the most important.

Reference 38 states that the cruise speed of Tier-II Plus is 0.6 M at 65,000 ft loiter for 24 hours. At cruise-climb ingress, from 50,000 to 65,000 ft, and at egress from 65,000 to 50,000 ft, in each part the flight distance is 3,000 nautical miles respectively.

Therefore, if the total mission time is 42 hours, then apart from loiter time, there is 18 hours left for cruise-climb ingress and egress. Then by evenly distribute 18 hours into these two parts, there is 9 hours in each segment. Assuming ingress speed varied from 0.54 to 0.6 M, and egress speed varied from 0.6 to 0.5 M, after 9 hours, the distance is

3,099 and 2953 nautical miles respectively, which is close to the requirement, 3,000 nautical miles in each segment.

In following paragraphs, we will calculate the performance of cruise-climb ingress, on station loiter and egress consecutively.

### 3.8.5.1 Cruise-climb ingress

There are 9 hours in cruise-climb segment, and the aircraft will climb from 50,000 to 65,000 ft. As was the case in chap. 2.8.7, after 9 hours, the aircraft only ascend to 53,728 ft. This is too far from the requirement to reach to 65,000 ft.

Since ingress climb is from 50,000 ft to 65,000 ft, which the aircraft needs to climb 15,000 ft in 9 hours, thus in 1 hour the aircraft should ascend 1,666.7 ft (0.463 fps). When the rate of climb as 0.463 fps is taken into account, after climbing for 8 hours, the altitude reaches to 62,853 ft, and aircraft weight reduces to 19,257 lbs. If again climb for 1 hour, the altitude should reach around 65,000 ft. But the problem is that after 8 hours climb, the engine net thrust available is less than thrust requirement. Therefore, the aircraft cannot continue to climb.

In order that the engine net thrust can meet the thrust requirement, we decide to reduce the rate of climb from 0.463 fps to 0.37 fps. And after 9 hours climb, the altitude is 61,539 ft. There is 3,461 ft difference from 65,000 ft.

The detailed computation about cruise-climb segment is listed in appendix B.8.1, and the result is listed in fig. 3.8.7.

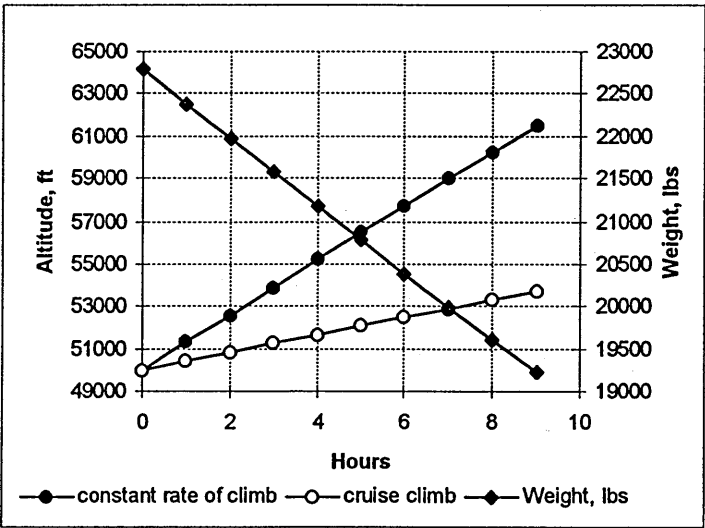


Fig. 3.8.7 Cruise-climb performance

3.8.5.2 Loiter

Reference 38 stated that Tier-II Plus loiters at 65,000 ft for 24 hours. But by the end of climbing for 9 hours, the altitude does not reach 65,000 ft. Therefore, at the beginning of loiter, the aircraft should continue climb to 65,000 ft and then loiter at 65,000 ft.

From 61,539 ft to 65,000 ft, the cruise-climb method is being used, after 9 hours the aircraft climbs to 65,091 ft. Therefore, there are 15 hours left for constant altitude loiter (the detailed calculation is in appendix B.8.2). The results of weight reduction (fuel consumption) and cruise altitude change are presented in fig. 3.8.8.

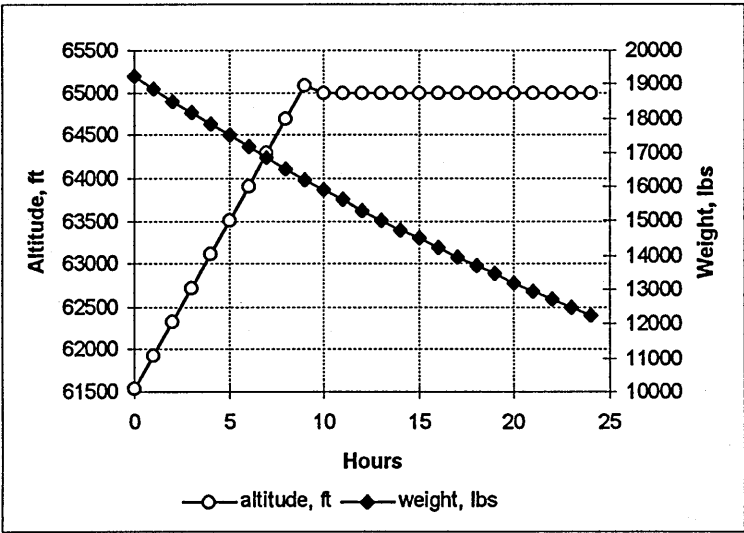


Fig. 3.8.8 Cruise performance

3.8.5.3 Egress

Reference 38 stated that egress segment is like cruise-climb segment, within 9 hours descending from 65,000 ft to 50,000 ft.

In this segment a variable speed, 0.6 to 0.5 M, and constant rate of descending, 0.45 fps, is taken to descend. By the end of egress, the aircraft weight is around 10,367 lbs. From appendix B.1, table B.1.2, the structure weight is 7,884 lbs. If plus the weight of payload 2,000 lbs, then by the end of egress segment, the aircraft weight should be no less than 9,884 lbs. From this simulation results, the aircraft still has 483 lbs mission fuel left.

The detailed computation is in appendix B.8.3, and the result is in fig. 3.8.9.

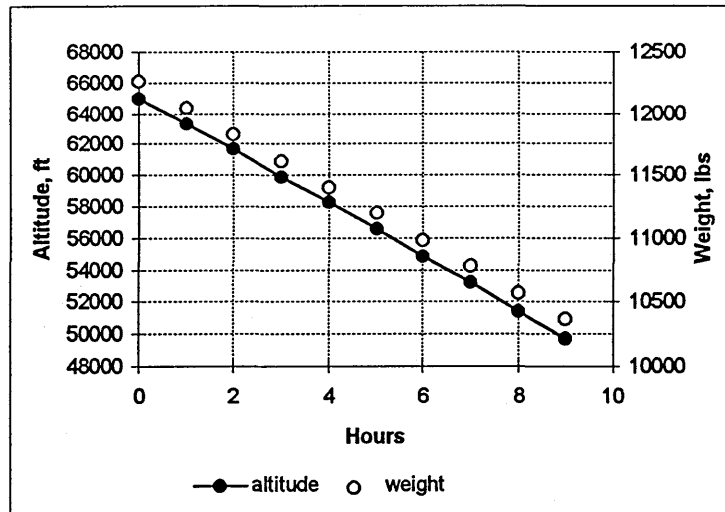


Fig. 3.8.9 Egress performance

### 3.8.6 Descend and loiter 1 hour at sea level

As in chap. 2.8.8, at the end of cruise, there is a descent from 50,000 ft to sea level. Assuming the descent time and number of segments are the same as in climb, first is from 50,000 ft to 36,000 ft, second is from 36,000 ft to 22,000 ft and last is 22,000 ft to sea level. In each segment the speed assumed as 0.5 M, 0.4 M and 0.3 M respectively.

With the same method as in chap.2.8.8, if it is assumed that the time needed for descend is the same as for climb, 26 minutes, and the number of segments are the same as in climb, then 9, 9 and 8 minutes in each segment is needed.

First at 50,000 ft and 36,000 ft, if the speed is 0.5 M, the fuel flow at full throttle is 0.142 and 0.278 lb/sec respectively. By taking the average and divided by 7, the fuel flow at idle throttle becomes 0.03 lb/sec. Therefore, if multiplied by the descending time, 9 minutes, then the fuel consumption is 16.2 lb. Second at 36,000 ft to 22,000 ft, if the speed is 0.4 M, the fuel flow at full throttle is 0.269 and 0.409 lb/sec respectively. By taking the average and divided by 7, the fuel flow at idle throttle becomes 0.048 lb/sec. Therefore, if multiplied by the descending time, 9 minutes, then the fuel consumption is 26.2 lb. Finally from 22,000 ft to sea level, if the speed is 0.3 M, the fuel flow at full throttle is 0.398 and 0.695 lb/sec respectively. By taking the average and divided by 7, the fuel flow at idle throttle becomes 0.078 lb/sec. Therefore, if multiplied by the descending time, 8 minutes, then the fuel consumption is 37.5 lb.

By adding together, the fuel consumption during descent, from 50,000 ft to sea level this gives 79.8 lb. Still there is 403 lb mission fuel left which can be used for 1 hour loiter at sea level.

Now the aircraft weight is 10,287.22 lb, and from fig. 3.8.2, the stall speed at sea level is around 60 nautical miles, 101.33 fps. If the aircraft loiter speed at sea level is taken as 0.25 M, 279.1 fps, more than twice the stall speed. Therefore, its reasonable to loiter at sea level with 0.25 M. From calculation, for the aircraft weight as 10,287 lb, the thrust required is around 612.9 lb, after 1 hour loiter at sea level the fuel consumption is around 367 lb, thus, finally the final aircraft weight becomes 9,920.22 lb, still there is around 36 lb more fuel left.

From above calculation, apparently the aircraft is capable of carrying out the mission.

### 3.8.7 Discussion

After computation, the “simulated” mission profile is listed in fig. 3.8.10, slightly different from fig. 3.2.1. The difference is shown by the dotted line, when cruise-climb from 50,000 to 65,000 ft, c to d. If we chose the rate of climb as 0.45 fps, then after 9 hours, the aircraft should reach to 65,000 ft. But, as the altitude increases, the deterioration rate of engine net thrust is greater than aircraft thrust required, finally the net thrust available is less than thrust required.

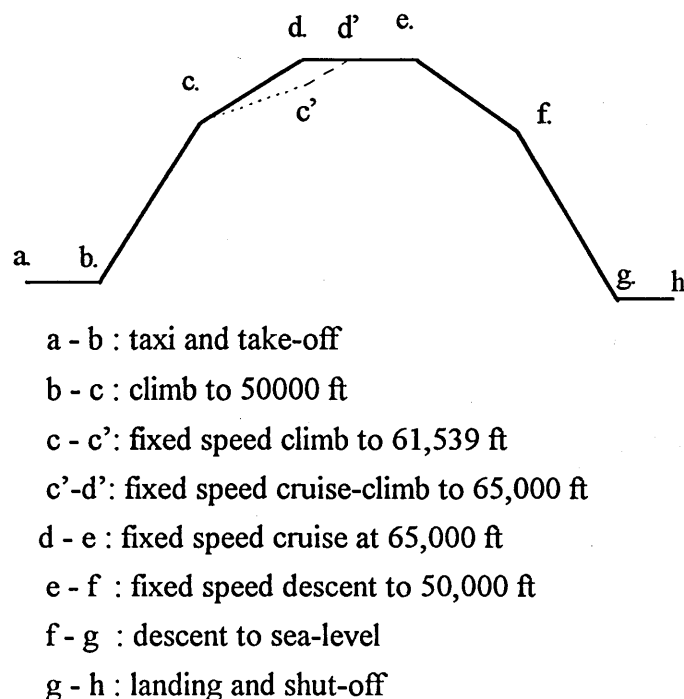


Fig. 3.8.10 Simulated mission profile

In order to make sure that the net thrust will be not less than thrust required, we reduce the rate of climb to 0.37 fps, after 9 hours climb, the altitude is less than 65,000 ft. Therefore, the cruise-climb segment becomes two parts, first climb from 50,000 to 61,539 ft, c to c', then use cruise-climb to 65,000 ft, c' to d.

The reason of the mission profile is slightly different from the mission specification is, due to the difficulty of acquiring true engine and aircraft information, instead a simulated engine data and roughly measured aircraft dimensions are used to explore aircraft performance. Therefore, some mistakes could be made due to the following;

**A. Lacking real aircraft dimension and engine data.**

If the real detailed aircraft dimensions and real engine data are available, then more accurate calculations can be performed, this could solve the deficiency. In addition, the Tier-II Plus on station loiter at high altitude is very long at 42 hours, so any minor inaccuracies in the calculation in the first place, could lead to big errors by the end.

**B. Fuel consumption**

The total fuel consumption, during the cruise segment, from c to f, is around 12,417.8 lbs. Compared with the total mission fuel 13,503.8 lbs, almost 92 percent of fuel is used for cruise. Therefore, if during cruise, the fuel consumption can be effectively reduced by only a very small amount, then 42 hours or even longer can be achieved.

**C. Aircraft drag**

As we know, the higher the lift drag ratio, the lower thrust required. Therefore, if we can reduce aircraft drag, then thrust required can be brought down. For Tier-II Plus, compared with the total aircraft drag, the wing drag is a dominant factor. The loiter segment is the most important part for Tier-II Plus, and in this segment, there are two parts, first cruise climb to 65,000 ft, second loiter at 65,000 ft. In 3.8.11, and fig 3.8.12, are the proportion of different component drag in cruise.

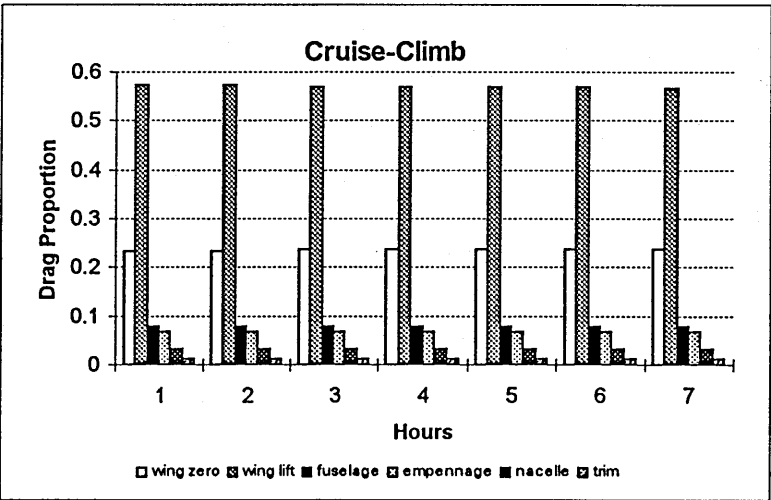


Fig. 3.8.11 Component drag proportion during cruise-climb

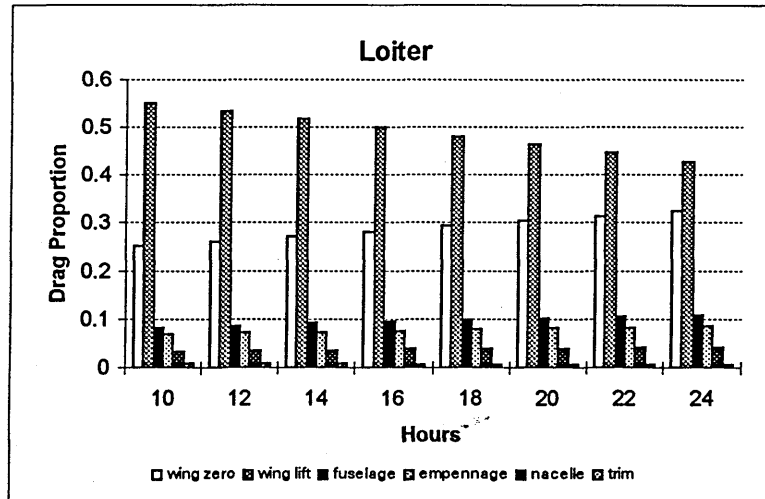


Fig. 3.8.12 Component drag proportion during cruise

From above figures, we find that either in cruise climb or in loiter segment, the wing drag proportion, zero drag plus lift drag, occupied 75 to 80 percent of the total drag. Therefore, if we can reduce wing drag, such as using laminar flow airfoil, then the wing drag can be brought down, finally the fuel consumption can be further reduced.

#### D. Oswald efficiency

The Oswald efficiency for high aspect ratio aircraft is very difficult to estimate, and its value has a high impact to aircraft performance. In this example, if its value is 0.8, then following above calculation, by the end of 42 hours cruise, the aircraft weight is around 10462 lbs, there is 95 lb fuel more than the previous calculation.

#### E. Operation

From the design point of view, if the detailed aircraft and engine data are given, the more accurate fuel consumption can be calculated, then it is not difficult to know whether the fuel is enough for 42 hours cruise.

Alternatively, as in section c, if we choose different operation style, such as at initial cruise, the aircraft is quite heavy, we can fly around aircraft minimum drag velocity. After few hours, the aircraft weight is gradually reduced, then we can fly at higher subsonic velocity. By doing so, large amount of fuel can be saved, and the long endurance of aircraft can be achieved.

### 3.8.8 Take-off performance

As in chap. 2.8.9, the take-off distance can be calculated by using appendix A, eq.

A.14.1. After calculation, the total take-off distance is 3,374.9 ft. The detailed calculation is listed in appendix B.9. The take-off chart is in fig. 3.8.13.

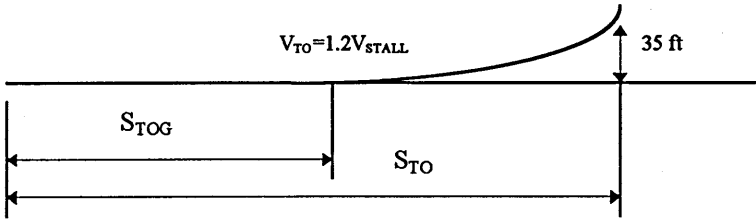


Fig. 3.8.13 Take-off performance

### 3.8.9 Landing performance

As in chap. 2.8.10, the landing distance can be calculated by using appendix A, eq. A.15.1. After calculation, the total landing distance is 1,216 ft. The detailed calculation is listed in appendix B.10. The landing chart is in fig. 3.8.14.

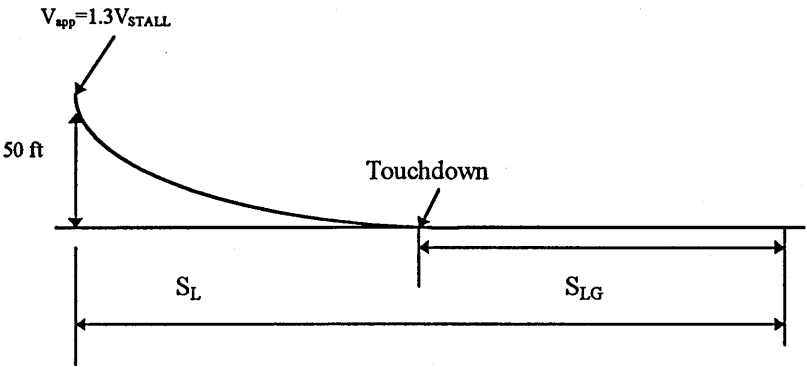


Fig. 3.8.14 Landing performance



### 3.9 Stability analysis

As in chap. 2.10, there is another equally important parameter which must be considered, safety. The aircraft needs to fly in a stable situation.

In this section, the stability of Tier-II Plus is going to be discussed.

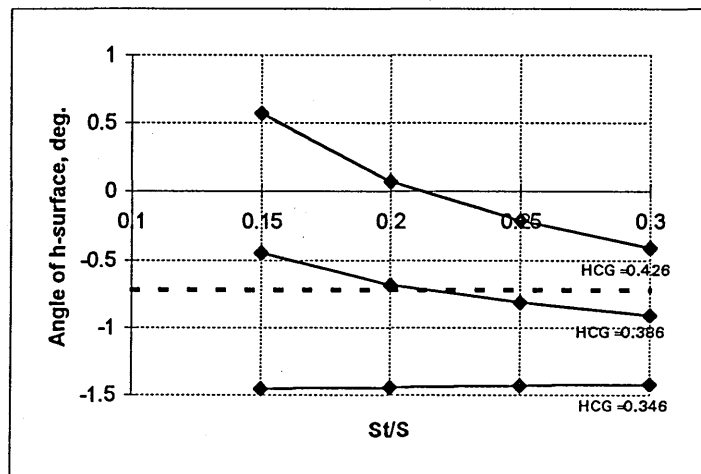
#### 3.9.1 Static stability

As in chap. 2.10.1, according to reference 45, the static stability of Tier-II Plus is estimated as follows :

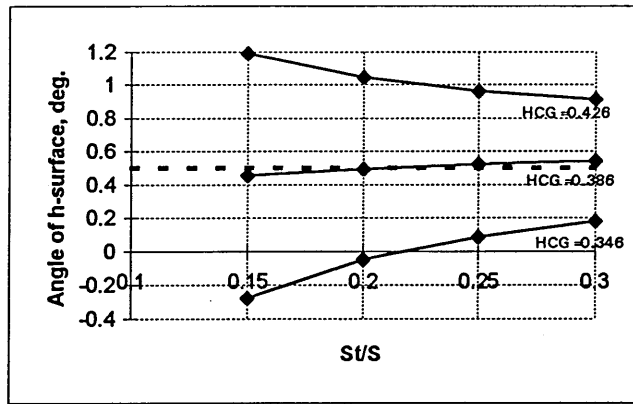
##### 3.9.1.1 Trim in cruise

As in chap. 2.10.1.1, the value of angle of incidence of horizontal surface,  $\alpha_T$ , during the cruise for trim can be calculated.

The Tier-II Plus, according to its specification, will cruise for 42 hours at the altitude ranging from 50,000 ft to 65,000 ft. After such a long period of time, the aircraft weight will have changed significantly. In order to obtain a reasonable figure of  $\alpha_T$  during cruise, both extreme condition during cruise, i.e. initial and final cruise, are chosen to calculate the incidence of horizontal surface,  $\alpha_T$ . After calculation (the detailed calculation is in appendix B.11.1), following results are obtained.



(a) Initial cruise



(b) Final cruise

Fig. 3.9.1 Incidence of horizontal surface,  $\alpha_T$

Since the range of aircraft C.G. movement is very small, the location of C.G. at the 0.386 as an average aircraft C.G. line, and the average value of  $\alpha_T$  has been taken. After calculation, the incidence of horizontal surface at initial cruise is -0.71 degrees. With the same calculation, the incidence of horizontal surface at the end of cruise as 0.51 degrees is obtained.

### 3.9.1.2 Trim on approach to land

As in chap. 2.10.1.2, after calculation, we obtain two different angle of elevator deflection,  $\eta$ , sets.

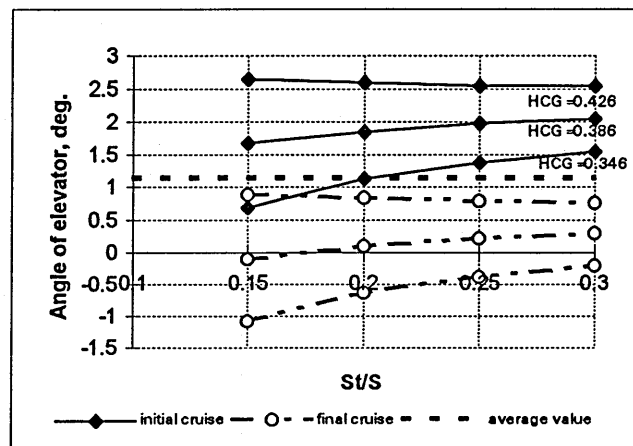
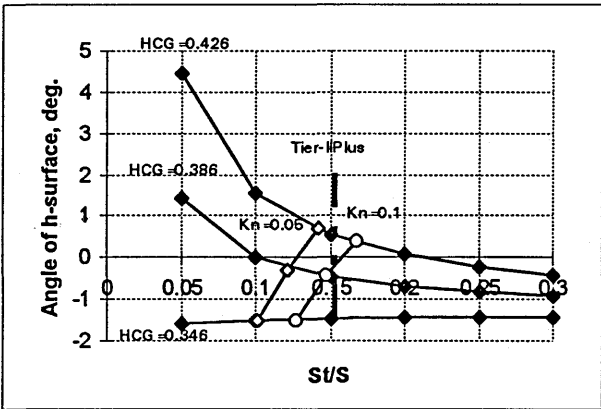


Fig. 3.9.2 Angle of elevator deflection,  $\eta$

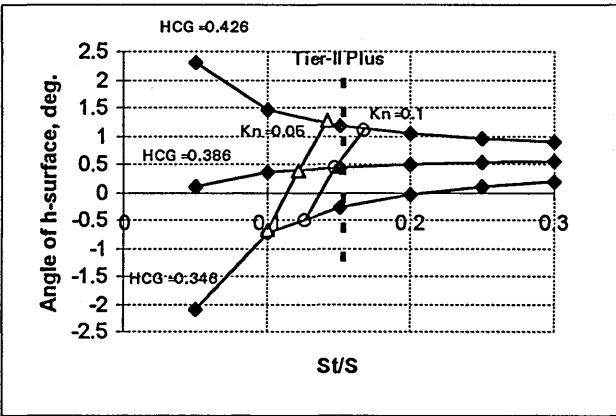
As in chap. 2.10.1.2, the trim angle of elevator deflection is not excess of 1.2 degrees. Therefore, there is no need to adjust the horizontal surface angle.

### 3.9.1.3 Cruise static stability

As in chap. 2.10.1.3, in this paragraph we will calculate the stick fixed static stability in cruise. Again we take two extreme points in cruise, initial and final cruise.



(a) Initial cruise



(b) Final curies

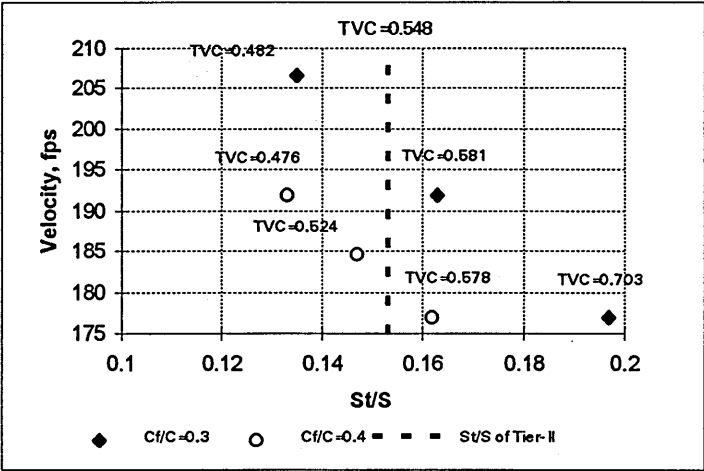
Fig. 3.9.3 Cruise static stability

The location of aircraft most aft C.G. is at line 0.386, with regard to wing MAC. Using this C.G. line and the static margin,  $K_n$ , as 0.05 or 0.1, the value of  $s_r/s$ , during cruise, is not required to exceed 0.15. For Tier-II Plus, the value of  $s_r/s$ , is 0.1535. Therefore, the cruise static stability meets the requirements.

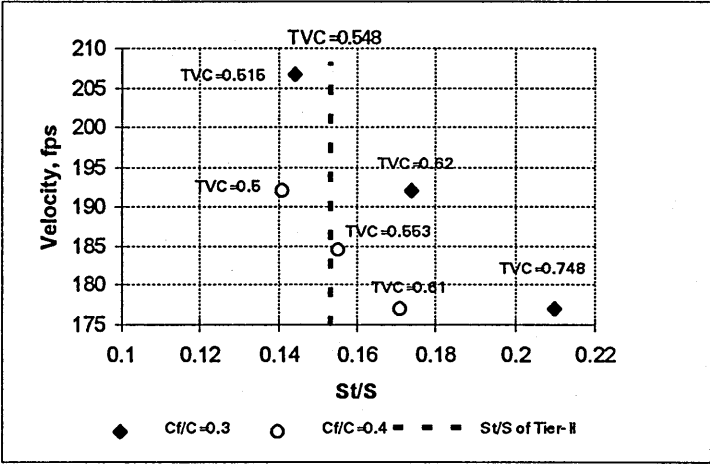
### 3.9.1.4 Rotation at take off

As in chap. 2.10.1.4, by using eq. A.18.31, the value of tail volume coefficient,  $\bar{V}$ , is 0.548.

Following calculation ensures the value of tail volume coefficient and  $s_r/s$  meets the requirement. As in chap 2.10.1.4, two different proportions of control surface chord with regard to the wing chord, 0.3 and 0.4 have been taken, and two angles, the incidence of horizontal surface as  $-0.71$ , and  $0.51$  degrees, were considered.



(a)  $\alpha_T$  is  $-0.71$  degrees



(b)  $\alpha_T$  is  $0.51$  degrees

Fig. 3.9.4 Rotation at take-off

From fig. 3.9.4 (a), (b), if the proportion of control surface chord (tailplane) to the wing chord,  $c_j/c$  is 0.3, the rotation speed is around 1.35 multiplied by the stall speed, the value of tail volume coefficient can meet the requirement as 0.548.

If the proportion of control surface chord (tailplane) to the wing chord,  $c_j/c$  is 0.4, the rotation speed is around 1.25 multiplied by the stall speed, the value of tail volume

coefficient can meet the requirement as 0.548.

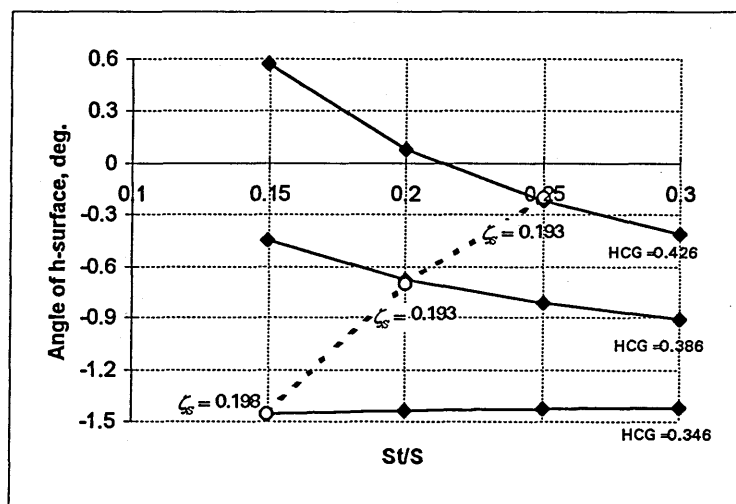
### 3.9.2 Dynamic stability

As in chap. 2.10.2, following is discussing short period mode and long period mode.

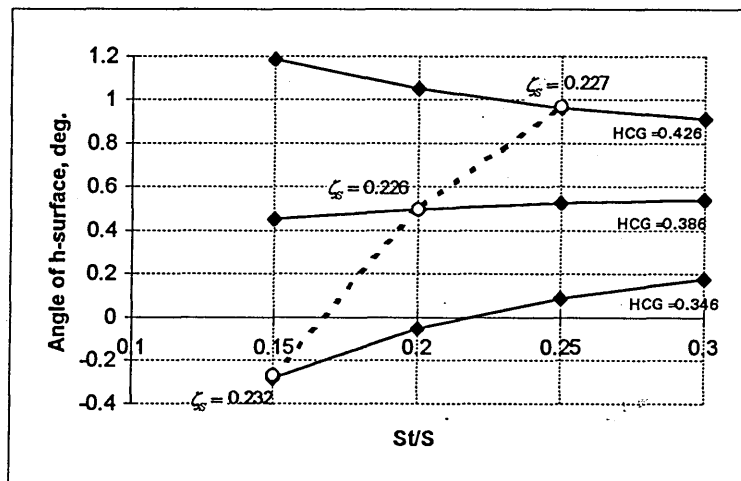
#### 3.9.2.1 Short period oscillation

In short period oscillation, there are two things which should be calculated, short period damping ratio,  $\zeta_s$  and undamped short period frequency,  $\omega_s$ .

a. Short period damping ratio,  $\zeta_s$ .



(a) Initial cruise,  $\alpha_T$  is -0.71 degrees



(b) Final cruise,  $\alpha_T$  is 0.51 degrees

Fig. 3.9.5 Short period damping ratio,  $\zeta_s$

The detailed calculation is listed in appendix B.11.5.1, the results for both initial and final cruise are listed in fig. 3.9.5 (a), (b).

b. Undamped short period frequency,  $\omega_s$

As in chap. 2.10.2.1, b., the undamped short period frequency are presented at fig. 3.9.6.

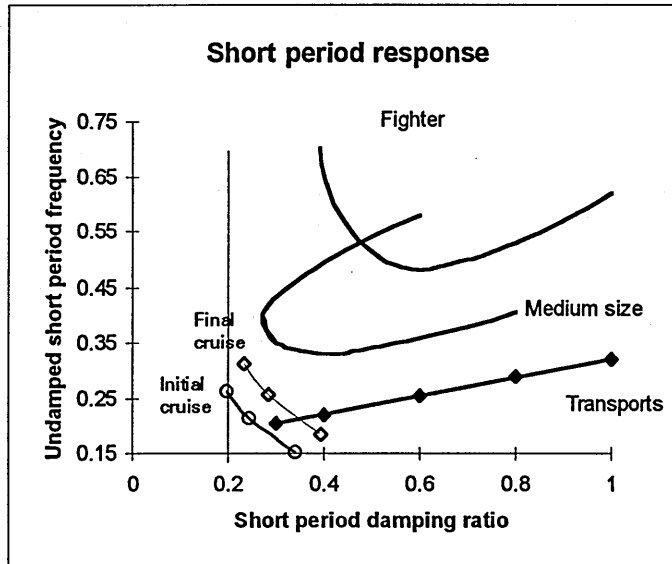


Fig. 3.9.6 Undamped short period frequency,  $\omega_s$

### 3.9.2.2 Long period mode

As in chap. 2.10.2.2, for the initial and final cruise, the damping ratio in the phugoid mode is (the detailed calculation is in appendix B.11.6);

a. Initial cruise

a.1 For aircraft C.G. is at 0.346,  $\zeta_L$  is 0.022.

a.2 For aircraft C.G. is at 0.386,  $\zeta_L$  is 0.021.

b. Final cruise

a.1 For aircraft C.G. is at 0.346,  $\zeta_L$  is 0.016.

a.2 For aircraft C.G. is at 0.386,  $\zeta_L$  is 0.0151.

The damping ratio should be around 0.2, as in chap. 2.10.2.2, a significant lower damping ratio assisted by a flight control system can provide adequate stability.

### 3.10 Application to new HALE UAV

In this section, a new design HALE, loitering at 70,000 ft, with a turbofan engine is considered.

#### 3.10.1 Mission profile of new HALE

The new HALE UAV is designed to carry 1,760 lbs (800 Kg) payload, take-off automatically, transit 4,100 nautical miles, loiter on station for three hours and then return to base. Fig. 3.10.1 and table 3.10.1. are the mission profile and the mission requirements presented .

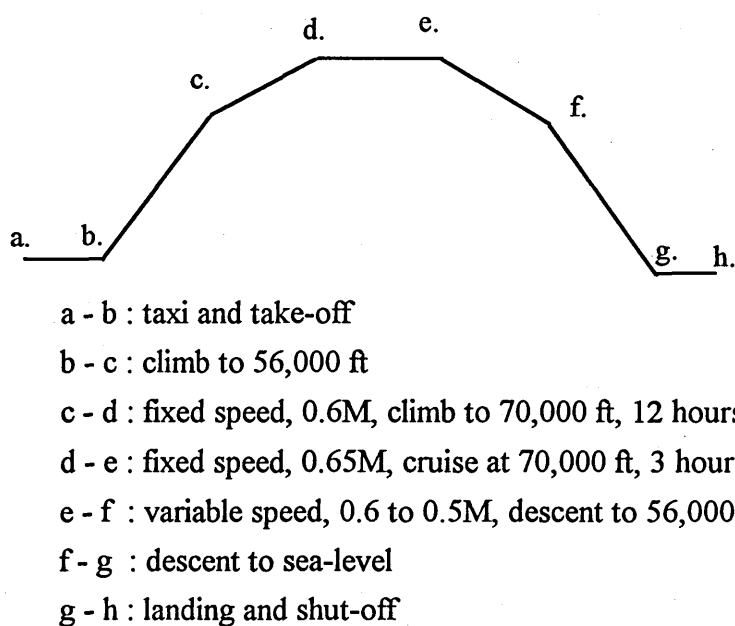


Fig. 3.10.1 New HALE UAV mission profile

| Phase                     | Description                        | Remark (fig. 3.10.1) |
|---------------------------|------------------------------------|----------------------|
| Take-off                  | From a 5,000 ft runway             | a - b                |
| Climb                     | From sea-level to 56,000 ft        | b - c                |
| Cruise : (total 27 hours) |                                    |                      |
| a. Ingress                | From 56000 ft to 70,000 ft, 0.6 M  | c - d, 12 hours      |
| b. Loiter                 | Loiter at 70,000 ft, 0.65 M        | d - e, 3 hours       |
| c. Egress                 | From 70,000 ft to 56,000 ft, 0.6 M | e - f, 12 hours      |
| Descend                   | From 56,000 ft to sea-level        | f - g                |

Table 3.10.1 New HALE UAV mission and requirements

### 3.10.2 Flowchart of new HALE UAV

Since the new HALE UAV is an application of using the program which developed in chap. 2, therefore, the flowchart is the same as in chap. 2, fig. 2.3.

### 3.10.3 Parametric study results

As in chap. 2.4, the results of parametric study are listed in fig. 3.10.2.

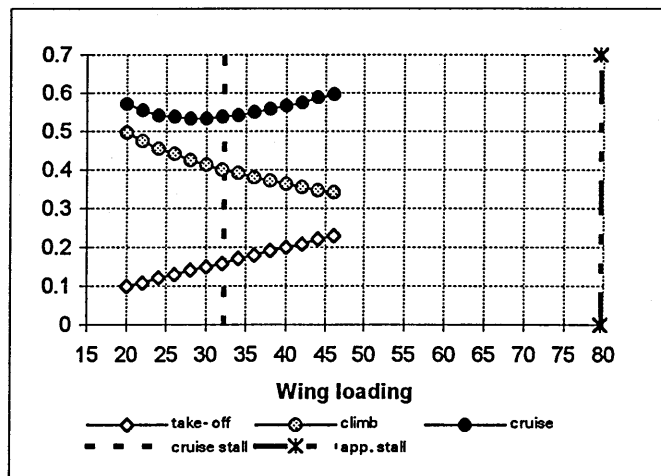


Fig. 3.10.2 Paremetric study results

From paremetric study, aspect ratio as 18, wing loading as 32, and thrust weight ratio as 0.538 are obtained.

### 3.10.4 Aircraft dimension and three-view diagram

After calculation, aircraft basic dimensions are listed in table 3.10.2, and the New HALE UAV three view diagram is in fig. 3.10.3.

| Wing                |                     |
|---------------------|---------------------|
| Wing span           | 82.16 ft            |
| Gross area          | 375 ft <sup>2</sup> |
| Aspect ratio        | 18                  |
| Wing loading (T.O.) | 32                  |
| Mean aero. chord    | 4.73 ft             |
| Root chord          | 6.09 ft             |
| Tip chord           | 3.045 ft            |
| Taper ratio         | 0.5                 |
| Root thickness      | 17 % (assumption)   |
| Tip thickness       | 13 % (assumption)   |



|                                     |                                                |
|-------------------------------------|------------------------------------------------|
| Leading edge angle                  | 4 degrees (backward)                           |
| Trailing edge angle                 | 0.24 degrees (forward)                         |
| Fuselage                            |                                                |
| Length                              | 35.72 ft                                       |
| Diameter (Max.)                     | 3.5 ft                                         |
| Tailplane (V - tail)                |                                                |
| Tail-span (horizontal,projected)    | 24.896 ft (16 ft)                              |
| Tail area (horizontal,projected)    | 105.6 ft <sup>2</sup> (67.87 ft <sup>2</sup> ) |
| Aspect ratio (horizontal,projected) | 5.869 (3.79)                                   |
| Tail mean aero. chord               | 4.4 ft                                         |
| Tilt angle ( vertical )             | 40 degrees                                     |
| Leading edge angle                  | 4 degrees (backward)                           |
| Trailing edge angle                 | 8.9 degrees (forward)                          |
| Root chord                          | 5.655 ft                                       |
| Tip chord                           | 2.8275 ft                                      |
| Taper ratio                         | 0.5                                            |
| Thickness                           | 12 %                                           |

Table 3.10.2 Dimensions of new HALE UAV

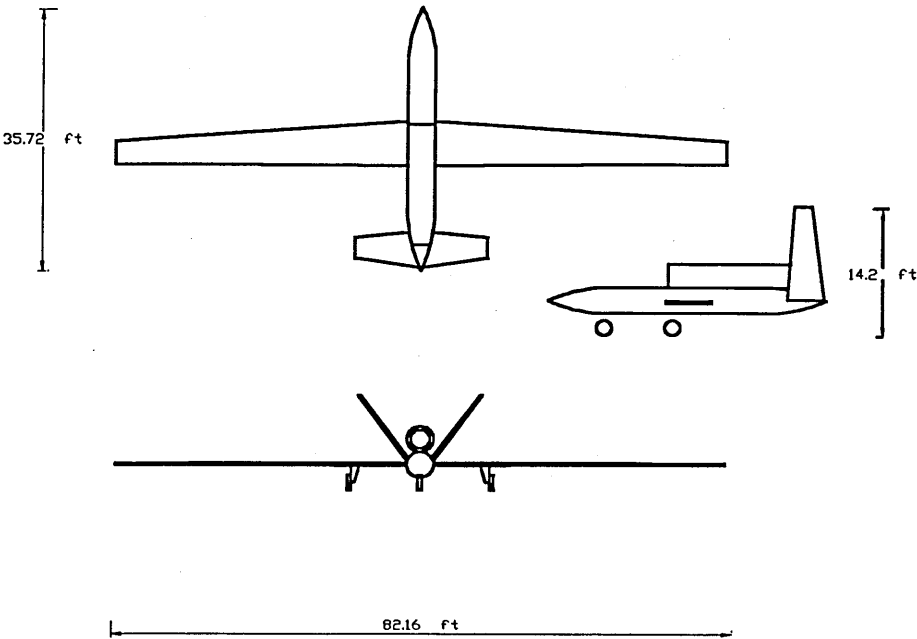


Fig. 3.10.3 New HALE UAV three-view diagram

### 3.10.5 Fuel fraction analysis

After calculation, the total mission ratio of weight is 0.562 , the results are presented in table 3.10.3.

| Phase | Description                             | ratio of weight |
|-------|-----------------------------------------|-----------------|
| 1.    | Engine start and take-off               | 0.975           |
| 2.    | Climb and accelerate to 56,000 ft       | 0.965           |
| 3.    | Fixed speed cruise-climb to 70,000 ft   | 0.814           |
| 4.    | Fixed speed loiter at 70,000 ft         | 0.924           |
| 5.    | Fixed speed cruise-descent to 56,000 ft | 0.814           |
| 6.    | Descent from 56,000 ft to Sea level     | 0.98            |
| 7.    | Landing and engine shut-off             | 0.995           |

Table 3.10.3 Ratio of weight in each phase

### 3.10.6 Weight and Balance calculation

The aircraft weight calculation and aircraft C.G. estimation are listed in table 3.10.4 and fig. 3.10.4 respectively.

| Component              | Weight, lbs | Percent total weight |
|------------------------|-------------|----------------------|
| wing                   | 1358.6      |                      |
| tailplane              | 103.37      |                      |
| fuselage               | 277.6       |                      |
| main landing gear      | 177.89      |                      |
| nose landing gear      | 54.236      |                      |
| engine install         | 2016.86     |                      |
| fuel system            | 309.        |                      |
| hydraulic system       | 12.         |                      |
| flight control         | 128.84      |                      |
| inst.+avion.+elect.    | 475.45      |                      |
| A. Weight empty        | 4913.85     | 0.407                |
| B. Mission payload     | 1760.       | 0.146                |
| C. Fuel                | 5411.26     | 0.448                |
| Gross weight ( A+B+C ) | 12085.11    |                      |

Table 3.10.4 Component weight of new HALE UAV

For new HALE, the thrust weight ratio (engine install weight divided by the initial aircraft weight) is 0.167, this figure compared with table 3.6.1, Tier-II Plus, 0.087 is obviously higher. The reason is that the requirement for new HALE UAV is to loiter at 70,000 ft, or even at higher altitude. This altitude is higher than the loiter altitude of Tier-II Plus, 65,000 ft (in our simulation, the new HALE UAV could reach the altitude of 75,000 ft). As stated in chap. 2.8.3, when the altitude increases, the engine net thrust will deteriorate. Therefore, if we want an engine can generate enough net thrust for aircraft to fly at high altitude, higher than 65,000 ft, the equivalent engine net thrust at sea level has to be increased substantially, and in general, the engine weight is proportion to engine net thrust at sea level.

Though even the aircraft gross take-off weight of new HALE UAV is almost the half gross take-off weight of Tier-II Plus, the engine thrust required of new HALE UAV at sea-level is nearly the same as the engine thrust required for Tier-II Plus. Finally, it leads to a big difference of the thrust weight ratio in between new HALE UAV and Tier-II Plus.

As in chap. 2.6.1, the aircraft C.G. movement is estimated and the results are presented in table 3.10.5 and fig 3.10.4.

| Component         | Weight, lbs | C.G location |          |
|-------------------|-------------|--------------|----------|
| wing              | 1358.6      | 261.92       | 355844.5 |
| tailplane         | 103.37      | 432.898      | 44748.67 |
| fuselage          | 277.6       | 204.315      | 56717.84 |
| main landing gear | 177.89      | 221.46       | 39395.52 |
| nose landing gear | 54.236      | 135.73       | 7361.452 |
| engine install    | 2016.86     | 304.16       | 613448.1 |
| fuel system       | 309         | 261.92       | 80933.28 |
| fix equipment     | 616.29      | 135.73       | 83649.04 |
| forward payload   | 760         | 157.163      | 119443.9 |
| aft. payload      | 1000        | 252.326      | 252326   |
| Fuel              | 5411.26     | 261.92       | 1417317  |
| Gross weight      | 12085.11    |              | 3071186  |

Table 3.10.5 Aircraft component C.G. locations

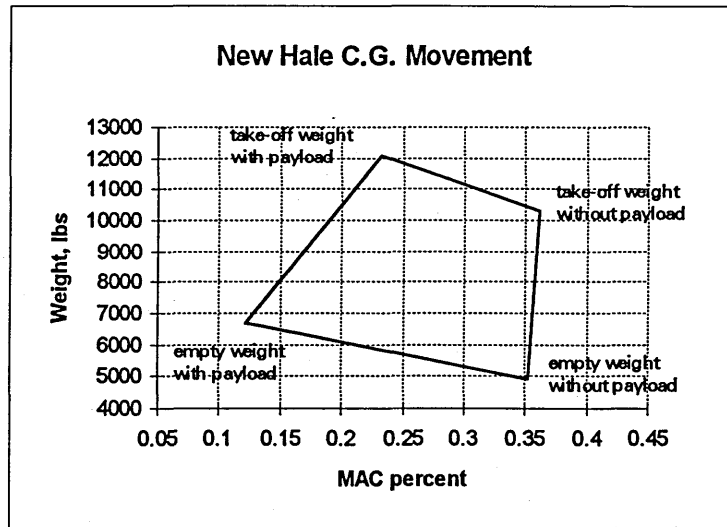


Fig. 3.10.4 Aircraft C.G. movement

From fig. 3.10.4, the aircraft C.G. movements is between 12 to 36 percent of wing mean aerodynamic chord. According to chap. 2.6.1, the C.G. movement should within 10 to 40 percent of wing C.G.. Therefore, the aircraft C.G. movement meets the requirement.

As in chap 2.6.2, by using the same method, the location of nose landing gear and main landing gear has been recalculated, after calculation, the location of nose landing gear and main landing gear has moved from the previous 135.73 and 221.46 inches to 175.73 and 261.46 inches.

### 3.10.7 Performance Analysis

As in chap. 2.7, first the lift coefficient assumed to be 1, aircraft weight, altitude and velocity assumed to be 12,085 lbs, 60,000 ft and 0.6M respectively. After calculation the value of the wing body incidence and lift drag ratio are 6.39 degrees and 26.45. However, the real value of wing body incidence, will be recalculated later.

As in chap. 2.8, in this section, the aircraft performance is estimated.

#### 3.10.7.1 Aircraft component drag analysis

In this example, the aircraft weight is 9,082.23 lbs, flight speed is 0.65 M, and altitude is 70,000 ft. After calculation, the total airplane drag coefficient is 0.03429, and the component drag coefficient is in table 3.10.6 ( the value lower than 0.0001, not listed).

| $C_{D0w}$ | $C_{diw}$ | $C_{D0f}$ | $C_{D0h}$ | $C_{Dih}$ | $C_{D0n}$ | $C_{Dlt}$ | $C_D$   |
|-----------|-----------|-----------|-----------|-----------|-----------|-----------|---------|
| 0.00872   | 0.01858   | 0.00237   | 0.00269   | 0.00069   | 0.00092   | 0.00023   | 0.03429 |

Table 3.10.6 Aircraft component drag coefficient

Compared with the total drag coefficient, the proportion of each component drag is presented in fig. 3.10.5.

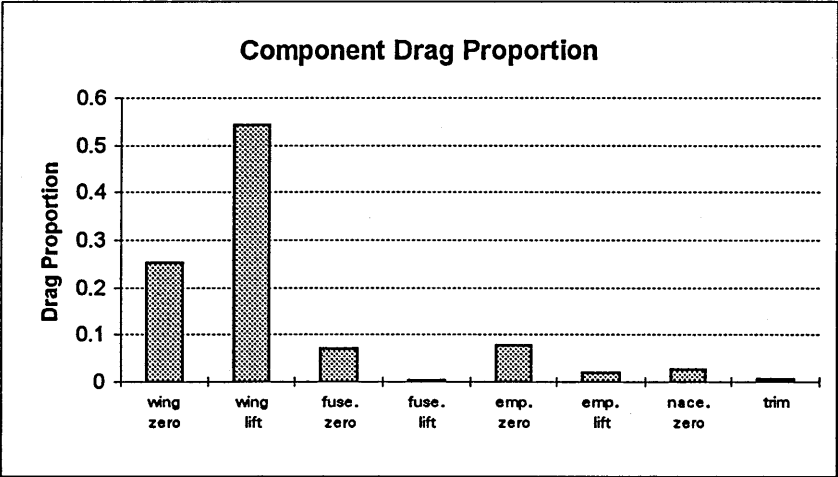


Fig 3.10.5 Component drag proportion

From fig. 3.10.5, the wing drag coefficient (wing zero drag plus wing lift drag coefficient), is nearly equal to 80 percent of total drag coefficient. Apparently, wing drag coefficient is a dominant factor regarding to the total drag coefficient.

3.10.7.2 Stall analysis

As in chap. 2.8.2, we assume the condition to calculate aircraft stall speed is at altitude 60,000 ft, weight is from 13,000 to 7,000 lbs.

After calculation, a stall speed chart, based on different weight and altitude, is presented in fig. 3.10.6.

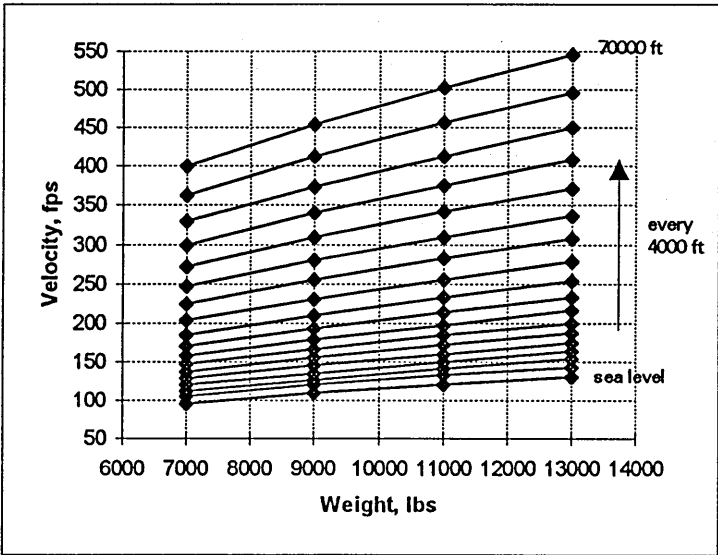


Fig. 3.10.6 Aircraft stall chart

### 3.10.7.3 Engine performance

Using the same engine as Tier-II Plus, and simulate the engine, with TURBOMATCH program, up to 76,000 ft. After simulation, the required engine data, such as specific fuel consumption, engine net thrust, fuel flow, are obtained. In fig. 3.10.7 and 3.10.8, are the results of fuel flow versus Mach number and net thrust versus Mach number with altitude respectively (in order to simplify the engine data map, the engine data listed is from 40,000 ft up to 76,000 ft).

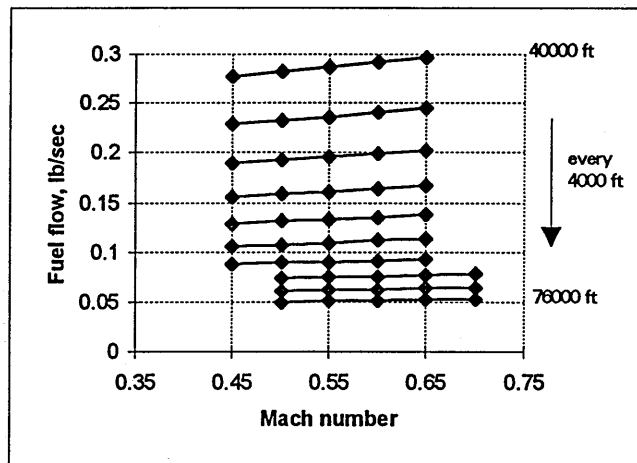


Fig. 3.10.7 Mach number VS Fuel flow

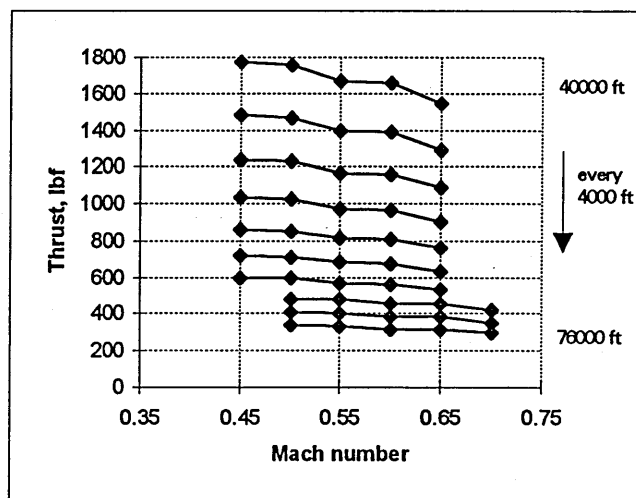


Fig. 3.10.8 Mach number VS thrust

### 3.10.7.4 Climb performance

As in chap. 2.8.4, we divide the climb segment into four parts. First part, from sea-level climb to 22,000 ft, with constant indicated airspeed 338 fps. Second part, from 22,000 to 36,000 ft, with constant indicated airspeed 270.4 fps. Third part, from

36,000 to 46,000 ft, with constant indicated airspeed 236.6 fps. Fourth part, from 46,000 to 56,000 ft, with constant indicated airspeed 202.8 fps.

After computation, the total climbing time (from sea level to 56,000 ft) is around 16 minutes, the total fuel consumed around 306.4 lbs, and the total climb distance is 78.44 nautical miles (results are listed in table 3.10.7).

| Altitude, ft | Time to climb, sec | Fuel used, lbs | Dist. of climb, nm | Altitude, ft | Time to climb, sec | Fuel used, lbs | Dist. of climb, nm |
|--------------|--------------------|----------------|--------------------|--------------|--------------------|----------------|--------------------|
| 2000.1       | 16.11              | 13.04          | 0.92               | 30001.5      | 24.24              | 9.71           | 1.76               |
| 4000.2       | 15.86              | 12.30          | 0.94               | 32001.6      | 26.86              | 10.25          | 2.03               |
| 6000.3       | 15.60              | 11.59          | 0.95               | 34001.7      | 27.40              | 9.89           | 2.15               |
| 8000.4       | 15.47              | 11.01          | 0.97               | 36001.8      | 27.96              | 9.54           | 2.28               |
| 10000.5      | 15.34              | 10.44          | 0.99               | 38001.9      | 32.11              | 9.80           | 2.40               |
| 12000.6      | 15.50              | 10.04          | 1.03               | 40002.0      | 37.60              | 10.58          | 2.95               |
| 14000.7      | 15.71              | 9.67           | 1.08               | 42002.0      | 40.92              | 10.55          | 3.37               |
| 16000.8      | 19.74              | 11.52          | 1.41               | 44002.1      | 45.77              | 10.80          | 3.95               |
| 18000.9      | 19.47              | 10.85          | 1.43               | 46002.2      | 51.60              | 11.15          | 4.67               |
| 20001.0      | 19.21              | 10.22          | 1.46               | 48002.3      | 58.45              | 11.28          | 4.76               |
| 22001.1      | 20.93              | 10.36          | 1.32               | 50002.4      | 66.41              | 11.73          | 5.68               |
| 24001.2      | 21.20              | 9.92           | 1.38               | 52002.5      | 75.96              | 12.29          | 6.81               |
| 26001.3      | 23.79              | 10.63          | 1.61               | 54002.6      | 88.74              | 13.15          | 8.35               |
| 28001.4      | 24.01              | 10.17          | 1.68               | 56002.7      | 102.24             | 13.93          | 10.10              |

Table 3.10.7 Climb performance

After climbing to 56,000 ft, the aircraft weight is 11,778.69 lbs, lift coefficient is 0.69, and the lift drag ratio is 27.48. The difference of these two lift drag ratios (one in section 3.10.7 is 26.45) is 1.03. From fig. 2.3, the difference is more than 0.5, thus we have to do the wing body incidence angle modification.

Now taking the aircraft weight as 11,778.69 lbs, the altitude as 56,000 ft, and the lift coefficient as 0.69, after calculation, the wing body incidence angle is 3.8 degrees, and the lift drag ratio is 27.5. The difference of these two lift drag ratio is 0.02, thus the value of 3.8 degrees has been taken as wing body incidence angle.

### 3.10.7.5 Cruise performance

There are three segments during cruise; cruise-climb ingress, egress and on station loiter. At cruise-climb ingress segment, the aircraft will cruise-climb from 56,000 to 70,000 ft, and at egress segment, it cruise-descend from 70,000 to 56,000 ft, in each

segment the flight distance is assumed around 4,100 nautical miles. On-station loiter, at 70,000 ft, is 3 hours.

### 3.10.7.5.1 Cruise-climb ingress

In order to let aircraft climb from 56,000 to 70,000 ft within 12 hours, with constant speed 0.6 M, the rate of climb should be no less than 1,167 ft per hour, and it is equal to 0.324 fps. By using the same method listed in appendix A.12.1, after 12 hours flight, the aircraft altitude is around 69,400 ft. The results is listed in table 3.10.8 and fig. 3.10.9.

| Hours | Weight, lbs | Height, ft | L/D   | Rate of climb, fps | Dist. of climb, nm | Climb speed, fps |
|-------|-------------|------------|-------|--------------------|--------------------|------------------|
| 1     | 11534.59    | 57218.07   | 27.51 | 0.34               | 343.92             | 580.8            |
| 2     | 11291.98    | 58438.36   | 27.43 | 0.34               | 687.84             | 580.8            |
| 3     | 11055.11    | 59645.32   | 27.32 | 0.34               | 1031.76            | 580.8            |
| 4     | 10825.47    | 60841.98   | 27.17 | 0.33               | 1375.68            | 580.8            |
| 5     | 10600.81    | 61929.77   | 26.98 | 0.3                | 1719.6             | 580.8            |
| 6     | 10379.53    | 63081.16   | 26.8  | 0.32               | 2063.52            | 580.8            |
| 7     | 10161.11    | 64255.59   | 26.57 | 0.33               | 2407.44            | 580.8            |
| 8     | 9945.41     | 65407.31   | 26.31 | 0.32               | 2751.36            | 580.8            |
| 9     | 9732.38     | 66546.18   | 26.03 | 0.32               | 3095.28            | 581.2            |
| 10    | 9518.39     | 67765.82   | 25.74 | 0.34               | 3439.42            | 581.7            |
| 11    | 9298.57     | 68888.87   | 25.4  | 0.31               | 3783.86            | 582.2            |
| 12    | 9082.23     | 69985.08   | 25.09 | 0.3                | 4128.57            | 582.6            |

Table 3.10.8 cruise-climb figures

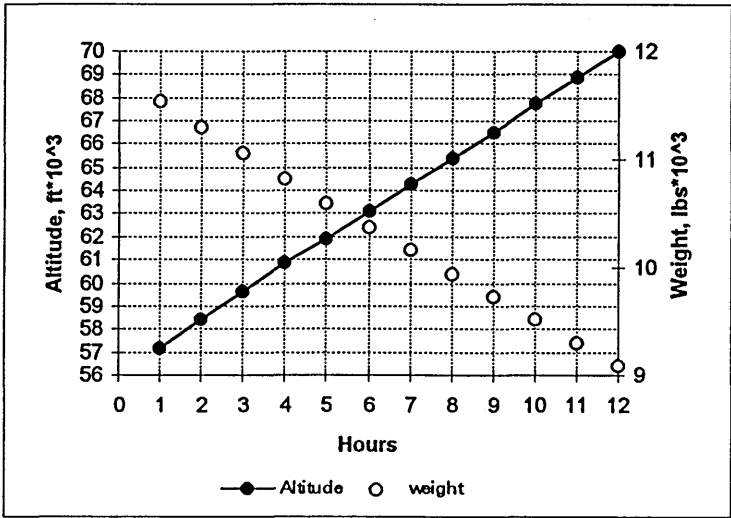


Fig. 3.10.9 Cruise-climb performance



**3.10.7.5.2 Loiter**

With regard to the specification, the new HALE UAV loiters at 70,000 ft for 3 hours, and the flight speed is 0.65 M. After calculation, the aircraft weight reduces from 9,082.23 to 8,444.514 lbs, consumes around 637.7 lbs fuel (results are presented in table 3.10.9).

| Hours | Weight, lbs | Height, ft |
|-------|-------------|------------|
| 1     | 8863.196    | 70000      |
| 2     | 8650.757    | 70000      |
| 3     | 8444.514    | 70000      |

Table 3.10.9 Loiter figures

**3.10.7.5.3 Egress**

As in section 3.10.6.5.1, the egress segment is from 70,000 ft descend to 56,000 ft. From the specification in this segment the total flight time is 12 hours, and the descend altitude is 14,000 ft, therefore, it is equivalent to 0.324 fps descend rate. After calculation the results are presented in table 3.10.10 and fig 3.10.10.

| Hours | Weight, lbs | Height, ft | L/D   | Rate of descend fps | Dist. of discend, nm | Climb speed, fps |
|-------|-------------|------------|-------|---------------------|----------------------|------------------|
| 1     | 8260.58     | 68811.5    | 25.3  | -0.33               | 339.92               | 574.09           |
| 2     | 8083.43     | 67568.95   | 25.7  | -0.35               | 674.79               | 565.55           |
| 3     | 7915.53     | 66317.11   | 26.04 | -0.35               | 1004.6               | 557.01           |
| 4     | 7757.65     | 65155.76   | 26.32 | -0.32               | 1329.49              | 548.7            |
| 5     | 7605.1      | 64002.05   | 26.51 | -0.32               | 1649.62              | 540.67           |
| 6     | 7457.25     | 62860.77   | 26.63 | -0.32               | 1964.99              | 532.63           |
| 7     | 7314.41     | 61596.82   | 26.7  | -0.35               | 2275.61              | 524.6            |
| 8     | 7175.91     | 60273.39   | 26.71 | -0.37               | 2581.47              | 516.56           |
| 9     | 7040.6      | 59174.48   | 26.64 | -0.31               | 2882.57              | 508.53           |
| 10    | 6908.96     | 57929.42   | 26.55 | -0.35               | 3178.92              | 500.49           |
| 11    | 6779.87     | 56849.06   | 26.38 | -0.3                | 3470.5               | 492.46           |
| 12    | 6653.81     | 55636.78   | 26.2  | -0.34               | 3757.33              | 484.42           |

Table 3.10.10 Egress figures

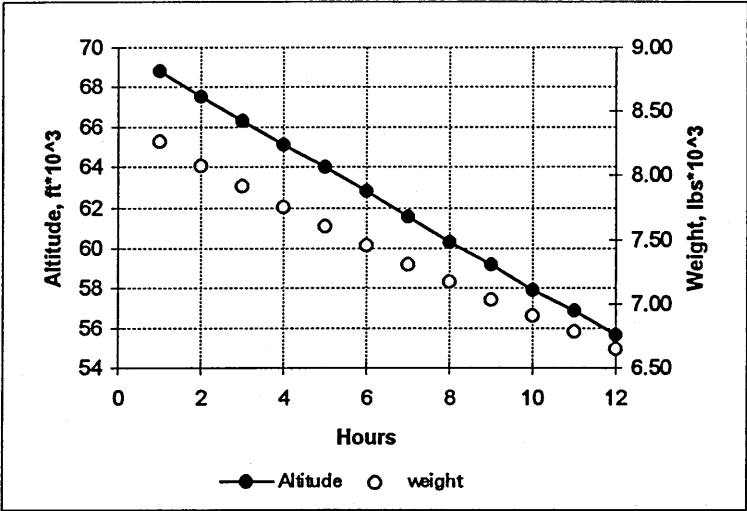


Fig. 3.10.10 Egress performance

### 3.10.7.6 Descend to sea level

As in chap. 2.8.8, at the end of cruise, there is a descent from 56,000 ft to sea level. Assuming the descent time and number of segments are the same as in climb, first, from 55636 to 46,000 ft, with constant airspeed 0.5 M. Second, from 46,000 to 36,000 ft, with constant airspeed 0.4 M. Third, from 36,000 to 22,000, with constant airspeed 0.3 M. Fourth, from 22,000 ft to sea-level with constant airspeed 0.25 M. With the same method as in chap.2.8.8, after calculation, the total fuel consumption during descent is 50.91 lbs, and the aircraft final weight becomes 6602.9 lbs.

Since the aircraft empty weight is 4913.85 lbs and the payload is 1760 lbs, the total is 6673.85 lbs. Therefore, compared with the aircraft final weight there is a difference about 70.95 lbs more fuel needed. When compared with the requirement of 10 lbs tolerance, apparently, we need to go back to re-calcul;ate.

After three iterations, finally the take-off aircraft weight from 12,085.11 lbs increased to 12,193.09 lbs and meets our requirement.

The take-off weight and the additional fuel weight required, during each iteration, are presented in fig. 3.10.11 and fig. 3.10.12 respectively. In fig. 3.10.11, the number 1,2,3 at x-axis means before the aircraft final weight meets the requirement of 10 lbs tolerance, there are 3 iterations through calculation. In fig. 3.10.12, the number 1,2,3 correspond to the number in fig. 3.10.11. Fig 3.10.12 shows the additional amount of fuel that would be needed to complete the mission over that placed on board at the start and this amount of extra fuel is placed on board for the next iteration.

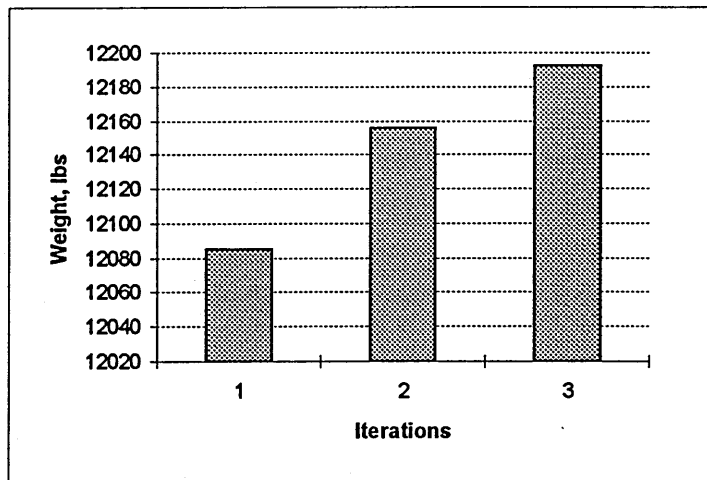


Fig. 3.10.11 Weight difference in each iteration

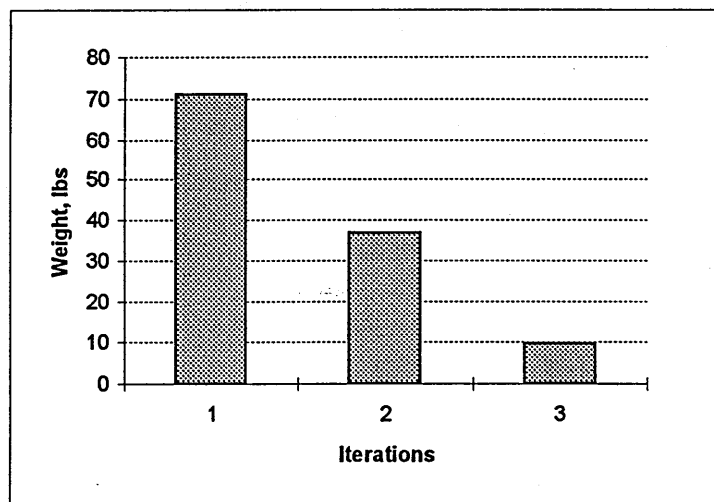


Fig. 3.10.12 Additional fuel required in each iteration

### 3.10.7.7 Take-off performance

As in chap. 2.8.9, the take-off distance may be calculated. After calculation, the aircraft take-off ground distance,  $S_{TOG}$ , is 715.4 ft, and the total take-off distance,  $S_{TO}$ , is 1,154 ft.

### 3.10.7.8 Landing performance

As in chap. 2.8.10, the landing distance may be calculated. After calculation, the landing ground run distance is 499 ft, and the total landing distance is 1,220 ft.

### 3.10.7.9 Sensitivity analysis

As in chap. 2.9, this section considers if the aspect ratio or the wing loading changes, what is the effect on the aircraft take-off weight.

### 3.10.7.9.1 Wing loading changes

Initially keeping the aspect ratio the same and changing the wing loading range from 32 to 42, the effect on the design is explored.

After calculation, the wing loading to aircraft take-off weight relationship is shown in fig. 3.10.13 and 3.10.14 Following figures provide a summary.

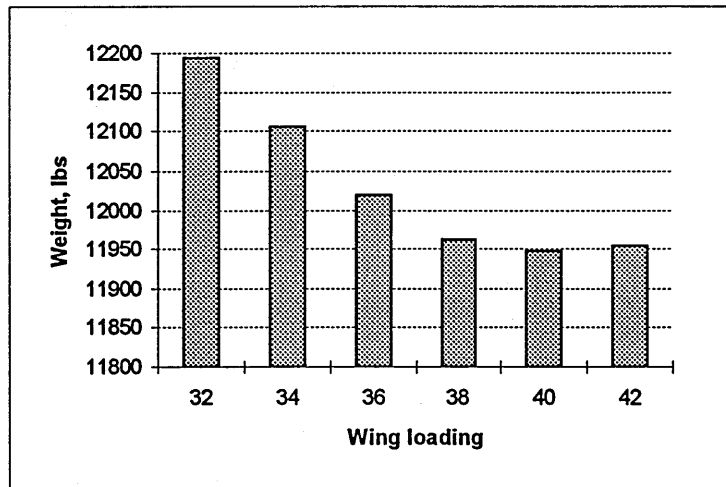


Fig. 3.10.13 Wing loading VS Take-off weight

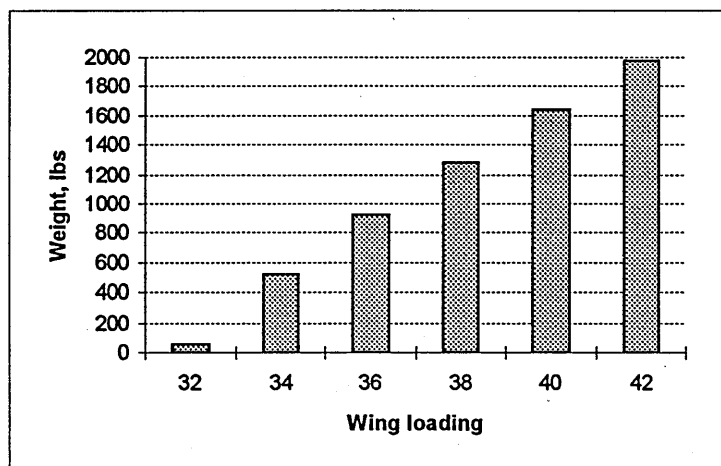


Fig. 3.10.14 Wing loading VS Fuel in fuselage

Fig. 3.10.13, as the same in chap. 2.9.1, shows the take-off gross weight decreases with increase in wing loading. This trend continues until the wing loading reaches a certain value, 40, then the aircraft take-off gross weight slightly increases again.

However, the benefit of higher wing loading and reduced take-off gross weight is not the only consideration. From fig. 3.10.14, as the wing loading increases, the wing area decreases and the wing fuel tank becomes smaller, and finally more and more fuel has to be put into the fuselage, this will constrain the payload installation. Therefore, which

value of wing loading is the best choice, is dependent on both the mission and payload requirement.

#### 3.10.7.9.1 Aspect ratio

From previous section, the aircraft wing loading as 32 psf has been selected. We substitute the aspect ratio as 17, 18 and 19 into this conceptual design program to do more calculation, and summarised into fig. 3.10.15.

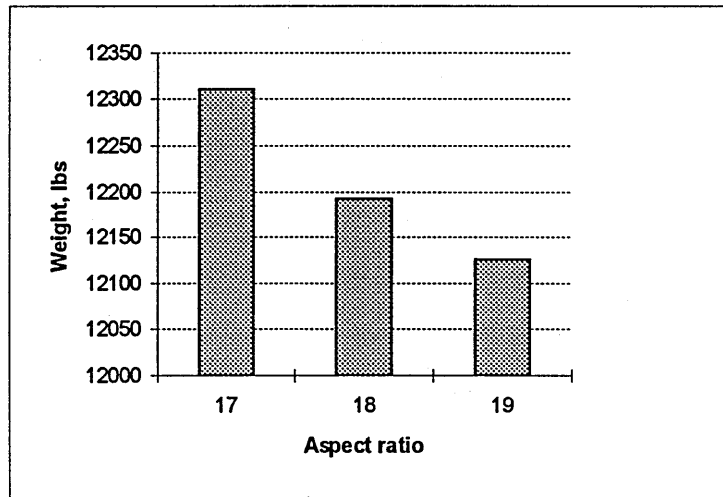


Fig. 3.10.15 Aspect ratio to take-off gross weight

From fig. 3.10.15, at different aspect ratio, the take-off weight change is within 0.9 percent. Therefore, in this case, the aspect ratio changes did not have too much effect to the aircraft take-off weight. However, it does suggest a higher aspect ratio than the 18 chosen might be advantageous.

### 3.10.8 Stability analysis

As in chap. 2.10, the aircraft needs to fly in a stable situation, stability.

#### 3.10.8.1 Static stability

The static stability of new HALE UAV is estimated as follows :

##### 3.10.8.1.1 Trim in cruise

As in chap. 2.10.1.1, the value of angle of incidence of horizontal surface,  $\alpha_T$ , during the cruise for trim can be calculated.

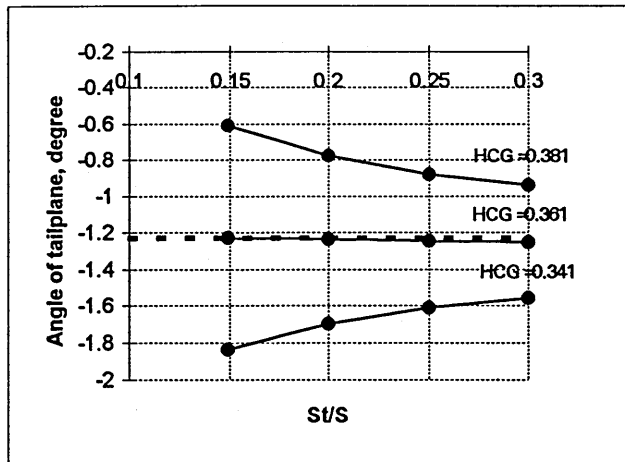


Fig. 3.10.16 Incidence of horizontal surface at cruise,  $\alpha_T$

After calculation, the incidence of horizontal surface at 70,000 ft, weight 9109 lbs, the trim angle of horizontal surface is -1.23 degrees.

### 3.10.8.1.2 Trim on approach to land

As in chap. 2.10.1.2, the angle of elevator deflection,  $\eta$ , is 1.1 degrees.

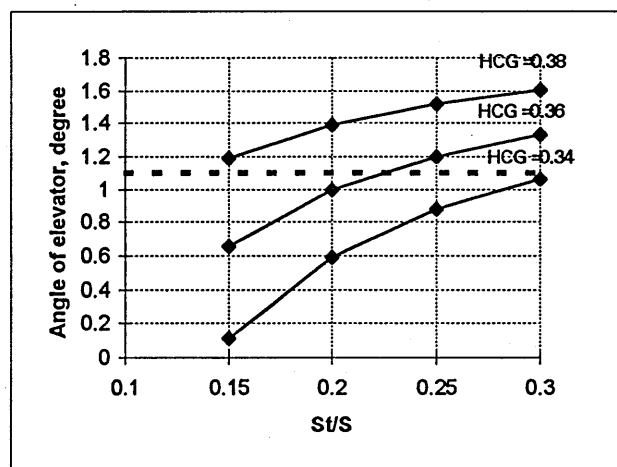


Fig. 3.10.17 Angle of elevator deflection,  $\eta$

### 3.10.8.1.3 Cruise static stability

As in chap. 2.10.1.3, the stick fixed static stability in cruise is in fig. 3.10.18.

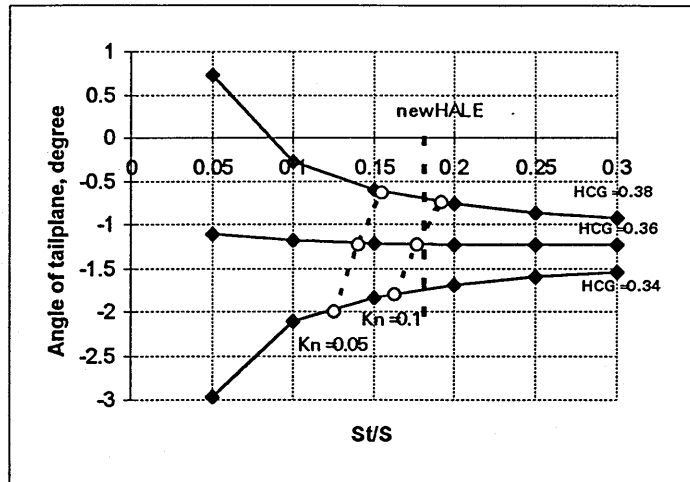


Fig. 3.10.18 Cruise static stability

### 3.10.8.1.4 Rotation at take off

The control surface chord regarding the wing chord, 0.3, and the rotation speed is 1.2 multiplied by stall speed, 158 fps. After calculation, the value of tail volume coefficient is 0.4, and the value of  $s_{T/S}$ , is 0.14.

By using eq. A.18.31, the value of tail volume coefficient,  $\bar{V}$ , of new HALE UAV is 0.527, the value of  $s_{T/S}$ , is 0.181. Therefore, it meets the requirements.

### 3.10.8.2 Dynamic stability

As in chap. 2.10.2, the following paragraphs are short period and long period mode.

#### 3.10.8.2.1 Short period oscillation

##### a. Short period damping ratio, $\zeta_s$

As in appendix A.18.5.1, short period damping ratio can be calculated. After calculation, the result is presented in fig. 3.10.19.

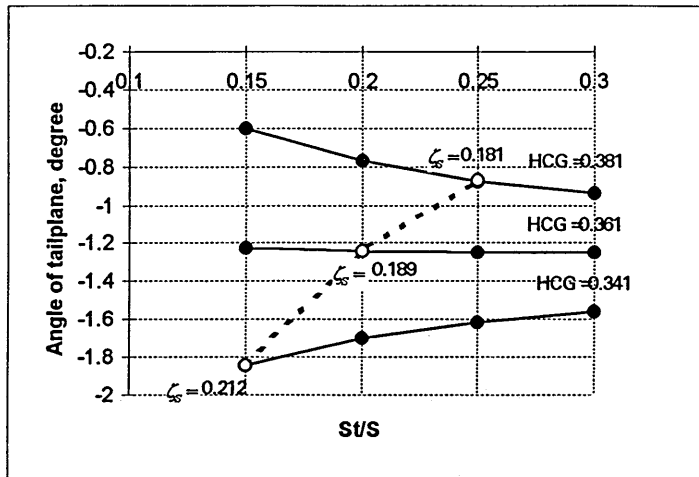


Fig. 3.10.19 Short period damping ratio,  $\zeta_s$

### b. Undamped short period frequency, $\omega_s$

As in appendix A.18.5.2, the undamped short period frequency can be estimated. After calculation, the results is listed in fig. 3.10.20.

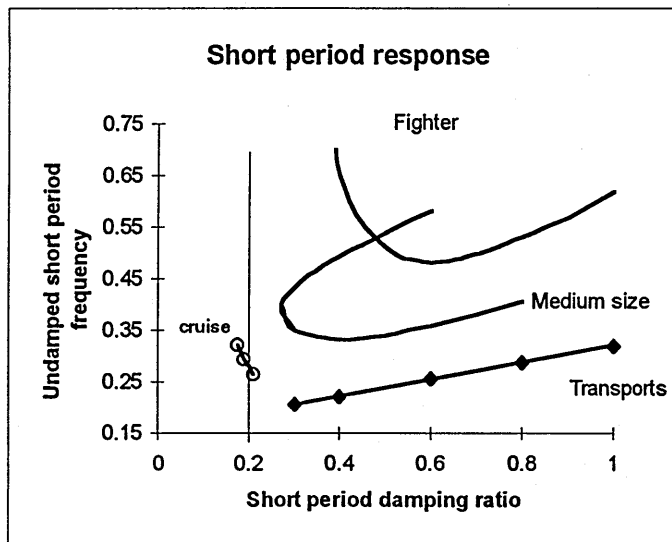


Fig. 3.10.20 Undamped short period frequency,  $\omega_s$

### 3.10.8.2.2 Long period mode

The damping ratio in the phugoid mode is;

- For aircraft C.G. is at 0.341,  $\zeta_L$  is 0.023.
- For aircraft C.G. is at 0.361,  $\zeta_L$  is 0.022.

As in chap. 2.10.2.2, a significant lower damping ratio assisted by a flight control system can provide adequate stability.



## **Chapter 4**

### **Cost analysis and general discussion**

In chapter 2, a conceptual design program of a high altitude long endurance UAV has been performed, and in chapter 3, by using the present Teledyne Tier-II Plus Global Hawk, we attempted to validate this program and even applied it to a new HALE UAV. The results of this work supported the use of this program in the conceptual design of High Altitude Long Endurance UAVs.

In this chapter we consider the cost of acquisition and operation of UAVs. For this work Tier-II Plus is chosen as an example because Tier-II Plus already exists, and United States is ready to build them. However, there is only limited information on Tier-II Plus costs, the cost analysis check can only be a rough comparison.

#### **4.1 Cost analysis**

Life cycle cost is a complex and wide ranging subject. It involves the application of significant resources over many years to collect. The accuracy is dependent upon the methodology and data availability.

Especially, in the 90's, considering the current emphasis on budgets reduction, in both national defence or civilian industries, the customers have become more concerned with the post purchase costs of a product instead of the acquisition cost. Thus, the life cycle cost becomes a very important factor to let the decision maker decide whether to proceed with purchasing a new product or not.

In reference 53, there is an example of comparing the cost of Sukhoi Su-27 and F-16 Eagle. Based on the market purchase price Su-27 is cheaper than F-16, but when taking life cycle cost into consideration, the F-16, costs of the operation, installation, maintenance and refitting, are much less than Su-27. Thus, in terms of the life cycle cost, F-16 is much cheaper than Su-27. Considering the gradually reduced defence budget, the F-16 become the winner.

Reference 52, chap 2, stated that the life cycle cost, LCC, of an airplane programme can be expressed as follows;

$$LCC = C_{RDTE} + C_{ACQ} + C_{OPS} + C_{DISP} \quad (4.1.1)$$

and  $C_{RDTE}$  is the research, development, test and evaluation cost.  $C_{ACQ}$  is the acquisition cost,  $C_{OPS}$  is the operating cost and  $C_{DISP}$  is the disposal cost.

Reference 10, fig. 18.2, stated the elements of life cycle cost.

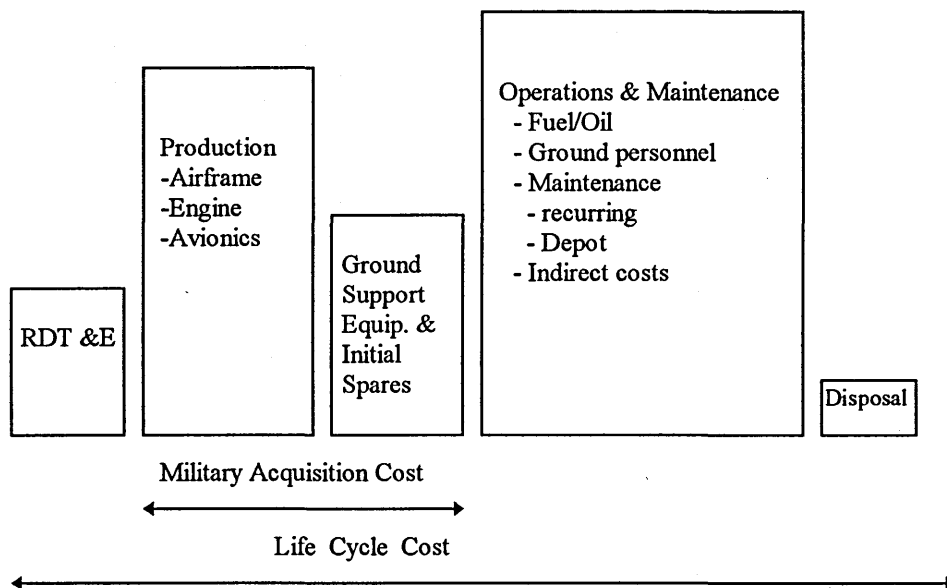


Fig. 4.1.1 Elements of Life Cycle Cost (from ref. 10)

In fig. 4.1.1, RDT&E stands for research, development, test and evaluation, which includes all the technology research, design engineering, prototype fabrication etc. Should an accurate RDT&E cost need to be estimated, detailed aircraft research and manufacturing historical data should be established. Otherwise, it is difficult to estimate RDT&E cost.

“Disposal” means when an airplane no longer has civilian or military value. It has either reached its safe structure life or is obsolete compared to modern aircraft. Usually the expense is not large, and its cost is dependent on different usage, so, its contribution to life cycle cost, is frequently ignored.

Due to the difficulty to estimate RDT&E and disposal cost, in this paragraph, we decided not to estimate a complete aircraft life cycle cost. Instead, in this paragraph, we are trying to estimate the most important part of life cycle cost, the acquisition cost,  $C_{ACQ}$ , and the operating cost,  $C_{OPS}$ . The detailed estimation process are listed in Appendix C.

#### 4.1.1 Operating cost estimation

Reference 8, chap. 24 stated that the operating cost are based upon a period of operation, a fleet size and number of flying hours per year. Of course, if the aircraft in an service longer, the fleet size is larger and the flying hours more, this will cause the operating cost to increase. From fig. 4.1.1, the operating cost is the most expensive part in whole airplane life cycle cost.

Reference 52, chap. 6 stated that the military airplane is acquired in a manner depending considerably on budgetary and political constraints. It is very difficult to translate such constraints into any rational formula. Therefore, some assumption have to be made before estimating operating cost, such as the number of airplanes and the years in military service.

Reference 52, chap. 6 stated that the operation cost,  $C_{OPS}$ , for military aircraft, comprises of seven parts. After calculation (the detail are in appendix C.1), we obtain the operation cost as in table 4.1.1.

| Items                                                       | Cost, USD                    | Remark                                                           |
|-------------------------------------------------------------|------------------------------|------------------------------------------------------------------|
| • The program cost of fuel , oil, and lubricants, $C_{pol}$ | 7,541,614                    |                                                                  |
| • The program cost of direct personnel, $C_{PDIR}$          | 208,808,972 or 417,617,944   | MHR <sub>flt</sub> is 25 hours<br>MHR <sub>flt</sub> is 50 hours |
| • The program cost of indirect personnel, $C_{PIND}$        | $0.14C_{OPS}$                |                                                                  |
| • The program cost of consumable materials, $C_{CMAT}$      | 30,159,522 or 60,319,045     | MHR <sub>flt</sub> is 25 hours<br>MHR <sub>flt</sub> is 50 hours |
| • The program cost of spares, $C_{SP}$                      | $0.25C_{OPS}$                |                                                                  |
| • The program cost associated with depots, $C_{DBP}$        | $0.2C_{OPS}$                 |                                                                  |
| • The program miscel. cost, $C_{MIS}$                       | $4C_{CMAT}$                  |                                                                  |
| The operation cost of military airplanes, $C_{OPS}$         | 917,870,490 or 1,816,886,958 | MHR <sub>flt</sub> is 25 hours<br>MHR <sub>flt</sub> is 50 hours |

Table 4.1.1 The operation cost of military airplanes

The difference in  $C_{OPS}$  is due to the number of maintenance manhours,  $MHR_{fl}$ , required per flight hour, which is assumed to be between 25 and 50.

### 4.1.2 Manufacturing and Acquisition cost estimation

Reference 52 stated that the difference between the manufacturing cost,  $C_{MAN}$ , and the acquisition cost,  $C_{ACQ}$ , is the profit made by the manufacturer. Thus, if we want to estimate the acquisition cost,  $C_{ACQ}$ , the value of the manufacturing cost,  $C_{MAN}$ , has to be obtained first.

The price paid by the customer of an airplane depends on a number of factors, such as the total number of airplane built, the manufacturers profit and the cost of the RDT&E. After calculation ( the detailed calculation is in appendix C.2), the results are listed in table 4.1.2.

| Items                                                     | Cost, USD                  | Remark |
|-----------------------------------------------------------|----------------------------|--------|
| • The airframe engineering and design cost, $C_{aedm}$    | 41,631,485                 |        |
| • The airplane program production cost, $C_{apcm}$        | 123,007,003                |        |
| • The production flight test operations cost, $C_{ftom}$  | 3,406,080 or 6,742,380     |        |
| • The cost to finance the manufacturing phase, $C_{finm}$ | 0.12 $C_{MAN}$             |        |
| The total airplane program manufacturing cost, $C_{MAN}$  | 190,959,736 or 194,750,986 |        |

Table 4.1.2 The total airplane program manufacturing cost

The profit is decided by the manufacturer and market, and will be included in the cost of airplane acquisition. From reference 52, p: 56, assumes the profit factor is 10 percent of manufacturing cost, after calculation, we put the results in table 4.1.3.

| Items                                                      | Cost, USD                  | Remark |
|------------------------------------------------------------|----------------------------|--------|
| • The total airplane program manufacturing cost, $C_{MAN}$ | 190,959,736 or 194,750,986 |        |
| • Profit, $C_{PRO}$                                        | 19,095,974 or 19,475,099   |        |
| Acquisition cost, $C_{ACQ}$                                | 210,055,710 or 214,226,085 |        |

Table 4.1.3 The acquisition cost

By Comparing table 4.1.1 and 4.1.3, it is easier to notice that the operation cost is much more expensive than acquisition cost. And from fig. 4.1.1, again proved the most important part is in the aircraft life cycle cost is the operation cost.

The acquisition cost per aircraft is around USD 20 million, much higher than the US government estimated USD 10 million each aircraft. However, from reference 114, the USD 10 million is the unit fly away cost, it includes the aircraft, sensors and datalink communications suite. If takes ground support systems into consideration, the price will go up to USD 20 to 25 million per set.

At the beginning we stated that the cost analysis is very difficult. In addition to those used during our calculation there are many parameters which are unknown, such as manhour rate, engine cost, avionics cost etc. Therefore, the above calculation is acceptable for academic study but, for real usage, we need more accurate data.

## **4.2 General discussion**

The scope of application UAVs is considerable, ranging from tactical to strategic military operations and increasingly in the civil applications. The size, operational duration and complexity varies enormously.

The core mission of military missions include day or night reconnaissance, surveillance and target acquisition (RSTA), combat assessment and battle management. Compared with manned aircraft, any UAVs loss does endanger pilots life. But the UAVs for military usage are susceptible to detection and countermeasures in the same way as manned aircraft. With the demand of longer operation duration, and the more sophisticated equipment we need to install on board, the price of UAVs is no longer less than manned aircraft. Therefore, any loss of UAVs is not easy to tolerate.

Due to the above reasons. In this paragraph, we are going to discuss the importance of survivability, the benefit of high altitude of UAVs, and future challenge of UAVs.

### **4.2.1 Survivability**

Reference 56 stated that with the advanced technology coming, in the near future, new and increasingly dangerous weapon will turn up in military operations. Thus, in order to counter the ever increasing quantity and quality of the threat, so that the mission can be effectively carry out, the aircraft survivability plays an important role, and becomes more and more important to modern aircraft.

Reference 57 states the definition of aircraft combat survivability is “the capability of

an aircraft to avoid and/or withstand a man-made hostile environment.” Furthermore, from reference 58, a survival system is a safer system during peacetime operations. Saving even a few aircraft during peacetime operations can provide significantly life cycle cost savings.

From reference 57, eq. 1.1, the probability of survival,  $P_s$ , can be expressed as a function of susceptibility,  $P_H$ , and vulnerability,  $P_{K/H}$ .

$$P_s = 1 - P_H P_{K/H} \quad (4.2.1)$$

That means, if the susceptibility and vulnerability can be improved, then the aircraft survivability will be increased.

In following sections, we will take an unmanned UAV, the Tier-II Plus, as an example, to evaluate the susceptibility and vulnerability respectively and combining them into effectiveness of a survivability.

#### 4.2.1.1 Susceptibility

Reference 56 states the definition of susceptibility is “the characteristics of a system which allow it to perform its essential functions as a result of its capability to avoid being detected in an unnatural hostile environment.” And from reference 58, it lists six basic concepts to enhance susceptibility. They are signature reduction, threat suppression, tactics, threat warning, noise jamming and deceiving, and expendable. Following is their evaluation on Tier-II plus.

##### A. Signature reduction:

From reference 57, the aircraft signature or observability are the characteristics of the aircraft that are used by the threat non terminal elements for detection and tracking. Aircraft have many different kinds of signature, but in this section, we only discussing radar signature: radar cross section (RCS).

When a radar antenna radiates an electromagnetic pulse or continuous wave in the direction of an airborne aircraft, a portion of the incident energy may be absorbed as heat, and the remainder may pass completely through parts of the aircraft or be radiated in many different directions by the aircraft surface. The radiated portion is known as the aircraft echo, and is the aircraft’s radar signature. The size of the signature is referred to as the aircraft’s radar cross section. Basically, from reference 57, the radar cross section is determined by the frequency and polarisation of the incident wave and the target aspect, such as aircraft size, shape and material.

As we know, if the radar cross section is reduced to a certain limit, and its echo return

is lower than the radar detection threshold, then we say this target is undetectable, or stealthy.

Due to the budget constraints, the Tier-II Plus is not designed for absolute stealth in the first place, but reference 26 stated that the Tier-II Plus program is still trying to reduce its radar cross section. They claim, depending on the look angle, the Tier-II Plus can have a radar cross section as low as one square meter. Reference 59, fig. 1.4 states that the typical values of radar cross section are:

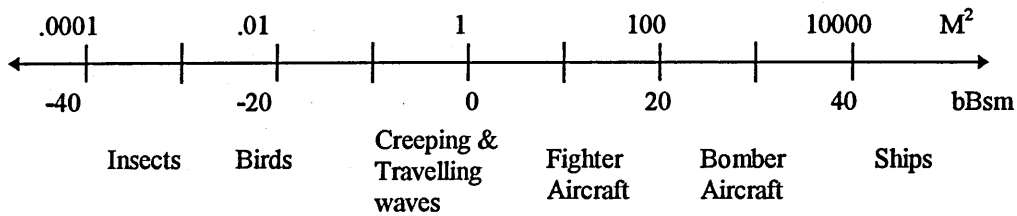


Fig. 4.2.1 Typical values of RCS

From fig. 4.2.1, the RCS for the fighter aircraft is from 10 to 100 square meter, for the bomber aircraft is about 1,000 square meter. Thus, if the RCS is one meter square, such a low RCS targets will be a difficult job for radar to detect. In this aspect, Tier-II Plus already has a very low delectability, or is stealthy.

#### B. Threat suppression:

From reference 57, p: 306, the threat suppression consists of actions taken by friendly forces with the intent to physically damage or destroy part or all of a threat system.

The Tier-II Plus has synthetic aperture radar, electro-optical and infrared sensors, and will transmit imagery via satellite communication. The near real time intelligence and targeting information is expected to significantly enhance battle situation awareness and allow a field friendly force to rapidly destroy enemy threat systems. Therefore, any hostile weapons, such as SAM, before launching should be reduce or destroyed to a minimum level.

#### C. Tactics:

Tactics is the employment of units in combat. Typically the tactics consist of the flight profiles, and operation to accomplish the mission. According to reference 26, the Russian-built SA-5,-10s and -12s are among the few weapons that have an effective operational altitude up to 90,000 ft, but the lethality of high altitude SAMs, they tend to be few in number and concentrated around high value targets.

The centre of the Tier-II Plus mission planning system, is an automated system in

which potential threats to the vehicle are detected, and the flight profile is automatically adjusted for threat avoidance. Thus, it can allow the Tier-II Plus avoid high density SAM threat area, and the altitude of Tier-II Plus should protect it from most ground launched weapons and subsonic air-launched missiles.

#### **D. Threat warning, noise jamming and deceiving, and expendable:**

Knowledge of the location, type, and status of the threat elements in the vicinity of an aircraft is extremely important to the aircraft's survival.

From reference 26, and 60, the Electronic Support Measures (ESM) onboard Tier-II Plus includes a Threat Warning system, Electronic Countermeasures and Towed Decoys. That means the aircraft will have threat warning and detection, in addition to countermeasures. It will include a jamming capability and a reel-out/reel-in decoy deception system. And the full suite is being developed by Raytheon/E-Systems.

The radar warning receiver detects the hostile radar-directed weapons which surround the aircraft. From this information it determines the classification (radar type), location (range and bearing) etc. Then the suite will identify the threats and to react in a specific way. The decoy system works in connect with an E-Systems ALR-89 radar warning receiver. The jammer system, also made by E-Systems, is operated in two electro-magnetic spectrums to engage older lower-frequency missiles and radars. While a second band will concentrate on the newer, high-frequency weapons, like Russian-made SA-10s and 12s.

#### **4.2.1.2 Vulnerability**

Reference 56 states the definition of vulnerability is "the characteristics of a system which cause it to suffer finite degradation in its capability to perform its essential functions after having been subjected to an unnatural hostile environment." And from reference 58, it lists six basic concepts to enhance vulnerability. They are component redundancy with separation, component location, passive damage suppression, active damage suppression, component shielding, and component elimination.

Due to lacking of real information, it is hard to predict if any damage is likely to occur to Tier-II plus. However, the numerous capabilities of avionics systems enhance the Tier-II Plus's ability to survive significant, proliferated threats anywhere in the world.

#### **4.2.2 The benefit of high altitude**

Reference 55 stated that the unmanned air vehicle market is set to grow to around USD 1 billion per year by the turn of the century. The reason is the understanding of UAVs in both military and civilian sector were increasing, and the reliability will be



improved by using modern technology. Thus, the cost of UAVs will fall and more and more countries will be involved in this field.

For many nations, range and payload requirement will increase. They need the UAV to fly higher and stay longer, so that the UAV can cover more strategic reconnaissance and transmit real time battle information.

Naturally, the higher the UAVs fly, the more range it can cover, but to define the altitude at which a UAV can most effectively carry out its mission requires more research. Following are the advantages and disadvantages of high altitude.

#### 4.2.2.1 Advantages

##### A. Surveillance bigger area

Reference 69 states that if the electro-optical aerial reconnaissance systems is the same, the higher the aircraft flies, the more area coverage it could provide. Fig. 4.2.2 is an example showing an aircraft flight at an altitude of 30,000 ft and 60,000 ft and carrying the same electro-optical aerial reconnaissance systems. If the flight speed is 0.6 M, in one hour flight, the operational coverage will be 6,041 and 11,757 square NM respectively. The difference of 30,000 ft and 60,000 ft, is two fold. Therefore, basically the longer and the higher the aircraft flies, the more operational coverage it could provide.

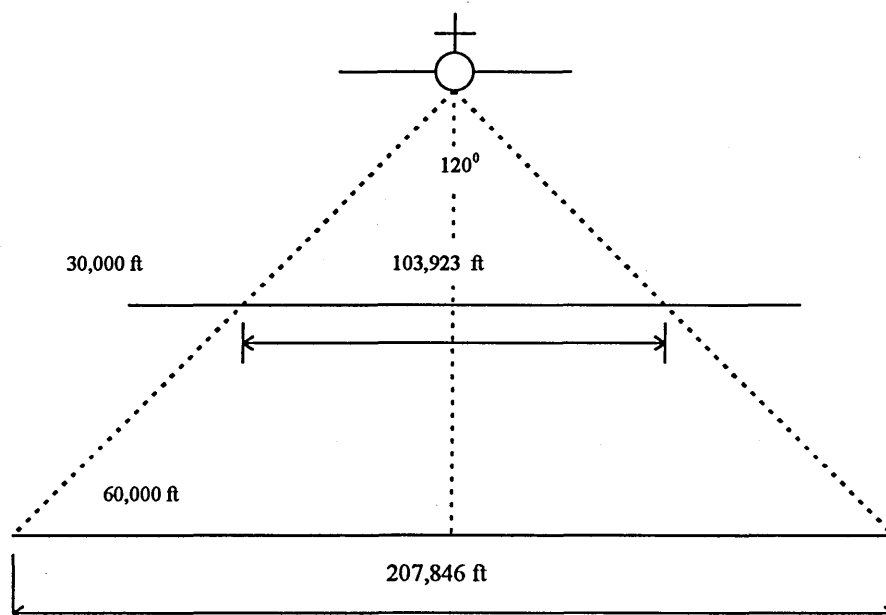


Fig. 4.2.2 Ground coverage illustration

### B. Increase survivability

The radar cross section is independent of altitude, with the technology improvements, there is no real technical difficulty in the detection of targets at altitude from 60,000 to 70,000 ft. Even with the assistance from space based satellite systems, the high altitude aircraft is not too difficult to detect.

However, the difficulty is from the design consideration and the budget limitation stand point. The former includes frequency, power, antenna size etc., the latter is the need to build this sophisticated military system, which may have requirements for more than just direction, giving a cost which could exceed the USD 100 million. Therefore, even if some countries have the capability to manufacture this special purpose radar, due to the defence budget limitation, it could prove too large a cost. On this basis, if the radar information can be collected, and input to the mission plan system, then we can avoid areas with capability and increase the survivability.

### C. The Calmest layer

Reference 50 stated that the atmosphere is an enormously energetic system. Practically all the power to drive it comes from the sun. From fig.1, reference 50, it is easy to see that the pressure and temperature changes with altitude.

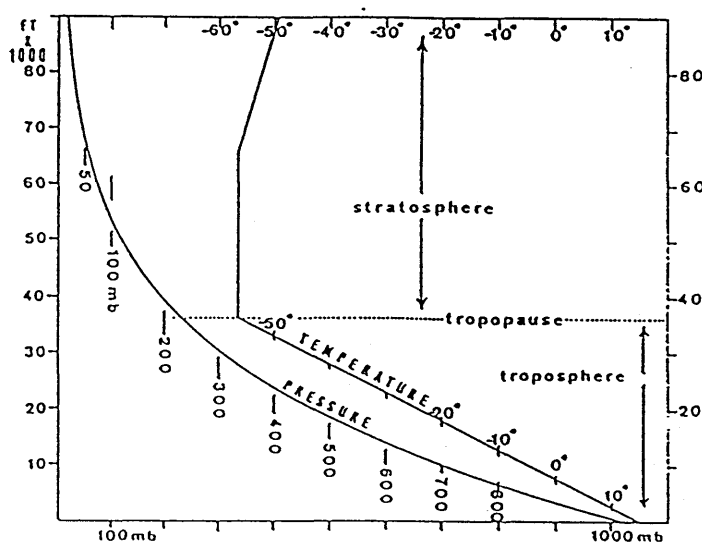


Fig. 4.2.3 Pressure and temperature varies with altitude

In fig. 4.2.3, the altitude is from sea-level to 90,000 ft. In this figure, the pressure decreases with altitude, and the temperature, in troposphere, decreases with altitude. In stratosphere, the temperature is constant until the altitude reaches 66,000 ft, then the temperature will increase slowly with altitude.

There are three main factors of the atmosphere which will influence the aircraft, these are Jet streams, cloud and air stability. The following sections will discuss these factors.

- **Jet-streams and wind shear**

Jet streams are bands of very strong winds which are usually found at levels between 30,000 to 40,000 ft. Sometimes a jet stream is more than a thousand miles long, one to two hundred miles wide and less than four miles deep. Most jet stream will occur where there is a marked contrast in temperature and its speed, in the core, could easily reaches 100 Knots per hour or more.

When a band of air is travelling at speeds of 100 Knots or more, a strong wind shear will be generated. The strong wind shear causes the airstream to break down into turbulent eddies moving with the wind. The result can be violent buffeting of an aeroplane, up and down and side to side. Eventually, the structure might be damaged and the control much more difficulty.

Assuming the height of jet stream is at 40,000 ft, then including the depth, four miles, the extreme height of jet stream may reach to 55,000 ft, still far less than 65,000 ft. Besides, with nowadays weather forecast system, the jet stream is not difficult to be avoided during climb and descend. Therefore, if the Tier-II Plus cruise at 65,000 ft, then the influence of jet stream and wind shear will reduce to a minimum.

- **Cloud**

Reference 51, fig 6.1, shows that in general the cloud be subdivided into three main classes: cumulus, stratus and cirrus.

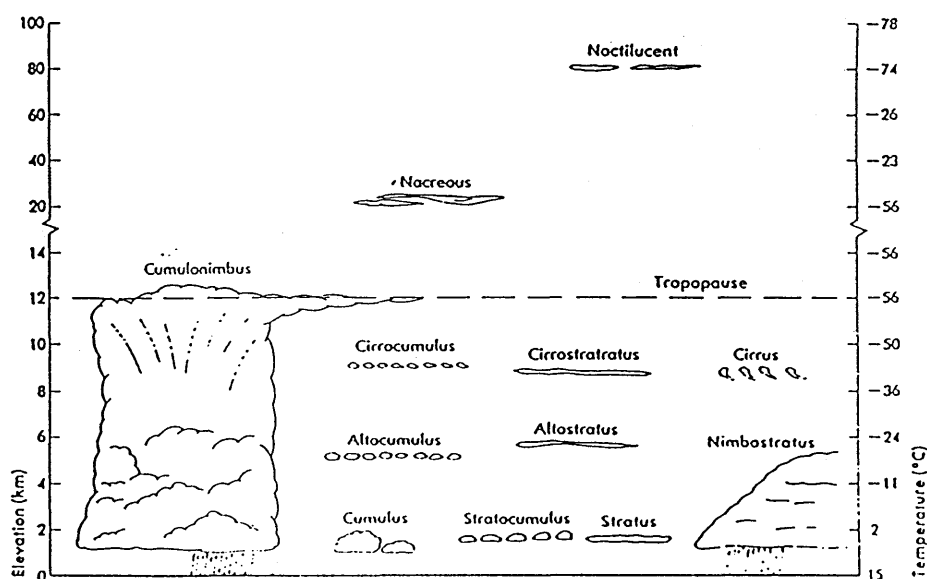


Fig. 4.2.4 Cloud genera in atmosphere

Potentially the most dangerous cloud to the aircraft is cumulus cloud, especially the cumulonimbus. They can produce violent up and down currents, destructive turbulence and very severe icing. The down drafts may reach the ground with sufficient force to blow down trees and the outflow may produce squalls strong enough to blow over grounded aircraft.

However, no matter how violent the cumulonimbus is or how destructive it will blow down the aircraft. From fig. 4.2.4, the highest cumulonimbus are below 50,000 ft. Therefore, if the aircraft cruises at 65,000 ft, the cumulonimbus will do no harm to it.

- **Air stability**

Air stability means a tendency for air to return to its original level after an initial vertical displacement. A parcel of air tends to rise or move down depending on density. In other words, if the air is less dense than its surroundings it will rise, if it is more dense it will sink. And the density depends on the mass of air within a given volume. If the air is heated it will expand to fill a larger volume of space and its density will become less. Thus, it will rise.

Reference 50, chap. 4 states that the absolute stability of air is when the environmental lapse rate, which temperature varies with altitude, is less than the saturated lapse rate, which the air can hold a limited amount of water vapour varies with air temperature. For example, if the temperature only decreased by 0.5 degrees per thousand feet the air would be absolutely stable. Because its lapse rate is less than the saturated rate.

From fig. 4.2.3, it's easy to tell that the temperature in stratosphere layer and for altitude below 66,000 ft, the temperature stays constant. Therefore, the temperature decrease per thousand feet is nearly 0 degrees. Thus, the air in this layer has absolute stability.

Apart from above factors, in this region the air is normally very dry, so it is very unusual for any clouds to appear. Therefore the visibility is also very good.

#### **4.2.2.2 Disadvantages**

##### **A. Need large engine**

For aircraft flight at high altitude, the most difficult problem will be the engine thrust available. In our examples, the aircraft propulsion system is turbofan engine. For turbofan engine, from the example shown in chap. 2.8.3, one pound thrust at sea level will become 0.074 pound as the altitude reaches to 65,000 ft. Therefore, if we need 500 pound thrust to maintain level flight at 65,000 ft, then the sea level net thrust should no less than 6,757 pounds.

However, the larger thrust the engine needs to generate, the more fuel consumption is needed. Apart from this, the weight of the engine will increase and the aircraft weight will increase as well.

#### **B. Cost increases**

Nowadays the latest electro-optical systems can be applied to get day and night accurate long range real time images without difficulty. However, the higher the aircraft flies, the more sophisticated the equipment it needs to generate accurate picture. In addition to on board systems, the ground systems, to receive very sensitive data transmit from high altitude, should be sophisticated as well. The more sophisticated the equipment the greater the cost, requiring a larger proportion of the defense budget and the greater the difficulty to convince government to buy.

#### **C. Increased risk**

As we know, any high technology has its own risk. High altitude long endurance UAV is no exception. Last year the crash of Dark Star during the second test flight, caused a million dollars loss, fortunately there was no “man” loss, and up to now the real problem is still under investigation. It proves that the automatic control system still has potential problems, and without “manned” on board real time correction, its safety and reliability needs to be enhanced.

### **4.2.3 Future challenge of UAVs**

Reference 70 stated that “the United States Department of Defence is developing a series of unmanned aerial vehicles to support tactical and strategic requirement of the Battle force Commander. These vehicles are being developed to complement manned aircraft during collection of information on targets that are potentially sensitive or protected by heavy defences that provide high risk to manned reconnaissance aircraft with the potential of loss of life or occurrence of prisoners of war.”

Indeed with the extremely successful employment of UAVs during the Persian Gulf war, the commanders of US Army, Navy and Marine are lauded of UAVs operational effectiveness. They were extremely successful in target selection, damage assessment, near real time reconnaissance and target confirmations and locations. The success of UAVs in supporting the operation, established the utility and importance of UAVs in combat. A few years ago, only a few companies such as Teledyne Ryan Aeronautical and Israel Aircraft Industries showed interest in UAVs, but now many companies are so certain, even with their own funds to develop UAVs programs.

However, even though UAVs are increasingly being considered as alternatives to traditional land, sea and air systems, there is still a long way to go and many challenges needs to be solved.

#### **4.2.3.1 Manned VS Unmanned**

Reference 70 stated that the issue of manned versus unmanned aerospace vehicles is an often controversial subject. The arguments are often framed in terms of past experience, technical considerations, cost etc. Usually the pro unmanned vehicle lobby will claim that when man is removed from the aircraft cabin, the reliability requirement is not that high and the aircraft dimension can be sub-scaled, therefore the cost will reduce to a certain amount. This is true for two reasons, first the unmanned UAVs are really sub-scaled. Second, the unmanned systems used to be expendable, therefore, they did not have the reliability and redundancy required of a manned system.

However, as the technology improves, and the usage of UAVs becomes versatile, some of the UAVs dimension, software and reliability requirements even exceed those of manned aircraft. The result is the UAVs cost may be more than manned vehicle. Therefore, the cost is no longer a big issue to argue.

Instead, as we stated previously, if there is no pilot, there will be great weight and cost saving associated with removing the cockpit and environmental, safety equipment. Without pilots, then the engine could be positioned more effectively and the aircraft could be manoeuvred more violently (aircraft technology could sustain 20g, but the limit for human beings is far less than this figure). Besides, in operational needs, some of the missions such as high risk, high pollution, political sensitive, and endurance flight (more than 40 hours, not good for human beings), may prove the unmanned air vehicles are superior than manned vehicles in some aspects.

#### **4.2.3.2 Usage for both civilian and military applications**

In the previous statements, we are focus on military usage of UAVs, such as reconnaissance, surveillance, battle assessment etc. However, as the cold war gradually fades away, nations conflicts and tension are not that high, or as due to the political sensitive, the demand for military high altitude endurance UAVs may very possibly decline. Then the cost of advanced UAVs would be very expensive and lose out in comparison with other aerospace vehicle. Therefore, in the long run, apart from military usage, the UAVs should take civilian usage into consideration, such as high altitude endurance atmospheric, storm research, meteorology and mapping, ocean fishing law enforcement, relay of radio, TV and communication etc.

All of these missions require continuous coverage and, by developing the multi-usage UAVs in both military and civilian market, there will be an increase the demand of UAVs. As the quantities increases, the cost can be brought down, and in the future, with the advantages of UAVs, they become more competitive.

#### **4.2.3.3 Environmental and Space management**

The hottest political issues nowadays are environmental protection and space management. Manned aircraft create problems such as noise, pollution, ozone layer depletion etc., and space management is a new issue worth discussing, up to now only a few countries, such as USA, Switzerland, South Africa have agreements to let UAVs transit civil air space.

For environmental protection, we should integrate this ideas into the initial aircraft design so that in the long term the UVAs could meet this requirement. In terms of space management of UAVs, a special conference should be held by those countries which are intend to build or interesting in UAVs, and the organisations or regulations, such as the International Civil Aviation Organisation, ICAO or the Civil Aviation Authority, CAA, should be organised. A certain degree of regulation is necessary in such areas as public safety, operational organisation and international relations should be define clear so that every country can follow.

#### **4.2.3.4 Safety and reliability**

Reference 72 stated that the reliability record of UAVs seems to be disappointing, largely due to unforeseen circumstances. Underestimates of the requirement for quality in the design, testing and production of a UAV. Taking the pilot out of the vehicle does not give us a free ride of quality. On the contrary, without manned operation, the reliability requirement has to be enhanced beyond that of manned vehicles.

In this aspect, we should put more efforts to the UAVs design and manufacture, so that the reliability and safety record of UAVs can compete with manned vehicle.

The UAVs challenge and future applications is a broad topic. The above four factors are only part of the problem. The more research performed, the more problems will arise. However, it is believed that in the 21 century, UAVs' future is bright.

## Chapter 5

### Conclusion and future work

#### 5.1 Introduction

Reference 73 stated that Teledyne Ryan and Boeing in the early 1970s had already developed the turbofan powered Compass Cope, and flew it up to 57,000 ft. However due to lacking the truly autonomous, long endurance and high altitude capability desired, this UAV was not truly employed. In 1990 Condor, a large, high altitude long endurance UAV, achieved two world records, one for endurance of 58 hours, 11 minutes, one for an altitude for 66,980 ft. Unlike other unmanned or remotely piloted air vehicles, Condor flies autonomously from take-off through landing. Rather than using turbofan engine, Condor uses turbocharged reciprocating engines. At high altitude, cruise speed is less than 200 knots (0.35 M), very close to stall, the maximum lift drag ratio is near 40.

Condor provided a very successful demonstration of an autonomously controlled high altitude long endurance UAV. The only concern is if we want to employ Condor to a battle field rapidly, with its speed, Condor can not meet the requirement of fast deployment. Therefore, Condor is not an ideal UAV in terms of military usage.

Within this thesis the discussion is mainly focused on military usage high altitude long endurance UAVs. The reason for focusing on military applications is that for the last two decades, the battle over UAVs has raged within military services. Following the successfully employment of Pioneer's in Desert Storm, suddenly every field



commander wants them. Recently Pentagon plans to field long endurance unmanned air vehicles for all weather, real time surveillance were passed in late May 1995. This announcement aroused huge response all over the world.

With the state of art technology, the previous difficulties such as structure, propulsion, aerodynamics and autonomous controls can be addressed. The main requirement to cover all battlefields all the time can be achieved by Tier-II Plus Global Hawk. For high risk and high political sensitivities areas, the stealthy Dark Star is the best selection. Due to cost of satellites, slow repositioning times, and dependence on weather and operational altitude, their applications are restricted. Lighter than air vehicles are limited in manoeuvrability, velocity and altitude. Therefore, there is no reason to expect the demand of Tier-II Plus and Dark Star to diminish.

## **5.2 Conclusion**

In this thesis, a conceptual design program for high altitude long endurance, subsonic, UAVs program has been developed. This assumes a turbofan engine as the propulsion system, the main reason being that with this kind of engine, can provide an aircraft which could fly up to 0.65M at high altitude. However, if the propulsion system is replaced by other power systems, such as turbocharged reciprocated engine, this program still can be used as a basis. The major difference will be if the speed is around or less than 0.3 M, the Prandtl-Glauert compressibility factor,  $\beta$ , can be ignored. Therefore, the main objective of developing a flexible, subsonic high altitude long endurance UAV methodology has been achieved. However, this program is not the same as other user friendly airplane design software, it is not simply follow the interactive windows, input the mission requirements and the output is an airplane which meets your need. Instead, before using this program to do conceptual design, you should go through this program, as stated in chap. 2.3, it comprises four different blocks, such as parametric study, airplane weight and fuel fraction estimation, performance estimation etc., then tailor your requirements into each block, finally the output may meet your requirements. Therefore, this program is a flexible conceptual design methodology, and the objective of this thesis has been achieved.

During the development process, a number of interesting topics has been encountered. These are summarised as follows;

### **5.2.1 Engine data**

As stated in the previous section, the propulsion system in this program is a turbofan engine. In general, turbofan engines are operating at the altitude around 40,000 ft,

above this altitude, it is difficult to acquire the real engine data map, which we need to estimate airplane performance.

Instead, we have to use a TURBOMATCH program to generate the engine data which we need to do analysis. By doing this, three problems appear, first, we have to have a whole picture about turbine engines, and to write an engine program, finally run the engine program. Second, the results have no comparison with other similar engines, therefore, its accuracy is not easy to define.

This process, to obtain a whole picture of turbine engine and run the engine program, takes a couple of months to finish, and the results, without validation, can only really be justified for academic study.

### **5.2.2 Engine dimension and weight**

Apart from engine data, the engine dimensions and weight are another issue. From Jane's all the World Aircraft, the only thing we can acquire is similar engine's dimensions and weight. It does not include engine nacelle dimensions and weight. Therefore, it is difficult to estimate the whole engine installation dimensions and weight.

### **5.2.3 Oswald efficiency**

As stated in chap. 2.7.2, if a wing has elliptic spanwise lift distribution, the Oswald efficiency is 1. Unfortunately, the spanwise lift distribution of most wings is not elliptic. Therefore we need a factor, Oswald efficiency, to provide some modification. Many textbooks introduce how to estimate Oswald efficiency, but most of them are only valid when the aspect ratio is less than a certain value. It is hard to find a method which can estimate Oswald efficiency when the aspect ratio is higher than 20.

In the previous calculation, for a high aspect ratio UAV, the proportion of wing drag, compared to total drag, is ranging from 75 to 80 percent. By further analysis, wing lift coefficient drag is larger than wing zero lift drag. However, the Oswald efficiency plays an important role in wing lift drag coefficient, so if the Oswald efficiency can be effectively increased, the wing lift drag will reduce, the total drag can be reduced, lift drag ratio will increase, and the thrust required can be reduced by a certain amount.

### **5.2.4 High aspect ratio and low Reynolds number**

Since the high altitude long endurance UAVs fly at an altitude around 65,000 ft, if the speed is within the subsonic range, the Reynolds number will be very low. Unfortunately, up to now, the information about low Reynolds number is not sufficient to provide confident predictions, such as estimating the maximum lift coefficient of

wing tip and tail airfoil sections at high altitude. In addition, the data on ESDU data sheets, tends to be limited to an aspect ratio within 10, there are no definitive answers for high aspect ratio wings.

### **5.2.5 Parametric study**

During conceptual design, the first step is to perform parametric study. By doing this, a matching point may be obtained, and the equivalent take-off thrust weight ratio, and wing loading can be acquired.

For common aircraft flying altitude and the aircraft weight variation is not that severe compared with that of high altitude long endurance UAVs. Therefore, when performing the parametric study, every steps such as take-off distance, climb, cruise, and landing needs to be considered. For high altitude long endurance UAVs, from the previous examples, we find only one step, especially for flying higher than 65,000 ft, is needed to perform parametric study. This is the cruise requirement, because it is the dominant segment for high altitude long endurance UAVs.

### **5.2.6 Aircraft weight estimation**

Most of the initial aircraft weight estimation methods presented in reference 1, 8, 9, 10 are for ordinary aircraft, such as fighter, cargo and commercial airliner aircraft, none of them mentioned how to estimate the aircraft weight of unconventional, high altitude long endurance UAVs. In order to simplify and obtain an estimated result which is not too far from the requirement, by comparing with reference 66, the method presented in reference 10, estimating weight of general aviation aircraft, has been chosen. Since this method is not designed for estimating high altitude long endurance UAVs, the results of predict initial aircraft weight should be treated with some caution.

## **5.3 Future work**

Though references 55 and 75 predicted the world-wide market of UAVs will grow from less than USD 150 million 1990 to more than USD 1 billion by the turn of the century, this does not mean that the UAV's future is without any challenge. On the contrary, after finishing the conceptual design program of a high altitude long endurance UAV, we find there is still a lot of challenges ahead requiring more research. The following are some of the topics worthy of further attention.

### **5.3.1 Integration of engine simulation program into aircraft design program**

As stated previously, the engine data map is a property of engine company. It is very difficult to acquire, but in order to estimate the aircraft performance, such as climb, cruise etc., an engine data map is indispensable for calculation.

Within our program, the engine data is separately simulated by the TURBOMATCH program. What we need to do is first, estimating the thrust the aircraft needs during cruise, then search relevant engine parameters, such as engine by pass ratio, mass flow rate, total pressure ratio etc. Second, substituting these parameters into engine simulated program and finally the engine data map can be acquired. Third, incorporating this engine data into our program, the aircraft performance can be estimated.

The above process is very complicated and it takes a lot of time to perform. However, both the conceptual aircraft design program and engine simulation program are existing programs. If we can integrate the engine program into aircraft design program, such as by using Visual Basic, the advantage is once the engine net thrust available is not big enough to support the aircraft cruise requirement, there is no need to alter mission profile or replaced by another engine, simply we just need to change engine parameters. Therefore, a lot of time can be saved, and a fully integrated aircraft design program can be achieved.

### **5.3.2 Oswald efficiency**

As stated in chap. 5.2.3, the Oswald efficiency is hard to estimate, especially when the aspect ratio is higher than 20. In the previous calculation, for high aspect ratio UAV, the Oswald efficiency plays an important role at wing lift drag coefficient.

In the previous example, we took 0.8 as Oswald efficiency rather than 0.78 for Tier-II Plus, the result is around 2 percent of fuel can be saved. Therefore, if we can find a method by which the value of Oswald efficiency for high aspect ratio aircraft can be effectively increased, the wing lift drag will reduce, the total drag can be reduced, and lift drag ratio will increase, making the achievement of long endurance much easier.

### **5.3.3 Elasticity of the wing**

Reference 74 stated that the wing tip of Condor bends down 16 ft when static and up 25 ft under 2g for a total excursion between the two limits of 41 ft. In this aspect the bending moment at wing root is very high.

Reference 76 also stated that both dynamic aeroelasticity such as flutter and the static aeroelasticity such as divergence will be improved by increasing of taper ratio and

reduction of aspect ratio. Unfortunately, considering high altitude long endurance UAVs, most of them have high aspect ratio wings. Therefore, the aeroelastic stability is an important topic to be considered.

#### **5.3.4 High Aspect ratio and low Reynolds number data**

As stated in section 5.2.4, the high altitude long endurance UAVs fly at an altitude around 65,000 ft. If the speed is within the subsonic range, the Reynolds number will be very low. Unfortunately, the information for values of aspect ratio higher than 10, and Reynolds number lower than 2 million on ESDU data sheets and in most of the text books is very limited. Therefore, if more accurate results need to be produced, then the data for both low Reynolds number and high aspect ratio wings needs to be established.

#### **5.3.5 Aircraft weight estimation**

As stated in section 5.2.6, most of the initial aircraft weight estimation methods presented are for ordinary aircraft, none of them mentioned how to estimate the aircraft weight of high altitude long endurance UAVs. Therefore, a method which is designed for estimating the initial aircraft weight of high altitude long endurance UAVs is needed.

#### **5.3.6 Stealth techniques**

Many modern weapon systems, and most air defence systems depends for their accuracy on exploiting the signature of targets. If an aircraft is to fly undetected, then all its potential signatures must be reduced as much as possible.

Though loss of unmanned air vehicles will not lead to human injury, a UAV with the state of art technology, may cost as much as a modern manned aircraft. In addition to this, during a war the battle field commander needs real time battle intelligence, thus any employed UAV being shot down may jeopardise the whole battle information collection, and even worse may lead to loss of the war. Therefore, as with manned aircraft, state of the art stealth technology is needed to design a high altitude long endurance UAV, providing it with the ability to penetrate the high risk area to collect the most useful real time enemy information.

Above are presented six topics as future research directions. However, these are only a very small portion of the possible areas compared to the whole design program. With the research going on, we believe more and more problems will turn up, also requiring addressing.

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# Appendices

## Appendix A

### Aircraft conceptual design methodology

In this appendix, we will go through each step of the conceptual design and list the detailed calculations.

#### A.1 UAV Chart

| Aircraft    | Lockheed<br>U-2 | Lockheed<br>TR-1 | Beriev<br>A-40 | Teledyne<br>YQM-98A | Boeing<br>YQM-94A |
|-------------|-----------------|------------------|----------------|---------------------|-------------------|
| Wto(lbs)    | 17270           | 40000            | 189595         | 14310               | 17220             |
| We (lbs)    | N/A             | N/A              | N/A            | 5600                | 5500              |
| S (ft*ft)   | 565             | 1000             | 2152.8         | 347                 | 485               |
| AR          | 10.2            | 10.6             | 8.6            | 16.5                | 16.7              |
| b (ft)      | 80              | 103              | 136.6          | 81.2                | 90                |
| W/S         | 30.57           | 40               | 88.07          | 41.24               | 30                |
| P-load lbs  | 3000            | N/A              | 14330          | 700                 | 700               |
| V (Knots)   | 399             | 373              | 388            | M 0.5-0.6           | M 0.5-0.6         |
| Height (ft) | 70000           | 70000            | 20000          | 50000-70000         | 60000             |
| Thrust (lb) | 11000           | 17000            | 26455 x2       | 4050                | 6000              |
| T/W         | 0.637           | 0.425            | 0.279          | 0.283               | 0.4167            |

Table A.1.1 High altitude aircraft and UAVs chart

\* Teledyne Ryan YQM-98A (Garret ATF3 YF104-GA-100 turbofan engine),

\* Boeing B-gull YQM- 94A (GE TF34-GE-100 turbofan engine)

In table A.1.1, the information of high altitude reconnaissance manned and unmanned aircraft collected from references 62, 63, 64 and 65 are presented (All the aircraft choose turbine engines as their propulsion system).

## A.2 Parametric study

When an aircraft is newly designed, the relevant aircraft data is a good reference to use. In table A.1.1, different kinds high altitude reconnaissance aircraft, manned and unmanned, are presented. From this data,  $C_{d0}$ ,  $C_d$  and  $C_l$  may be estimated.  $L/D$  may be obtained as well.

According to reference 1, eq. 3.20, that the zero-lift drag coefficient,  $C_{d0}$ , can be expressed as:

$$C_{d0} = \frac{f}{S} \quad (\text{A.2.1})$$

Where  $f$  is the equivalent parasite area and  $S$  is the wing area.

Reference 1, eq. 3.21, relates the equivalent parasite area to wetted area,  $S_{wet}$ .

$$\log_{10} f = a + b \log_{10} S_{wet} \quad (\text{A.2.2})$$

Reference 1, eq. 3.22, relates the wetted area to airplanes take-off weight.

$$\log_{10} S_{wet} = c + d \log_{10} W_{TO} \quad (\text{A.2.3})$$

In order to obtain the value of  $S_{wet}$ , the constants  $c$  and  $d$ , in eq. A.2.3, should be given first. Reference 1, table 3.5, divided airplanes into 12 categories. We assume the high altitude reconnaissance aircraft is the same category as military patrol, bomb and transport. Therefore, the constants  $c$  and  $d$  are obtained, provided that the airplane take-off weight is given, then  $S_{wet}$  may be calculated.

Substituting the value of  $S_{wet}$  into reference 1, fig 3.21c, then the value of  $C_f$  is known. Again we substitute the value of  $C_f$  reference 1, table 3.4, then the values of  $a$ ,  $b$  are obtained. Then, substituting the value of  $a$ ,  $b$  and  $S_{wet}$  into eq. A.2.2, the value of  $f$  is obtained.

If the value of wing area,  $S$ , is given, substituting this values into eq. A.2.1, then  $C_{d0}$  can be obtained.

If the aircraft weight, cruise altitude and speed are given, from reference 15, eq. 15.1,  $C_l$  can be calculated.

$$W = L = \frac{1}{2} \rho S C_l V^2 \quad (\text{A.2.4})$$

Substituting  $C_l$  and  $C_{d0}$  into reference 1, eq. 3.19, then the value of  $C_d$  is obtained.

$$C_d = C_{d0} + \frac{C_l^2}{\pi A Re} \quad (\text{A.2.5})$$

The value of  $L/D$  is equal to the value of  $C_l$  divided by the value of  $C_d$ . Therefore, the value of  $L/D$  can be obtained. Following is an example, Teledyne Ryan YQM-98A, used for calculation.

#### Step 1: Calculating the value of $C_{d0}$

From Reference 1, Tab 3.5(10), with eq. A.2.2 and eq. A.2.3. The following aircraft  $C_{d0}$ , in table A.1.1, is calculated.

First, the value of  $c$  is 0.1628,  $d$  is 0.7316.

For Ryan YQM-98A, assuming at 60,000 ft the aircraft weight,  $W_{to}$ , is 14000 lbs, wing area,  $S$ , is 347 ft<sup>2</sup>.

Substituting these values into eq. A.2.3, then the value of  $\log S_{wet}$  is 3.203 ( $S_{wet}$  is 1595.88).

Substituting the value of  $S_{wet}$  into reference 1, Fig 3.21c, the value of  $C_f$  is 0.0040.

Substituting the value of  $C_f$  into reference 1, Table 3.4, the value of  $a$  and  $b$  are obtained as -2.3979 and 1.0 respectively.

Finally, from eq. A.2.2, after calculation, the value of  $\log f = 0.806$ , finally  $f$  is 6.3979.

Substituting these values into eq. A.2.1, then the value of  $C_{d0}$  is 0.0184.

#### Step 2: Calculate aircraft lift coefficient, $C_l$

The aircraft lift coefficient can be calculated by using eq. A.2.4.

From table A.1.1, Teledyne Ryan YQM-98A cruises at 60,000 ft, the density is 0.0002256 slug/ft<sup>3</sup>, and its cruise speed is 0.55M.

Assuming the initial and final weight of Teledyne Ryan YQM-98A during cruise is 14000 and 8000 lbs respectively. Substituting these variables into eq. A.2.4, then the value of lift of coefficient is 1.26 and 0.72 respectively.

#### Step 3: Calculate aircraft drag coefficient, $C_d$

The aircraft drag coefficient can be calculated by eq. A.2.5.

From table A.1.1, the aspect ratio of YQM-98A is 16.5, and  $C_{d0}$  is 0.0184. In reference 15, fig 11.8, the maximum aspect ratio is 12. So, from reference 66, the following equation can be used to estimate Oswald efficiency.

$$K = \frac{1}{e} = \left( 1 + 0.16 + 0.003AR + \frac{0.1}{(AR + 4)^{0.8}} \right) \quad (A.2.6)$$

After calculation the Oswald efficiency is 0.82. For conservative reason, we take 0.8 as its value.

Substituting these values into eq. A.2.5, then the value of drag coefficient is 0.0585.

#### Step 4 : Calculate the value of lift drag ratio, $L/D$



Lift drag ratio can be calculated by using eq. A.2.7.

$$L/D = \frac{C_l}{C_d} \quad (A.2.7)$$

From step 2 and 3, the lift coefficient of YQM-98A is 1.26 and 0.72, the drag coefficient is 0.0567. and 0.031 respectively. Substituting these variables into eq. A.2.7, then the value of lift drag ratio is 22.2 and 23.2 respectively.

### A.3 Initial aircraft sizing

Initial aircraft sizing includes estimating wing area, take-off thrust ( $T_{TO}$ ), climb requirement, and cruise requirement etc., so that we can make sure an airplane is designed to meet our requirement.

Before doing initial aircraft sizing, from the results of section A.2, the similar aircraft parametric study, the value of zero lift-drag coefficient as 0.0175, aspect ratio as 15, and Oswald efficiency as 0.8, of HALE are assumed respectively.

#### A.3.1 Sizing to stall velocity requirement

According to Reference 1, chap 3, power-off stall speed of an airplane may be calculated by:

$$V_s = \left( \frac{2(W/S)}{\rho C_{l_{max}}} \right)^{1/2} \quad (A.3.1)$$

Rewrite eq. A.3.1, we get

$$\frac{W}{S} = \frac{\rho C_{l_{max}} V_s^2}{2} \quad (A.3.2)$$

Due to lacking of proper information, we assume the approach velocity of HALE is 110 Knots. And from reference 10, p. 85, the required stall speed is found by division by 1.3, 1.2 or 1.15. For military aircraft this value should no less than 1.2, but we assume the approach velocity is 1.3 times stall speed.

$$V_{app} = 1.3V_s \quad (A.3.3)$$

Substituting the approach velocity into eq. A.3.3, then the stall speed is 84.6 Knots.

From reference 9, p. 253, for plain flap, the maximum lift coefficient during takeoff is from 1.4 to 1.6. For HALE, we assume 1.5 is the maximum lift coefficient.

Substituting these values into eq. A.3.2. We may obtain the value of wing loading during approach,  $(W/S)_{app}$ , is 36.44 psf.

In order to analyze these parameters at the same condition, we have to keep the weight as the same. Since during approach the weight of aircraft is around half of take-off weight, we have to double the approach weight so as to keep the same condition.

When converting into take-off situation,

$$\left(\frac{W}{S}\right)_{TO} = 2\left(\frac{W}{S}\right)_{app} = 72.88 \text{ psf}$$

The wing loading becomes 72.88 psf, compared to the weight at takeoff.

### A.3.2 Take-off requirement:

Sizing take-off distance is given in terms of take-off field length requirement.

For military airplanes the takeoff runway length is dependent on the type of mission and runway length. For HALE, we assume take-off length,  $S_{TOfl}$ , is within 5000 ft.

From reference 1, eq. 3.8, the takeoff distance requirement is

$$S_{TOfl} = 37.5 \frac{\left(\frac{W}{S}\right)_{TO}}{\sigma C_{lmaxTO} \left(\frac{T}{W}\right)_{TO}} = 37.5 TOP \quad (A.3.4)$$

In eq. A.3.4, the TOP is takeoff parameter and its value is equal to the takeoff distance divided by 37.5.  $\sigma$  is density ratio, at sea level its value is 1. During takeoff, from reference 9, table 7.2, the maximum lift coefficient for plane flap is from 1.4 to 1.6. For HALE, during take-off, we assume the maximum lift coefficient is 1.5.

If we assume  $(W/S)_{TO}$  is 20, 30, 40, 50 and  $S_{to fl} \leq 5000$  ft, and we rewrite eq. A.3.4.

$$\left(\frac{T}{W}\right)_{TO} = \frac{\left(\frac{W}{S}\right)_{TO}}{C_{lmaxTO} \sigma TOP} \quad (A.3.5)$$

Substituting these values into eq. A.3.5, and put the results into table A.3.1.

| $(W/S)_{TO}$ | $C_{lmax}$ | $(T/W)_{TO}$ |
|--------------|------------|--------------|
| 20           | 1.5        | 0.1          |
| 30           | 1.5        | 0.15         |
| 40           | 1.5        | 0.2          |
| 50           | 1.5        | 0.25         |

Table A.3.1 Takeoff requirement

### A.3.3. Climb requirement:

In this section the climb speed and climb rate will be calculated. From section A.3, the zero drag coefficient,  $C_{d0}$ , and Oswald efficiency,  $e$ , are assumed. As we did in previous section, assume  $W/S$  is 20, 30, 40, 50. And if the climb rate is to be maximized, then from reference 1, eq. 3.35, the climb speed,  $V_{cl}$ , can be calculated.

$$V_{cl} = \left( \frac{2\left(\frac{W}{S}\right)}{\rho(C_{D0}\pi A Re)^{1/2}} \right)^{1/2} \quad (A.3.6)$$

For eq. A.3.6, the density is assumed at sea-level.

The aspect ratio, AR, is 15. From reference 1, eq. 3.36, then the value of maximum lift drag ratio,  $(L/D)_{\max}$ , can be calculated.

$$\left(\frac{L}{D}\right)_{\max} = \frac{1}{2} \left( \frac{\pi A Re}{C_{D0}} \right)^{\frac{1}{2}} \quad (A.3.7)$$

For HALE, its flight path angles is less than 15 degrees, it's not like fighter aircraft. If we assume the climb time from sea level to absolute ceiling height 65000 ft is 150 minutes, then from Reference 1, eq. 3.33, the rate of climb, RC, can be calculated.

$$RC = \frac{h_{abs}}{t_{cl}} \ln \left( 1 - \frac{h}{h_{abs}} \right)^{-1} \quad (A.3.8)$$

Finally, substituting these data into reference 1, eq. 3.34, the value of T/W, during climb, is obtained.

$$\frac{T}{W} = \frac{RC}{V_{cl}} + \frac{1}{(L/D)_{\max}} \quad (A.3.9)$$

The results are listed in table A.3.2.

| W/S | RC (fps) | V <sub>cl</sub> | L/D  | T/W   |
|-----|----------|-----------------|------|-------|
| 20  | 18.52    | 144             | 23.2 | 0.172 |
| 30  | 18.52    | 176.3           | 23.2 | 0.148 |
| 40  | 18.52    | 203.6           | 23.2 | 0.134 |
| 50  | 18.52    | 227.6           | 23.2 | 0.124 |

Table A.3.2 Climb requirement

### A.3.4 Cruise requirement:

According to Reference 1, eq. 3.59, can be used to calculate the thrust required to maintain level flight at high altitude. For HALE, provided it flies at 60,000 ft, cruise speed is 0.55 M. While the values of  $C_{D0}$ ,  $e$ , AR and W/S are given in section A.3 and A.3.2.

$$\left(\frac{T}{W}\right)_{CR} = C_{D0} q \left(\frac{S}{W}\right)_{CR} + \left(\frac{W}{S}\right)_{CR} \left(\frac{1}{qAR\pi e}\right) \quad (A.3.10)$$

In eq. A.3.10,  $q$  is the dynamic pressure.

Substituting these values into eq. A.3.10, then the thrust required to maintain level flight is obtained. The results are in table A.3.3 ( $\rho$  is 0.0002236,  $h$  is 60,000 ft).

| $(W/S)_{CR}$ | $Cd_0$ | $(T/W)_{CR}$ |
|--------------|--------|--------------|
| 20           | 0.0175 | 0.0459       |
| 30           | 0.0175 | 0.0432       |
| 40           | 0.0175 | 0.0455       |
| 50           | 0.0175 | 0.05         |

Table A.3.3 cruise requirement

Table A.3.3 is calculated by assuming an altitude of 60,000 ft. If the cruise weight is assumed equal to 0.96 times the take-off weight. And from chap.2, eq.2.8.3, the net thrust at 60,000 ft, is equal to 0.094 times the sea level static net thrust. Thus, we convert the cruise condition into sea level situation, then the results in table A.3.3 will change to the results in table A.3.4.

| W/S  | $Cd_0$ | T/W   |
|------|--------|-------|
| 19.2 | 0.0175 | 0.488 |
| 28.8 | 0.0175 | 0.459 |
| 38.4 | 0.0175 | 0.485 |
| 48   | 0.0175 | 0.532 |

Table A.3.4 Cruise requirement (convert into sea level)

Besides, refer to the previous example of Teledyne Ryan YQM-98A, the lift coefficient,  $C_{lcr}$ , during cruise is assumed around 1 (from eq. A.2.5, if  $C_l$  is 1, then the lift drag ratio is 22.7). Although lift coefficient is based on assumptions, this figure will be checked later. The wing loading during cruise is;

$$(w/s)_{cr} = qC_{lcr} = 31.716 \text{ psf}$$

When converting into sea level condition,

$$(w/s)_{SL} = \frac{31.716}{0.96} = 33 \text{ psf}$$

### A.3.5 Results

After calculation, the results from section A.3.1 to A.3.4 are presented at table A.3.5.

| $(W/S)_{TO}$ | $(T/W)_{TO}$ | $(T/W)_{cl}$ | $(T/W)_{cr}$ | $(W/S)_{app}$ | $(W/S)_{sl}$ |
|--------------|--------------|--------------|--------------|---------------|--------------|
| 20           | 0.1          | 0.172        | 0.488        | 72.88         | 33           |
| 30           | 0.15         | 0.148        | 0.459        | 72.88         | 33           |
| 40           | 0.2          | 0.134        | 0.485        | 72.88         | 33           |
| 50           | 0.25         | 0.124        | 0.532        | 72.88         | 33           |

Table A.3.5 Wing loading and thrust weight ratio

In table A.3.5, the value of thrust weight ratio during take-off, climb and cruise with respect to wing loading at take off is presented (the value of thrust weight ratio during cruise is much higher than the value of thrust weight ratio during take-off and climb). With following pseudo program, the value of optimum matching point for sizing of HALE UAV can be solved.

```

(W/S)cr = 33
(W/S)1 = 25    !assume a wing loading value less than 31.716
(T/W)1 = 0.55  !assume a engine thrust weight ratio, bigger than any value
do I = 0 , 10
  (T/W)cr = -0.000005011*((W/S)1 + I)3 + 0.0007173295*((W/S)1 + I)2
            - 0.0291603535*((W/S)1 + I) + 0.8234545455
  if ((T/W) < (T/W)1) then
    (T/W)1 = (T/W)
  else
    (W/S)2 = (W/S)1 + I
    go to 1
  end if
end do
1  if ((W/S)2 < (W/S)cr) then
  T/W = (T/W)cr
  W/S = (W/S)2
else
  (T/W)cr = -0.000005011*((W/S)cr)3 + 0.0007173295*((W/S)cr)2
            - 0.0291603535*((W/S)cr) + 0.8234545455
  T/W = (T/W)cr
  W/S = (W/S)cr
end if

```

## A.4 Estimate take-off weight, $W_{TO}$

According to reference 8, eq. 5.1, the takeoff weight can be broken down as:

$$W_{TO} = W_{fuel} + W_{pay} + W_{emp} \quad (A.4.1)$$

In eq. A.4.1,  $W_{fuel}$  is the mission fuel weight plus reserved mission fuel weight, its value is obtained from chapter 2.4.2.  $W_{pay}$  is the payload weight needed for carrying out a mission, and its value is given from mission requirement. The only unknown parameter is  $W_{emp}$ , the empty weight of the aircraft which includes structure, propulsion, avionics, instruments, etc., and will be calculated in this section.

Before calculating the empty weight of aircraft, the aircraft layout, such as wing area, fuselage length, tailplane area etc., has to be given first.

### A.4.1 Layout considerations and initial sizing of components

According to reference 2, chap. 6, the configuration of wing planform and initial sizing of aircraft components may be decided.

As we said in chap. 2.5.1, the aircraft component sizes are not fixed until the final aircraft design is finished. The following aircraft component sizes are just an example how to calculate each component size during iteration process.

In this example, the aircraft take-off weight is assumed to be 10,000 lbs. According to this weight, we calculate fuel weight, and each aircraft component size and weight, and finally add together. If the difference of the final take-off weight minus the first assumed weight is less than 3 percent, then we accept. Otherwise, we will add or subtract 200 lbs to the first take-off weight, and iterate, until it meets our requirement. When running this program to estimate the aircraft take-off weight, after 1 loop, the assumed weight is 10,200 lbs and the estimated weight is 10,489.7 lbs. The difference is 2.8 percent which less than 3 percent. Therefore, the aircraft component size is based on the take-off weight, 10,200 lbs, to estimate.

#### A.4.1.1 Wing layout

First, an aircraft with conventional cantilever wing and the wing-body combination, low wing configuration, has been selected. From the result in chapter 2.4.1, wing loading,  $W/S$ , is 30. Thus, if the take off weight,  $W_{TO}$ , is 10200. lbs, then the wing area,  $S$ , can be calculated by:

$$S = \frac{W_{TO}}{W/S} \quad (A.4.2)$$

Then the wing area is 340.  $\text{ft}^2$ .

The aspect ratio, AR, is 15, when we substitute these values into eq. A.4.3, then the value of wing span can be obtained.

$$AR = \frac{b^2}{S} \quad (A.4.3)$$

then the wing span, b, is 71.41 ft.

During preliminary design, the basic airfoil data is assumed as below :

- Sweep angle of quarter chord line,  $\Lambda_{c/4}$ , is 0.
- The wing taper ratio,  $\lambda$ , is 0.5.
- The thickness ratio of root section,  $(t/c)_r$

From reference 9, p:236, the thickness ratio of wing root section,  $(t/c)_r$ , is from 15 percent to 20 percent. It can provide adequate room for fuel and retracting the undercarriage.

- The thickness ratio for tip section,  $(t/c)_t$

The thickness ratio for tip section,  $(t/c)_t$ , is from 10 percent to 15 percent, this reduction proportion is good for lower structure weight.

Therefore we select the wing root and wing tip thickness ratio as below :

$$(t/c)_r = 0.17$$

$$(t/c)_t = 0.13$$

In order to obtain higher maximum lift coefficient,  $C_{l \max}$ , in subsonic flight. From reference 28, the MS(1)-series airfoil can generate high lift coefficient in subsonic flight. Thus, we select the MS(1)-0317 airfoil for wing root, and MS(1)-0313 airfoil for wing tip.

According to reference 9, eq. A.7, for a straight tapered wing the root chord can be calculated as

$$C_r = \frac{2}{1+\lambda} \left( \frac{S}{AR} \right)^{\frac{1}{2}} \quad (A.4.4)$$

Substituting these values into eq. A.4.4, the length of wing root chord,  $C_r$ , is 6.35 ft.

From reference 9, eq. 7.28, the length of wing tip can be calculated as

$$C_t = \lambda C_r \quad (A.4.5)$$

So, the length of wing tip is 3.175 ft.

From reference 9, eq. A.15, the Mean Aerodynamic Chord, MAC, for straight tapered wing can be calculated.

$$MAC = \frac{2}{3} C_r \frac{1+\lambda+\lambda^2}{1+\lambda} \quad (A.4.6)$$

Substituting these values of  $C_r$  and  $\lambda$  into eq. A.4.6, then the value of MAC is 4.94 ft.

#### A.4.1.2 Wing fuel tank volume

According to reference 9, eq. B.12, the total tank volume available in a wing can be calculated.

$$V_{wf} = 0.54 \frac{S^2}{b} \left( \frac{t}{c} \right)_r \frac{1 + \lambda \sqrt{\tau} + \lambda^2 \tau}{(1 + \lambda)^2} \quad (\text{A.4.7})$$

There are 5 variables in eq. A.4.7, only the ratio of thickness ratio at tip and root,  $\tau$  is unknown. From reference 9, p:449,  $\tau$  can be calculated by:

$$\tau = \frac{\left( \frac{t}{c} \right)_t}{\left( \frac{t}{c} \right)_r} \quad (\text{A.4.8})$$

From eq. A.4.8, the value of  $\tau$  is 0.765. Substituting these values into eq. A.4.7, the wing fuel tank volume is 107.55 ft<sup>3</sup>. Assume the fuel for turbofan engine is JP-4, from reference 14, p:536, the specific weight of JP-4 is 49 lb/ft<sup>3</sup>. Thus, the fuel weight in wing fuel tank is 5269.95 lbs.

Comparing the fuel weight in wing fuel tank with total mission fuel weight,  $W_{\text{fuel}}$ , from chapter 2.4.2, 5946.24 lbs. Obviously the wing fuel tank is not big enough, less than 676.29 lbs, to carry the total mission fuel. Thus, we have to put some fuel into fuselage tank, about 13.8 ft<sup>3</sup> fuselage fuel tank.

#### A.4.1.3 Aileron sizing

Comparing the aileron sizing with the similar mission aircraft, YQM - 98A and YQM - 94A, we select aileron size of HALE as in table A.4.1.

| Aircraft   | Aileron to wing semispan | $\eta_i$ | $\eta_o$ | Aileron to wing chord |
|------------|--------------------------|----------|----------|-----------------------|
| YQM - 98 A | 0.23                     | 0.43     | 0.66     | ~0.25                 |
| YQM - 94 A | 0.32                     | 0.51     | 0.83     | ~0.2                  |
| HALE       | 0.30                     | 0.50     | 0.80     | ~0.3                  |

Table A.4.1 Aileron sizing

#### A.4.1.4 The fuselage length

From reference 12 , the actual aircraft length and fuselage length are presented as;

| Aircraft  | Length inches | fuselage length inches |
|-----------|---------------|------------------------|
| YQM - 98A | 460           | 448                    |
| YQM - 94A | 600           | 600                    |

Table A.4.2 Aircraft length



From reference 10, table 6.3, the fuselage length is relevant to aircraft take off weight, it can be estimated by eq. A.4.9.

$$L = aW_{To}^c \tag{A.4.9}$$

In eq. A.4.9, L is the length of the aircraft, and  $W_{To}$  is the aircraft take-off gross weight. For general aviation - single engine aircraft, the value of the coefficient a is 4.37 and c is 0.23.

Substituting these values into eq. A.28, after calculation the values of YQM - 98A and YQM - 94A aircraft fuselage length are 473.64 and 494.16 inches respectively. Comparing with the fuselage length in table A.4.2. The YQM - 98A aircraft fuselage length is longer by 25.64 inches and YQM - 94A aircraft fuselage length is smaller by 105.84 inches. From above calculation, the differences are around 9 ft. Obviously, this formula is not suitable for such aircraft.

Therefore, we have to use another method. By comparing with the proportion of the length of wingspan and fuselage of YQM - 98A and YQM - 94A, and put the results into table A.4.3.

| aircraft | wingspan inches | fuselage inches | proportion |
|----------|-----------------|-----------------|------------|
| YQM-98A  | 974.4           | 448             | 2.175      |
| YQM-94A  | 1080            | 600             | 1.8        |

Table A.4.3 The proportion of wingspan and fuselage

From table A.4.3, the value of proportion is from 1.8 to 2.175. By taking the average, the proportion value is around 2. Therefore we assume the proportion of the length of wingspan and fuselage for HALE is 2. Thus, because the length of wingspan is 71.41 ft, when dividing by 2, then the length of HALE fuselage is 428.48 inches.

#### A.4.1.5 Estimating the size of horizontal and vertical tailplane

For the initial layout, a historical approach is a good estimation for the size of horizontal and vertical tailplane. According to reference 9, table 6.4, the horizontal and vertical tail volume coefficient of sailplane and general aviation-single engine airplane, are from 0.5 to 0.7 and from 0.02 to 0.04. And from reference 9, eq. 6.26 and eq. 6.27, the vertical and horizontal tail volume coefficient can be calculated.

$$c_v = \frac{L_v S_v}{b S} \tag{A.4.10}$$

$$c_H = \frac{L_H S_H}{MAC * S} \tag{A.4.11}$$

In eq. A.4.10 and A.4.11, the moment arm,  $L_V$  and  $L_H$ , is the distance from the horizontal and vertical tailplane quarter chord to the wing quarter chord.  $S_V$  and  $S_H$  are the area of vertical and horizontal tailplane.  $b$  is wingspan,  $S$  is wing area, and  $MAC$  is mean aerodynamic chord.

By analyzing YQM-98A, the moment arm of  $L_V$  and  $L_H$  are 216 inches and 183.6 inches respectively. The wing area is 49968 inch<sup>2</sup>, wingspan is 786 inches,  $MAC$  is 51.24 inches, horizontal and vertical tailplane area are 7929 and 4032 inch<sup>2</sup> respectively.

Substituting these variables into eq. A.4.10 and A.4.11, after calculation, the values of the vertical,  $c_V$ , and horizontal tail volume coefficient,  $c_H$ , are 0.022 and 0.569.

Since this is initial layout design, when comparing these values, we select 0.026 and 0.55 as the value of vertical and horizontal tail volume coefficient of HALE.

For HALE, the dimension of wingspan, wing area and  $MAC$  are given from previous section. Therefore, in order to calculate the area of horizontal and vertical tailplane, there are two variables,  $L_V$  and  $L_H$  still unknown.

From reference 12, we obtained the proportion of the horizontal and vertical tail arm length to the fuselage length, listed in table A.4.4 and A.4.5.

| Aircraft | Fuselage inches | H-tail arm inches | Proportion     |
|----------|-----------------|-------------------|----------------|
| YQM- 98A | 488.8           | 183.6             | 37.5 percent   |
| YQM- 94A | 600             | ~ 250             | ~ 41.7 percent |

Table A.4.4 Proportion of the horizontal-tail arm length to the fuselage length

| Aircraft | Fuselage inches | V-tail arm inches | Proportion     |
|----------|-----------------|-------------------|----------------|
| YQM- 98A | 488.8           | 196.5             | 40.2 percent   |
| YQM- 94A | 600             | ~ 260             | ~ 43.3 percent |

Table A.4.5 Proportion of the vertical-tail arm length to the fuselage length

From table A.4.4, the proportion of the horizontal-tail arm length to the fuselage length of similar aircraft is from 37.5 to about 41.7 percent. For HALE, in initial sizing phase, the value of 40 percent is selected. From A.4.1.4, the length of HALE fuselage is 428.48 inches. Thus, the horizontal tail moment arm,  $L_H$ , is 171.39 inches.

From table A.11, the proportion of the vertical-tail arm length to the fuselage length of similar aircraft is from 40.2 to about 43.3 percent. For HALE, in initial sizing phase, the value of 42 percent is selected. Thus, the vertical tail moment arm,  $L_V$ , is 179.88 inches.

Substituting these variables into eq. A.4.10 and A.4.11, after calculating, the area values of horizontal and vertical tailplane are 9308.45 and 6061.68 inch<sup>2</sup> respectively. The following is to calculate the horizontal and vertical tail span and aspect ratio. From reference 12, the proportion of wingspan to horizontal tailplane of YQM-98A is 3.8. Thus, we take 3.9 as the proportion of wingspan to horizontal tailplane of HALE. Since the wingspan is 71.41 ft, so the horizontal tailplane span is 18.31 ft. And from eq. A.4.3, the tailplane aspect ratio is 5.187. The horizontal tailplane taper ratio of HALE is 1, that means the tailplane root chord and tip chord is the same. Thus, from eq. A.4.5, the value of tailplane root chord is 3.53 ft, and from eq. A.4.6, the horizontal tailplane MAC is 3.53 ft.

Since the number of vertical tailplane is 2, so the area of each vertical tailplane is 21.05 ft<sup>2</sup>. We assume the vertical tailplane taper ratio is 0.5, leading edge sweep angle is 30 degrees, and the root chord is 1.25 multiply by the length of horizontal tailplane chord, so it is 4.41 ft. Then from eq. A.4.5, the value of tip chord is 2.2 ft. And from eq. A.4.6, the vertical tailplane MAC is 3.43 ft.

## A.5 Weight estimation

According to reference 10, chap. 15, in preliminary design, the weight estimation method of General-aviation aircraft has been chosen. Here we have to calculate each component-weight and finally add them together, and obtain the total aircraft gross weight,  $W_{TO}$ .

Therefore, in this program, first assume a take-off weight, then by using General-aviation method calculate each component weight. Sum together, then compare with the take-off weight. If the difference is within a certain limit, then it is accepted. Otherwise, assume a new take-off weight, and repeat calculation. The following is the pseudo program to calculate take-off weight.

$W_{TO} = 10000.$

do  $I = 0, 20$

$W_{TO} = W_{TO} + I * 200.$

Initial aircraft component sizing

- a. Calculate wing weight
- b. Calculate horizontal tailplane weight
- c. Calculate vertical tailplane weight
- d. Calculate fuselage weight
- e. Calculate main landing gear weight
- f. Calculate nose landing gear weight

```

g. Calculate install engine weight
h. Calculate fuel system weight
i. Calculate fixed equipment weight
 $W_{TO1} = a.+b.+c.+d.+e.+f.+g.+h.+i.$ 
If ( ABS(  $W_{TO1} - W_{TO}$  )/ $W_{TO}$  .LT. 0.03) then
     $W_{TO} = W_{TO1}$ 
    go out do loop
else
    continue
end if
end do

```

Before computation, some parameters, such as wing area,  $S$ , wingspan,  $b$ , taper ratio,  $\lambda$ , weight of fuel in wing,  $W_{fw}$ , and aspect ratio,  $AR$ , horizontal and vertical tailplane area,  $S_H$ ,  $S_V$ , horizontal tail arm length,  $L_t$ , horizontal and vertical tailplane taper ratio,  $\lambda_h$ ,  $\lambda_v$ , vertical tailplane leading edge sweep angle,  $\Lambda_v$ , and fuselage length,  $L$ , has to be given, and in previous section these values are being estimated.

Wing average thickness ratio,  $t/c$ , is 0.15, wing sweep at quarter M.A.C.,  $\Lambda$ , is 0 degrees, design gross weight,  $W_{TO}$ , is around 10,200 lbs, limit landing load factor,  $N_l$ , is 2, landing design gross weight,  $W_l$ , is 6000 lbs, the length of both main landing gear,  $L_m$ , and nose landing gear,  $L_N$ , are 3 ft, engine weight,  $W_{en}$ , is 1000 lbs, number of engines,  $N_{en}$ , is 1, integral tanks volume,  $V_i$ , is 804.5 gals, total mission fuel volume,  $V_t$ , is 907.83 gals, number of fuel tanks,  $N_t$ , is 3, maximum cruise range to operational area,  $R$ , is 500 nautical miles, the value of  $H_i/H$ , for conventional tail is 0.

Apart from above parameters, still there are two variables, fuselage wetted area,  $S_f$ , and limit load factor,  $N_z$ , need to be solved.

According to reference 42, table 4, Mr. Baullinger select 2.5 as limit load factor for his aircraft. In order to select a proper limit load factor for HALE, we chose at the beginning of cruise, the altitude is 50,000 ft, Mach No. is 0.55. In terms of safety and not increase too much weight of aircraft structure. A value of 3 as limit load factor, at 20,000 ft has been chosen.

The reason to select 20,000 ft for limit load factor test is following reference 67, the gust velocity of 66 ft/s between sea level and 20,000 ft, thereafter decreasing linearly to 38 ft/sec at 50,000 ft being constant thereafter. Therefore, at 20,000 ft the effect of gust is the greatest. According to reference 66, it suggests the calculation should be done at the condition of both aircraft first climb through 20,000 ft and descent through 20,000 ft when the mission is finished.

In order to make sure whether 3 is a proper figure and the structure will not suffer damage when aircraft encounters a gust, there are two different formulas to verify.

**a. According to reference 10, p:339**

If the aircraft is in subsonic flight, then following formulae can be used to calculate gust loads.

$$\mu = \frac{2 * \left( \frac{W}{S} \right)}{\rho * g * mac * C_{L\alpha}} \quad (A.5.1)$$

$$K = \frac{0.88 * \mu}{5.3 + \mu} \quad (A.5.2)$$

$$U = K * U_{de} \quad (A.5.3)$$

$$\Delta L = \frac{1}{2} * \rho * V_T * S * C_{L\alpha} * U \quad (A.5.4)$$

$$\Delta n = \frac{\Delta L}{W} \quad (A.5.5)$$

$$N_z = 1 + \Delta n \quad (A.5.6)$$

Where K is gust alleviation factor,  $\mu$  is mass ratio, U is gust velocity,  $U_{de}$  is the standard vertical gust velocity,  $\Delta L$  is the change in aircraft lift,  $\Delta n$ , is the change in load factor, and  $N_z$  is the limit load factor.

**a.1 Climb through 20,000 ft**

Assuming the aircraft weight, W, is 9,800 lbs, density,  $\rho$ , at 20,000 ft is 0.0012668, aircraft lift curve slope,  $C_{L\alpha}$ , is 6.01, aircraft true velocity,  $V_T$ , is 463 fps, mean aerodynamic chord is 4.89 ft, aircraft wing area, S, is 333.33 ft<sup>2</sup>, standard vertical gust,  $U_{de}$ , is 30 fps, and then replace these value into eq A.5.1 to eq A.5.6. After computation, the value of mass ratio,  $\mu$ , is 49, gust alleviation factor, K, is 0.794, upward gust velocity, U, is 23.8 fps, the change in aircraft lift,  $\Delta L$ , is 13996.5 lbf, the change in load factor,  $\Delta n$ , is 1.43, and the limit load factor,  $N_z$ , is 2.45.

**a.2 Descend through 20,000 ft**

Assuming the aircraft weight is 5,000 lbs, aircraft velocity and lift curve slope is the same. After calculation the value of  $\mu$  is 25, K is 0.725, U is 21.77,  $\Delta n$  is 2.56. So, the limit load factor is 3.56.

**b. According to reference 27, p:399**

The gust load factor is calculated as follows:

$$N_z = 1 + \frac{\rho * V_e * U_{de} * C_{L\alpha}}{2 * (W/S)} \quad (A.5.7)$$

**b.1 Climb through 20,000 ft**

Assume aircraft weight,  $W$ , density,  $\rho$ , and aircraft lift curve slope,  $C_{L\alpha}$ , is the same as in a., equivalent aircraft velocity,  $V_e$ , is 338 fps. Substituting these values into eq. A.5.7, the value of the limit load factor is 2.31.

**b.2 Descend through 20,000 ft**

Assume aircraft weight is 5,000 lbs, aircraft velocity is the same. After calculation the value of the limit load factor is 3.57.

Comparing the results of limit load factor from a. and b., we find that when aircraft climb through 20000 ft, in both method, the limit load factor is far less than 3. When the aircraft descend through 20000 ft, the limit load factor is higher than 3. Since the critical situation is at climb segment, because the weight at climbing is much higher than descend, we can select the figure 3 as limit load factor and should be a reasonable value.

From reference 9, eq. B.6, the fuselage wetted area may be calculated.

$$S_f = \pi DL \left(1 - \frac{2}{\lambda_f}\right)^{\frac{2}{3}} \left(1 + \frac{1}{\lambda_f^2}\right) \quad (A.5.8)$$

In eq. A.5.8,  $D$  is the aircraft diameter, 3 ft,  $L$  is the fuselage length, from section A.4.1.4, the length is 35.7 ft, and  $\lambda_f$  is fineness ratio, its value, from eq. A.5.9, is 11.9.

$$\lambda_f = \frac{L}{D} \quad (A.5.9)$$

Substituting these values into eq. A.5.8, then the fuselage wetted area is 299.77 ft<sup>2</sup>.

Finally assume the HALE is cruising from 60000 ft, and the cruise speed is 0.55 M. Thus, the dynamic pressure,  $q$ , is 31.716 lb/ft<sup>2</sup>. Following is the aircraft weight estimation.

**A.5.1 Wing group**

$$\begin{aligned} W_{wing} &= 0.036 S^{0.758} W_{fw}^{0.0035} \left(\frac{AR}{\cos^2 \Lambda}\right)^{0.6} q^{0.06} \lambda^{0.04} \left(\frac{100 l/c}{\cos \Lambda}\right)^{-0.3} (N_z W_{To})^{0.49} \quad (A.5.10) \\ &= 1309.35 \text{ lbs} \end{aligned}$$

### A.5.2 Horizontal tailplane group

$$W_{HT} = 0.016\lambda_h^{-0.02} \left( \frac{AR}{\cos^2 \Lambda_H} \right)^{0.043} q^{0.168} S_H^{0.896} \left( \frac{100 t/c}{\cos \Lambda} \right)^{-0.12} (N_Z W_{TO})^{0.414} \quad (A.5.11)$$

$$= 69.998 \text{ lbs}$$

### A.5.3 Vertical tailplane group

$$W_{VT} = 0.073 \left( 1 + \frac{0.2H_t}{H_v} \right) (N_Z W_{TO})^{0.376} q^{0.122} S_V^{0.873} \left( \frac{100 t/c}{\cos \Lambda_V} \right)^{-0.49} \left( \frac{AR}{\cos^2 \Lambda_V} \right)^{0.357} \lambda_v^{0.039} \quad (A.5.12)$$

$$= 54.21 \text{ lbs}$$

### A.5.4 Fuselage group

$$W_{fuse} = 0.052 S_f^{1.086} (N_Z W_{TO})^{0.177} L_t^{-0.051} \left( \frac{L}{D} \right)^{-0.072} q^{0.241} + W_{pres} \quad (A.5.13)$$

$$= 266.22 \text{ lbs}$$

### A.5.5 Main landing gear

$$W_{ml} = 0.095 (N_l W_l)^{0.768} \left( \frac{L_m}{12} \right)^{0.409} \quad (A.5.14)$$

$$= 202.15 \text{ lbs}$$

### A.5.6 Nose landing gear

$$W_{nl} = 0.125 (N_l W_l)^{0.566} \left( \frac{L_n}{12} \right)^{0.845} \quad (A.5.15)$$

$$= 61.63 \text{ lbs}$$

### A.5.7 Install engine

$$W_{ien} = 2.575 W_{en}^{0.922} N_{en} \quad (A.5.16)$$

$$= 1502.37 \text{ lbs}$$

### A.5.8 Fuel system

$$W_{fue} = 2.49 V_t^{0.726} \left( \frac{1}{1 + \frac{V_i}{V_t}} \right)^{0.363} N_t^{0.242} N_{en}^{0.157} \quad (A.5.17)$$

$$= 386.17 \text{ lbs}$$

### A.5.9 Flight control system

$$\begin{aligned} W_{fli} &= 0.053L^{1.536}b^{0.371}(N_ZW_{TO}10^{-4})^{0.8} \\ &= 153.33 \text{ lbs} \end{aligned} \quad (\text{A.5.18})$$

### A.5.10 Hydraulic system

$$\begin{aligned} W_{hyd} &= 0.001W_{TO} \\ &= 10.2 \text{ lbs} \end{aligned} \quad (\text{A.5.19})$$

### A.5.11 Miscellaneous (Avionics, electronics and instruments)

$$\begin{aligned} W_{mis.} &= 0.575W_{emp}^{0.556}R^{0.25} \\ &= 327.82 \text{ lbs} \end{aligned} \quad (\text{A.5.20})$$

### A.5.12 The aircraft empty weight

$$\begin{aligned} W_{emp} &= W_{wing} + W_{HT} + W_{VT} + W_{fuse} + W_{ml} + W_{nl} + W_{ien} + W_{fue} + W_{fli} + W_{hyd} + W_{mis} \quad (\text{A.5.21}) \\ &= 4343.467 \text{ lbs} \end{aligned}$$

The aircraft take-off weight now becomes:

$$\begin{aligned} W_{TO1} &= W_{emp} + W_{pay} + W_{fuel} \\ &= 10489.71 \text{ lbs} \end{aligned} \quad (\text{A.5.22})$$

Substituting this amount of weight into eq. A.5.23.

$$dif = \frac{W_{TO1} - W_{TO}}{W_{TO}} \quad (\text{A.5.23})$$

The value of dif is 2.8 percent. Obviously, the value is meet the requirement of less than 3 percent. Thus, a new aircraft take-off weight,  $W_{TO}$  as 10,489.71 lbs has to be reassumed.



## A.6 Balance calculation

From the previous section, each aircraft component weight is obtained. Reference 1, chap 2, gives the following as the aircraft C.G. movement calculation and aircraft landing gear balance calculation.

As stated before, the aircraft take-off weight is not fixed until the final design is finished. Thus, each aircraft component weight is not fixed either. The following aircraft components C.G. location provides us example for calculation.

### A.6.1 Aircraft C.G. movement

The aircraft weight breakdown has been listed in table 2.5.2. According to reference 2, chap 10, each component C.G. can be estimated as follows ( the reference axis is 50 inches ahead of aircraft, and the pitot tube length is 12 inches ahead of aircraft nose ):

#### A. Fuselage :

Since the fuselage type of the HALE is like a airliner, then the C.G location in the fuselage is assumed equal to 0.39 times the fuselage length.

$$CG_{fuse} = 0.39L \quad (A.6.1)$$

Since the fuselage length,  $L$ , is 428.486 inches, from eq. A.6.1, the fuselage C.G.,  $CG_{fuse}$ , is 167.1 inches. Plus the distance from the reference axis, the fuselage C.G., is 229.1 inches.

#### B. Wing :

In preliminary design phase, the range of Wing C.G,  $CG_w$ , is assumed between 0.37 to 0.42 mean aerodynamic chord, MAC, for HALE, the wing C.G. location is assumed equal to 0.4 multiply by MAC.

$$CG_w = 0.4MAC \quad (A.6.2)$$

From chap. 2.5.1.1, the mean aerodynamic chord, MAC, is 59.244 inches, so the wing C.G.,  $CG_w$ , is 23.697 inches

For the aircraft, the wing C.G. should include the length from the leading edge of wing M.A.C. to the leading edge of wing root, 6.7 inches, the length from the leading edge of wing root to the nose of aircraft and the distance to reference axis.

The distance from the leading edge of wing root to the nose of aircraft is determined from similar aircraft ( from YQM-98A, the proportion of the fuselage length to the distance from aircraft nose to the leading edge of wing root is 2.65). For HALE, the proportion of the fuselage length to the distance from aircraft nose to the leading edge of wing root is assumed to be 2.6. Thus, the distance from aircraft nose to wing root

leading edge is 164.8 inches. Therefore, the whole aircraft wing C.G. is

$$CG_w = 23.697 + 6.7 + 164.8 + 62 = 257.199 \text{ inches}$$

### C. Horizontal tailplane :

Since the profile of horizontal tailplane is rectangular, the C.G. location is assumed equal to 0.5 multiplied by the horizontal tailplane mean aerodynamic chord.

$$CG_{htail} = 0.5MAC_{htail} \quad (A.6.3)$$

From chap.2.5.1.5, the horizontal tailplane MAC is 42.36 inches, therefore the horizontal tailplane C.G. location is

$$CG_{htail} = 21.18 \text{ inches}$$

The distance from reference axis to  $C.G_{tail}$  is calculated from the reference axis (62 inches), plus the distance from aircraft nose to aircraft wing root (164.8 inches), plus the  $\frac{1}{4}$  wing root chord (19.04 inches), plus the horizontal tail arm length (171.38 inches) plus the horizontal tailplane C.G. location (21.18 inches), then minus the  $\frac{1}{4}$  horizontal tailplane root chord (10.59 inches). Therefore, the horizontal tailplane C.G. is 427.83 inches.

Since the fuselage length, L, is 428.48 inches, including the distance of reference axis, 62 inches, the distance becomes 490.48 inches. The horizontal tailplane location is from 417.24 to 438.42 inches. Thus, the horizontal tailplane is mounted on the tail fuselage.

### D. Vertical tailplane :

The profile of vertical tailplane is the same as wing profile. Therefore, the C.G. location of vertical tailplane is assumed the same as wing, equal to 0.4 multiplied by the vertical tailplane mean aerodynamic chord.

$$CG_{vtail} = 0.4MAC_{vtail} \quad (A.6.4)$$

From chap.2.5.1.5, the vertical tailplane MAC is 41.16 inches, so the horizontal tailplane C.G. location is

$$CG_{vtail} = 16.464 \text{ inches}$$

The distance from reference axis to  $C.G_{vtail}$  is calculated from the reference axis (62 inches), plus the distance from aircraft nose to aircraft wing root (164.8 inches), plus the  $\frac{1}{4}$  wing root chord (19.04 inches), plus the vertical tail arm length (179.96 inches) plus the vertical tailplane C.G. location (16.464 inches), then minus the  $\frac{1}{4}$  vertical tailplane root chord (13.23 inches). Therefore, the vertical tailplane C.G. is 429.046 inches.

### E. Engine :

The C.G. location of engine is approximately 40 percent of the entire engine length.

$$CG_{eng} = 0.4L_{eng} \quad (A.6.5)$$

Since engine length is 108 inches, so,

$$CG_{eng} = 43.2 \text{ inches}$$

For the whole aircraft, the engine C.G. ( $CG_{eng}$ ) location is equal to engine C.G., plus the distance from aircraft nose to engine inlet (equal to the distance from aircraft nose to wing root minus 1 foot, 152.8 inches) and plus the distance from reference axis (62 inches) to aircraft nose.

$$CG_{eng} = 258 \text{ inches}$$

#### **F. Nose landing gear & Main landing gears' C.G. location:**

Assuming the nose landing gear C.G. location is at  $\frac{1}{4}$  fuselage length. Thus, the C.G. is

$$CG_{nl} = 62 + \frac{1}{4}L \quad (A.6.6)$$

$$= 169.1 \text{ inches}$$

Assume the main landing gear C.G. location is behind the nose landing gear C.G. about 30 percent fuselage length.

$$CG_{ml} = CG_{nl} + 0.3L \quad (A.6.7)$$

$$= 297.66 \text{ inches}$$

#### **G. Fixed equipment :**

Assuming the C.G. location of fixed equipment is around 26 percent fuselage length. Thus, The C.G. location of fixed equipment is plus the distance from reference axis to aircraft nose.

$$CG_{fix} = 0.26L + 62 \quad (A.6.8)$$

$$= 173.384 \text{ inches}$$

#### **H. Payload C.G. :**

Assuming the C.G. location of payload is 3 ft ahead of  $\frac{1}{2}$  aircraft fuselage length. Thus, the payload C.G. location plus the distance from reference axis to the aircraft nose is:

$$CG_{pay} = \frac{1}{2}L - 36 + 62 \quad (A.6.9)$$

$$= 240.2 \text{ inches}$$

#### **I. Fuel system C.G. location :**

For straight unswept wing, and mainly the fuel is put into the wing. Thus, the fuel system C.G. location may be assumed the same as wing C.G. location.

$$CG_{fuel} = CG_w \quad (A.6.10)$$

When the location of each component C.G. and each component weight has been obtained, the next step is to check whether the aircraft C.G. movement within the

length of wing MAC.

First we put each component weight in one column, and put each component C.G. location into another column, then multiply each component weight by each component C.G. location, and put the results into another column.

After calculation, the results are listed into table A.6.1.

| TYPE OF COMPONENT    | Wi (lbs) | Xi (in) | Wi*Xi       |
|----------------------|----------|---------|-------------|
| Fuselage group       | 266.22   | 229.1   | 60991.002   |
| Wing group           | 1309.35  | 257.199 | 336763.5107 |
| Horizontal tailplane | 69.998   | 427.83  | 29947.24434 |
| Vertical tailplane   | 54.21    | 429.046 | 23258.58366 |
| Engine group         | 1502.37  | 258     | 387611.46   |
| Nose landing gear    | 61.63    | 169.1   | 10421.633   |
| Main landing gear    | 202.15   | 297.66  | 60171.969   |
| Fixed equipment      | 491.35   | 173.384 | 85192.2284  |
| Fuel weight          | 5946.244 | 257.199 | 1529368.011 |
| Fuel tank weight     | 386.17   | 257.199 | 99322.53783 |
| Payload              | 200      | 240.2   | 48040       |
| TAKE OFF WEIGHT      | 10489.7  |         |             |

Table A.6.1 Component weight and C.G.

According to reference 2, chap 10, it suggests that we should calculate all possible loading scenarios of the movement of aircraft C.G. For HALE, in order to calculate the aircraft C.G. movement, we take four different situations. The following is our calculation.

**a. The aircraft C.G. of empty weight,  $E_W$**

$$\begin{aligned}
 Var1 &= W_W CG_W + W_{HT} CG_{HT} + W_{VT} CG_{VT} + W_{fuse} CG_{fuse} + W_{nl} CG_{nl} \\
 Var2 &= W_{ml} CG_{ml} + W_{ien} CG_{eng} + W_{fue} CG_{fue} + W_{fix} CG_{fix} \\
 Var3 &= W_W + W_{HT} + W_{VT} + W_{fuse} + W_{nl} + W_{ml} + W_{ien} + W_{fue} + W_{fix} \\
 E_W &= \frac{Var1 + Var2}{Var3}
 \end{aligned} \tag{A.6.10}$$

**b. The aircraft C.G. of empty weight with payload,  $E_P$**

$$E_P = \frac{Var1 + Var2 + W_{pay} CG_{pay}}{Var3 + W_{pay}} \tag{A.6.11}$$

**c. The aircraft C.G. of empty weight with fuel,  $T_W$**

$$E_F = \frac{Var1 + Var2 + W_{fuel} CG_{fuel}}{Var3 + W_{fuel}} \tag{A.6.12}$$

**d. The aircraft C.G. of the take-off weight,  $T_P$**

$$T_P = \frac{Var1 + Var2 + W_{fuel}CG_{fuel} + W_{pay}CG_{pay}}{Var3 + W_{fuel} + W_{pay}} \quad (A.6.13)$$

After calculation the following result is obtained.

- a. Empty weight,  $E_w$ , the aircraft C.G. is 251.8 inches
- b. Empty weight with payload,  $E_P$ , the aircraft C.G. is 251.29 inches
- c. Empty weight with fuel,  $E_F$ , the aircraft C.G. is 254.92 inches
- d. Take-off weight,  $T_P$ , the aircraft C.G. is 254.64 inches

According to reference 9, table 8.16, for jet engine airplanes, the C.G. movement should around 10 to 40 percent of wing MAC. Since the range of wing MAC is form 233.46 to 292.74 inches. Thus, from above calculation, the C.G. movement is from 251.29 to 254.92 inches, is within 30 to 36 percent of wing MAC. Obviously, the aircraft C.G. movement is within limit.

### **A.6.2 Landing gear balance calculation**

According to reference 66, the aircraft nose landing gear and main landing gear are sharing one tenth and nine tenth of aircraft weight respectively. Thus, in order to make sure that the landing gear meets the requirement, we have to do some calculation. The following is the pseudo program listing for such a calculation.

```
D1 =  $T_w - CG_{nl}$     !D1 is the distance between A/C aft. CG and nose landing gear.
D2 =  $CG_{ml} - T_w$     !D2 is the distance between A/C aft. CG and main landing gear.
V1 = D1
V2 = D2
M1 =  $W_{To} * 0.1 * (D1 + D2)$  !Moment of A/C weight on nose landing gear and
                             the distance of D1+D2
M2 =  $W_{To} * D2$        !Moment of aircraft weight and the distance of D2
if ( M1 .GT. M2 ) then
  do i = 0, 40
    D1 = V1 - i        !Keep the distance of D1+D2 the same, and move
    D2 = V2 + i        !the nose and main landing gear backward
    M1 =  $W_{To} * 0.1 * (D1 + D2)$ 
    M2 =  $W_{To} * D2$ 
    if ((M1 - M2) .GT. 0 ) then
      continue
    else
      go to 1
```

```

end if
end do
else
do i = 0, 40
D1 = V1 + i    !Keep the distance of D1+D2 the same, and move
D2 = V2 - i    !the nose and main landing gear forward
M1 = WTO * 0.1 * ( D1 + D2)
M2 = WTO * D2
if (( M1 - M2 ) .GT. 0 ) then
continue
else
go to 1
end if
end do
end if
1   CGnl = Tw - D1
    CGml = Tw + D2

```

After calculation, the nose and main landing gear move forward from previous 169.12 and 297.67 inches to a new location 138.12 and 266.67 inches.

Since the weight of nose and main landing gear compared to the take-off gross weight is very small, their effect to whole aircraft CG movement is trivial. Therefore, although the nose and main landing gear location are moving forward, we didn't recalculate the whole aircraft CG movement.

## A.7 Wing body incidence

During cruise, according to requirements, the speed of HALE is 532.44 fps ( $M$  is 0.55). From reference 27, p: 212-213, it states that as long as the air flow over the airfoil remains entirely subsonic, the free stream,  $M_\infty$ , on airfoil can be affected by a factor, the Prandtl-Glauert compressibility factor,  $\beta$ .

$$\beta = (1 - M^2)^{\frac{1}{2}} \quad (\text{A.7.1})$$

In eq. A.7.1, the aircraft velocity,  $M$ , is 0.55. Substituting it into eq. A.7.1, the value of  $\beta$  is 0.835.

Before calculating the wing body incidence, the basic information about wing and tailplane should be computed first.

A.7.1 Wing airfoil

From chap. 2.5.1.1, the airfoil of wing root is MS(1)-0317 and wing tip is MS(1)-0313, in order to generating more lift in subsonic flight.

According to reference 28, we put both airfoil data into table A.7.1.

|        | MS-0317   |           |        |          | MS-0313   |           |        |          |
|--------|-----------|-----------|--------|----------|-----------|-----------|--------|----------|
| x/c    | z/c upper | z/c lower | camber | thickens | z/c upper | z/c lower | camber | thickens |
| 0      | 0.0010    | 0.0010    | 0.0010 | 0.0000   | 0.0010    | 0.0009    | 0.0010 | 4E-05    |
| 0.002  | 0.0125    | -0.0086   | 0.0020 | 0.0211   | 0.0095    | -0.0068   | 0.0014 | 0.0162   |
| 0.0125 | 0.0310    | -0.0211   | 0.0050 | 0.0520   | 0.0243    | -0.0152   | 0.0045 | 0.0395   |
| 0.025  | 0.0432    | -0.0287   | 0.0073 | 0.0719   | 0.0345    | -0.0206   | 0.0069 | 0.0550   |
| 0.0375 | 0.0521    | -0.0342   | 0.0089 | 0.0863   | 0.0419    | -0.0244   | 0.0087 | 0.0663   |
| 0.05   | 0.0589    | -0.0387   | 0.0101 | 0.0976   | 0.0474    | -0.0275   | 0.0100 | 0.0749   |
| 0.075  | 0.0684    | -0.0454   | 0.0115 | 0.1138   | 0.0552    | -0.0322   | 0.0115 | 0.0874   |
| 0.1    | 0.0751    | -0.0506   | 0.0123 | 0.1257   | 0.0606    | -0.0361   | 0.0123 | 0.0967   |
| 0.125  | 0.0803    | -0.0548   | 0.0128 | 0.1351   | 0.0648    | -0.0392   | 0.0128 | 0.1040   |
| 0.15   | 0.0845    | -0.0582   | 0.0132 | 0.1427   | 0.0682    | -0.0418   | 0.0132 | 0.1100   |
| 0.175  | 0.0881    | -0.0610   | 0.0135 | 0.1490   | 0.0710    | -0.0440   | 0.0135 | 0.1150   |
| 0.2    | 0.0910    | -0.0633   | 0.0138 | 0.1543   | 0.0733    | -0.0458   | 0.0137 | 0.1191   |
| 0.225  | 0.0934    | -0.0653   | 0.0141 | 0.1587   | 0.0752    | -0.0473   | 0.0140 | 0.1225   |
| 0.25   | 0.0954    | -0.0669   | 0.0143 | 0.1622   | 0.0768    | -0.0485   | 0.0142 | 0.1252   |
| 0.275  | 0.0969    | -0.0681   | 0.0144 | 0.1651   | 0.0780    | -0.0494   | 0.0143 | 0.1274   |
| 0.3    | 0.0982    | -0.0691   | 0.0145 | 0.1672   | 0.0790    | -0.0501   | 0.0144 | 0.1291   |
| 0.325  | 0.0990    | -0.0698   | 0.0146 | 0.1688   | 0.0797    | -0.0502   | 0.0147 | 0.1299   |
| 0.35   | 0.0995    | -0.0702   | 0.0147 | 0.1697   | 0.0801    | -0.0505   | 0.0148 | 0.1306   |
| 0.375  | 0.0997    | -0.0704   | 0.0147 | 0.1701   | 0.0803    | -0.0511   | 0.0146 | 0.1314   |
| 0.4    | 0.0996    | -0.0702   | 0.0147 | 0.1698   | 0.0802    | -0.0509   | 0.0146 | 0.1311   |
| 0.425  | 0.0991    | -0.0697   | 0.0147 | 0.1688   | 0.0800    | -0.0505   | 0.0148 | 0.1305   |
| 0.45   | 0.0983    | -0.0688   | 0.0147 | 0.1671   | 0.0795    | -0.0498   | 0.0149 | 0.1293   |
| 0.475  | 0.0970    | -0.0676   | 0.0147 | 0.1646   | 0.0788    | -0.0488   | 0.0150 | 0.1276   |
| 0.5    | 0.0954    | -0.0659   | 0.0147 | 0.1613   | 0.0778    | -0.0475   | 0.0151 | 0.1253   |
| 0.525  | 0.0932    | -0.0639   | 0.0147 | 0.1571   | 0.0765    | -0.0459   | 0.0153 | 0.1224   |
| 0.55   | 0.0907    | -0.0614   | 0.0147 | 0.1521   | 0.0749    | -0.0440   | 0.0154 | 0.1189   |
| 0.575  | 0.0878    | -0.0585   | 0.0147 | 0.1462   | 0.0729    | -0.0418   | 0.0156 | 0.1147   |
| 0.6    | 0.0845    | -0.0550   | 0.0147 | 0.1395   | 0.0706    | -0.0393   | 0.0157 | 0.1099   |
| 0.625  | 0.0808    | -0.0511   | 0.0149 | 0.1319   | 0.0680    | -0.0364   | 0.0158 | 0.1044   |
| 0.65   | 0.0767    | -0.0467   | 0.0150 | 0.1235   | 0.0649    | -0.0333   | 0.0158 | 0.0982   |

|       |         |         |         |         |         |         |         |         |
|-------|---------|---------|---------|---------|---------|---------|---------|---------|
| 0.675 | 0.0727  | -0.0421 | 0.0153  | 0.1149  | 0.0615  | -0.0300 | 0.0157  | 0.0915  |
| 0.7   | 0.0676  | -0.0374 | 0.0151  | 0.1050  | 0.0577  | -0.0266 | 0.0156  | 0.0843  |
| 0.725 | 0.0627  | -0.0326 | 0.0151  | 0.0952  | 0.0537  | -0.0231 | 0.0153  | 0.0768  |
| 0.75  | 0.0576  | -0.0278 | 0.0149  | 0.0854  | 0.0494  | -0.0196 | 0.0149  | 0.0690  |
| 0.775 | 0.0523  | -0.0231 | 0.0146  | 0.0753  | 0.0449  | -0.0161 | 0.0144  | 0.0610  |
| 0.8   | 0.0469  | -0.0186 | 0.0142  | 0.0654  | 0.0402  | -0.0129 | 0.0136  | 0.0531  |
| 0.825 | 0.0413  | -0.0143 | 0.0135  | 0.0557  | 0.0353  | -0.0099 | 0.0127  | 0.0452  |
| 0.85  | 0.0358  | -0.0105 | 0.0126  | 0.0463  | 0.0303  | -0.0073 | 0.0115  | 0.0375  |
| 0.875 | 0.0301  | -0.0072 | 0.0115  | 0.0373  | 0.0252  | -0.0051 | 0.0100  | 0.0303  |
| 0.9   | 0.0244  | -0.0046 | 0.0099  | 0.0290  | 0.0201  | -0.0035 | 0.0083  | 0.0236  |
| 0.925 | 0.0187  | -0.0029 | 0.0079  | 0.0216  | 0.0150  | -0.0026 | 0.0062  | 0.0176  |
| 0.95  | 0.0130  | -0.0023 | 0.0054  | 0.0153  | 0.0096  | -0.0029 | 0.0033  | 0.0125  |
| 0.975 | 0.0072  | -0.0032 | 0.0020  | 0.0104  | 0.0047  | -0.0035 | 0.0006  | 0.0082  |
| 0.99  | 0.00365 | -0.0046 | -0.0005 | 0.00828 | 0.00158 | -0.0049 | -0.0017 | 0.00648 |
| 1     | 0.0013  | -0.0060 | -0.0024 | 0.0072  | -0.0005 | -0.0061 | -0.0033 | 0.0056  |

Table A.7.1 The airfoil data of MS(1)-0317 and MS(1)-0313

Following the definition in reference 19, fig 1 and 3, we can estimate both the inner and outer trailing edge angle,  $\tau$  and  $\tau_a$  (illustrated in fig A.7.1)

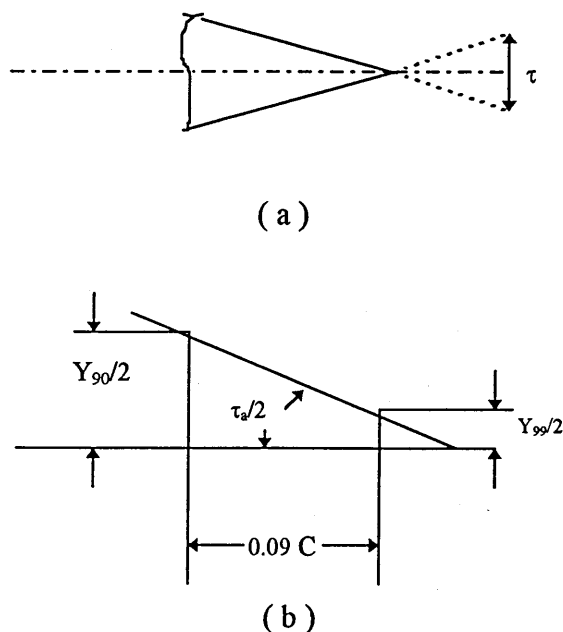
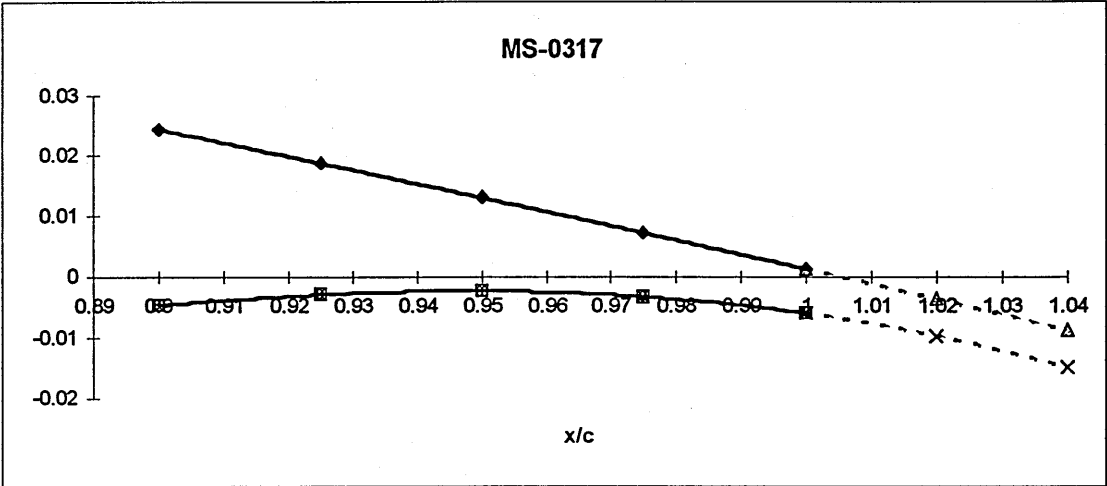


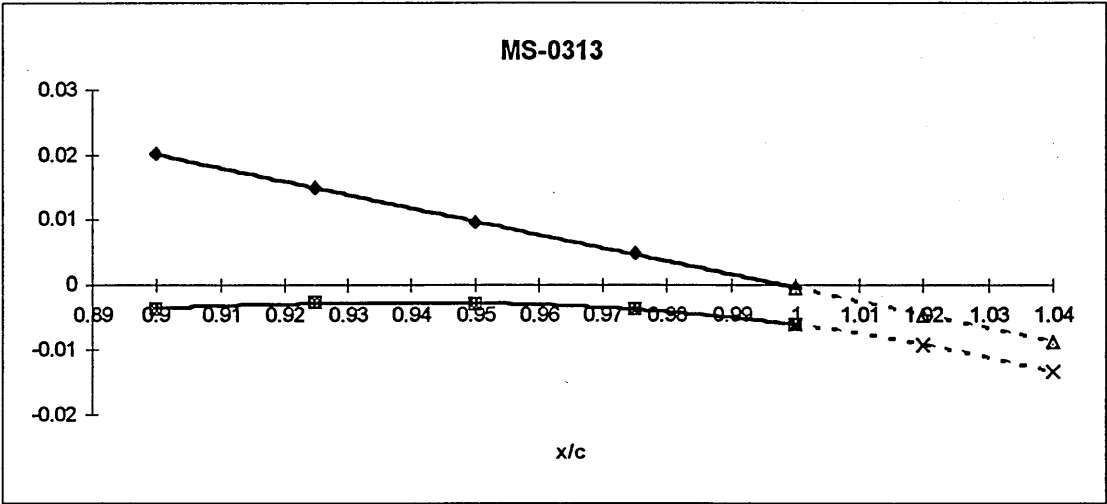
Fig A.7.1 The illustration of outer (a) and inner (b) trailing edge angle



Based on the airfoil data listed in table A.7.1, and following the definition which is illustrated in fig. A.7.1. The rear part, from 90 to 100 percent airfoil chord, of MS(1)-0317 and MS(1)-0313 airfoil is illustrated in fig A.7.2 (a), (b), and the inner and outer trailing edge angle,  $\tau$  and  $\tau_a$  of MS(1)-0317 and MS(1)-0313 airfoil may be obtained from fig A.7.2 and eq. A.7.2.



(a)



(b)

Fig A.7.2 The rear part of airfoil, (a).MS-0317, (b).MS-0313

From the data listed in table C.4, the value of  $(y_{90} - y_{99})$  of MS-0317 and MS-0313 airfoil section are 0.021 and 0.0171 percent chord respectively. Substituting this data into eq. A.7.2

$$\tan\left(\frac{1}{2} \tau_a\right) = \frac{y_{90} - y_{99}}{0.18} \tag{A.7.2}$$

After calculation :

$$\tan\left(\frac{1}{2}\tau_a\right) = 0.1153, \text{ for MS-0317 airfoil}$$

$$\tan\left(\frac{1}{2}\tau_a\right) = 0.095, \text{ for MS-0313 airfoil}$$

The airfoil shape is different at wing root and wing tip. In order to simplify the calculation and get a reasonable figure, the author decided to take the mean inner trailing edge value. After calculation, the mean inner trailing edge is

$$\tan\left(\frac{1}{2}\tau_a\right) = 0.1052$$

From trailing edge airfoil shape, listed in fig. A.7.2. The outer trailing edge angle,  $\tau$ , of MS-0317 and MS-0313 airfoil are 0 degrees.

The thickness ratio of airfoil at wing root is 17 percent wing chord, and at wing tip is 13 percent wing chord. In conceptual design, we decided to take average value of wing root and wing tip. Thus, the wing thickness ratio,  $\frac{t}{c}$ , becomes 15 percent wing chord. When we substitute the above data into eq. A.7.3.

$$C_{l\alpha 0T} = 2\pi + (4.75 + 0.02\tau) * \frac{t}{c} \quad (\text{A.7.3})$$

Then the theoretical airfoil lift curve slope in incompressible flow becomes,  $C_{l\alpha 0T}$ , 6.996.

Some basic airfoil data, such as the inner trailing edge angle,  $\tan\left(\frac{1}{2}\tau_a\right)$ , the outer trailing edge angle,  $\tau$ , and the theoretical airfoil lift curve slope in incompressible flow is constant during our calculation.

Other parameters, such as Reynolds number, airfoil lift curve slope, etc., are varied with speed and altitude. Thus, the following computation is only an example, and we should bear in mind these values will vary when flight condition changes.

From chap. 2.2, table 2.1.1, the cruise speed of HALE is 0.55 M. If the altitude is 60000 ft, then the Reynolds number on airfoil is

$$R = \frac{\rho V l}{\mu} \quad (\text{A.7.4})$$

Since at 60000 ft, the density,  $\rho$ , is 0.000224 slugs/ft<sup>3</sup>,  $l$  is the characteristic length. For a wing,  $l$  is mean aerodynamic chord length which equals to 4.937 ft and the dynamic viscosity,  $\mu$ , is 0.000000297 lb-s/ft<sup>2</sup>.

Substituting these variables into eq. A.7.4, the wing Reynolds number is 1,982,550.

According to reference 29, fig 4, if the aircraft leading edge sweep angle is less than 10 degrees, then the transition from laminar to turbulent flow will occur when Reynolds number reaches 12 million. For HALE, its leading edge sweep angle is 2 degrees and

during cruise the Reynolds number at wing of HALE is  $1.98 \times 10^6$ , which is far lower than 10 million. Thus, the air flow on the airfoil can maintain laminar flow until the airfoil maximum thickness point. From table A.7.1, it is easy to find in both MS-0317 and MS-0313 airfoil, the thickness point is at 0.375 percent chord length. That means the chordwise location of boundary layer transition ( $x_t/c$ ) point at wing is 0.375.

From reference 19, eq. 1, may be used.

$$\frac{(C_{l\alpha})_{atM=0}}{C_{l\alpha 0T}} = 1 - \frac{0.1 + (1.05 - 0.5 * \frac{x_t}{c}) * \tan(\frac{1}{2} * \tau_a)}{(\log_{10} R - 5)^{(1 - 2.5 * \tan \frac{1}{2} * \tau_a)}} \quad (A.7.5)$$

$(C_{l\alpha})_{atM=0}$  is the airfoil lift curve slope in incompressible flow. Substituting these variables into eq. A.7.5, the result of the airfoil lift curve slope in incompressible flow divided by the theoretical airfoil lift curve slope in incompressible flow is 0.8424.

From eq. A.7.3, the value of the theoretical airfoil lift curve slope in incompressible flow is given, Thus, from eq. A.7.6, the airfoil lift curve slope in incompressible flow may be calculated by:

$$(C_{l\alpha})_{atM=0} = C_{l\alpha 0T} * \frac{(C_{l\alpha})_{atM=0}}{C_{l\alpha 0T}} \quad (A.7.6)$$

Substituting these variables into eq. A.7.6, then the value of the airfoil lift curve slope in incompressible flow is 5.893.

After modified by Prandle-Glauert factor, by using eq. A.7.7, the airfoil lift curve slope in compressible flow may be calculated.

$$(C_{l\alpha})_{atM} = \frac{(C_{l\alpha})_{atM=0}}{\beta} \quad (A.7.7)$$

After calculation, the airfoil lift curve slope in compressible flow is 7.056.

From reference 6, eq. 8.25, the constant K may be calculated.

$$K = \frac{\frac{(C_{l\alpha})_{atM}}{2 * \pi}}{\beta} \quad (A.7.8)$$

After calculation, the value of K is 0.9379.

From reference 6, eq. 8.22, the whole wing lift curve slope may be calculated.

$$C_{L\alpha w} = \frac{2 * \pi * AR}{2 + \left( \left( \frac{AR^2 * \beta^2}{K^2} \right) \left( 1 + \frac{1}{\beta^2} \tan^2 \Lambda_{\frac{\alpha}{2}} \right) + 4 \right)^{\frac{1}{2}}} \quad (A.7.9)$$

After calculation, the whole wing lift curve slope is 6.078.

From reference 6, eq. 8.46, 8.47 and 8.48, these values of constant  $K_A$ ,  $K_\lambda$ ,  $K_h$  may be calculated.

$$k_A = \left( \frac{1}{AR} \right) - \frac{1}{1 + AR^{1.7}} \quad (\text{A.7.10})$$

$$k_\lambda = \frac{10 - 3\lambda}{7} \quad (\text{A.7.11})$$

$$k_h = \frac{1 - \frac{h_h}{b}}{\left( \frac{2l_h}{b} \right)^{1/3}} \quad (\text{A.7.12})$$

Substituting these variables into eq. A.7.10, A.7.11 and A.7.12, then these values of  $K_A$ ,  $K_\lambda$  and  $K_h$  are 0.05675, 1.214 and 1.334 respectively.

From reference 6, eq. 8.45 may be used to calculate the down wash gradient at the horizontal tail.

$$\frac{d\varepsilon}{d\alpha} = 4.44 * \left( \left( k_A k_\lambda k_h \left( \cos \Lambda_{\frac{1}{4}} \right)^{1/2} \right)^{1.19} \right) \frac{(C_{L\alpha w})_{atM}}{(C_{L\alpha w})_{atM=0}} \quad (\text{A.7.13})$$

There are 5 variables in eq. A.7.13, and only one is still unknown, the wing lift curve slope in incompressible flow. From eq. A.7.9, substituting the variable  $\beta$  with 1, and the value of whole wing lift curve slope in incompressible flow,  $(C_{L\alpha w})_{atM=0}$ , becomes 5.021.

Substituting these variables into eq. A.7.13, then the value of the down wash gradient at the horizontal tail is 0.303.

## A.7.2 Tailplane

The tailplane of HALE, according to reference 9, p:323 to 339, suggesting the frequently use is made of approximately symmetrical airfoils with a thickness ratio of 9 to 12 percent chord and a large nose radius. Thus, the NACA-0012 airfoil as the tailplane of HALE has been chosen. From reference 11, we list the tailplane airfoil data in table A.7.2.

Following the method in previous section, the outer trailing edge angle,  $\tau$ , and the inner trailing edge angle,  $\tau_a$ , can be obtained. From table A.7.2, the value of  $(y_{90} - y_{99})$  of NACA 0012 airfoil section is 0.01183 percent chord length. Substituting these variables into eq. A.7.2, then the inner trailing edge angle,  $\tan(\frac{1}{2}\tau_a)$ , is 0.06575.

The trailing edge airfoil profile, from 90 percent to whole chord length, of NACA 0012, listed in fig. A.7.3. We can obtain the outer trailing edge angle,  $\tau$ , is 10 degrees.

| x/c    | z/c upper | Z/C lower | camber | thickness |
|--------|-----------|-----------|--------|-----------|
| 0      | 0         | 0         | 0      | 0         |
| 0.0125 | 0.00947   | -0.00947  | 0      | 0.01894   |
| 0.025  | 0.013075  | -0.01308  | 0      | 0.02615   |
| 0.05   | 0.017775  | -0.01778  | 0      | 0.03555   |
| 0.075  | 0.021     | -0.021    | 0      | 0.042     |
| 0.1    | 0.023415  | -0.02342  | 0      | 0.04683   |
| 0.15   | 0.026725  | -0.02673  | 0      | 0.05345   |
| 0.2    | 0.028685  | -0.02869  | 0      | 0.05737   |
| 0.25   | 0.029705  | -0.02971  | 0      | 0.05941   |
| 0.3    | 0.03001   | -0.03001  | 0      | 0.06002   |
| 0.4    | 0.029015  | -0.02902  | 0      | 0.05803   |
| 0.5    | 0.02647   | -0.02647  | 0      | 0.05294   |
| 0.6    | 0.022815  | -0.02282  | 0      | 0.04563   |
| 0.7    | 0.01832   | -0.01832  | 0      | 0.03664   |
| 0.8    | 0.013115  | -0.01312  | 0      | 0.02623   |
| 0.9    | 0.00724   | -0.00724  | 0      | 0.01448   |
| 0.95   | 0.004035  | -0.00404  | 0      | 0.00807   |
| 1      | 0.00063   | -0.00063  | 0      | 0.00126   |

Table A.7.2 The NACA 0012 airfoil data

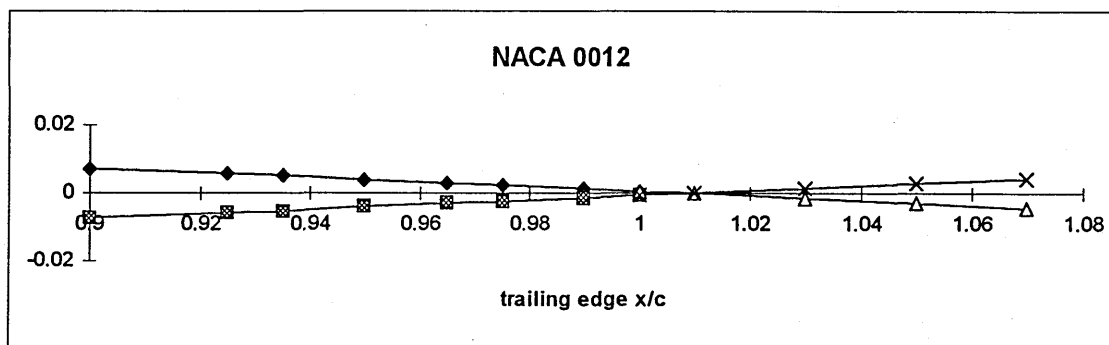


Fig A.7.3 The trailing edge profile of NACA 0012

From reference 11, the thickness ratio of NACA 0012 airfoil is 12 percent chord length. Substituting these values of thickness ratio and outer trailing edge angle into eq. A.7.3, then the theoretical airfoil lift curve slope in incompressible flow,  $C_{l\alpha 0T}$ , is 6.877.

Since the density , the dynamic viscosity and velocity are the same as in previous section, and the tailplane MAC is 3.53 ft.

Substituting these variables into eq. A.7.4, the Reynolds number of tailpane during cruise is 1416500.

The Mach number of air flow on the tailplane is less than 10 million and the leading edge of tailplane is 0 degrees, less than 10 degrees. So, the air flow on tailplane is similar to the flow on the wing, it can maintain laminar flow until it reaches the airfoil maximum thickness point. From table A.7.2, the maximum thickness point is at 0.3 percent chord length. That means the chordwise location of boundary layer transition ,  $x/c$  , point at wing is 0.3. Substituting these variables into eq. A.7.5, then the result of the airfoil lift curve slope in incompressible flow divided by the theoretical airfoil lift curve slope in incompressible flow is 0.8585.

From eq. A.7.6, the tailplane airfoil lift curve slope,  $(C_{l\alpha})_{atM=0}$ , is 5.9.

After modified by Prandle-Grauer factor, from eq. A.7.7, then tailplane airfoil lift curve slope,  $(C_{l\alpha})_{atM}$ , is 7.069.

From eq. A.7.8, the constant  $K_T$  of tailplane, is 0.94.

Substituting these variables into eq. A.7.9, then tailplane lift curve slope,  $C_{L\alpha T}$ , is 4.64.

From reference 6, eq.8.42, the aircraft lift curve slope may be calculated.

$$C_{L\alpha} = C_{L\alpha wf} + C_{L\alpha T} * \eta_h * \left( \frac{S_{HT}}{S} \right) * \left( 1 - \frac{d\varepsilon}{d\alpha} \right) \quad (A.7.14)$$

From the above calculation, in eq. A.7.14, there are two unknown variables,  $\eta_h$  and the wing body lift curve slope,  $C_{L\alpha wf}$ . According to reference 6, eq. 8.37 and 8.43 may be used to solve these two variables. But eq. 8.37, itself has other variables which are not easy to obtain. Since the value of  $\eta_h$  is approach to 1, thus, in preliminary design phase, we assume its value is 0.95. The eq. 8.43 is:

$$C_{L\alpha wf} = K_{wf} C_{L\alpha w} \quad (A.7.15)$$

There is only one variable in eq. A.7.15, and from reference 6, eq. 8.44 may be used.

$$K_{wf} = 1 + 0.025 \frac{D}{b} - 0.25 \left( \frac{D}{b} \right)^2 \quad (A.7.16)$$

From section A.4.1.1 and A.5, the wingspan, b, and fuselage diameter, D, are 70.7 and 3 ft respectively. Substituting these variables into eq. A.7.16, the  $K_{wf}$  is 1.0006.

From eq. A.7.9, the wing lift curve slope,  $C_{L\alpha w}$ , is 6.078. Substituting these variables into eq. A.7.15, then the wing body lift curve slope,  $C_{L\alpha wf}$ , is 6.082.

Substituting these variables into eq. A.7.14, then the value of the aircraft lift curve slope,  $C_{L\alpha}$ , is 6.666.

### A.7.3 Wing body incidence

From reference 9, eq. 7.42, may be used to calculate the wing body incidence angle.

$$i_w = \frac{C_{Lcr}}{C_{L\alpha}} + \alpha_{0l}\epsilon_t + \alpha_{0lr} \quad (A.7.17)$$

As we used an iteration method to calculate the take-off weight in section A.4, we now use the iteration method to calculate wing body incidence angle.

$C_{Lcr}$  is the cruise lift coefficient, initially, we assume its value is 1.  $C_{L\alpha}$  is the aircraft lift curve slope, from eq. A.7.14, its value is 6.666.  $\alpha_{0l}$  is the change in zero-lift angle of the wing per degree of positive twist at the tip. In preliminary design, according to reference 9 pp : 258 its value is approximately -0.4.  $\alpha_{0lr}$  is the zero lift angle of the root section, from reference 28, for MS-0317 and -0313 airfoil, its value is -3 degrees.  $\epsilon_t$  is the wing twist angle, for HALE, at wing tip is -3 degrees.

Substituting these variables into eq. A.7.17, after calculation, then the value of wing body incidence is 6.796 degrees.

As we said earlier, the value of wing body incidence is not fixed yet. When the aircraft performance is calculated, we will go back to recheck the wing body incidence angle.

### A.7.4 Oswald efficiency

Aircraft drag can be computed by a number of methods, for example either using a roughly equivalent drag formula as eq. A.7.18 or using a detailed drag breakdown formula to calculate each component zero lift drag and lift induced drag. One important factor Oswald efficiency,  $e$ , has to be calculated first since.

$$C_D = C_{D0} + \frac{C_L^2}{\pi A Re} \quad (A.7.18)$$

Though many textbooks have introduced how to estimate Oswald efficiency factor, most of them are only valid when aspect ratio less than a certain value, such as reference 6, eq. 4.12, is only valid when aspect ratio is in between 2 to 10.5. Therefore, it is hard to find one method which can really estimate Oswald efficiency when the aircraft aspect ratio is higher than 20. After referring to reference 66, it was suggested a formula, eq. A.2.6, may be used to estimate Oswald efficiency of the high aspect ratio aircraft.

As stated in section A.4.1.1, the aspect ratio of HALE is 15. Substituting its value into eq. A.2.6, after calculation, the Oswald efficiency is 0.823.

Since there is no proof that eq. A.2.6 is accurate for the high aspect ratio aircraft, we tried to find another method mentioned in reference 9 and 27. Before calculation, some factors such as taper ratio,  $\lambda$ , Prandtl-Glauert compressibility factor,  $\beta$ , and aspect ratio, AR, should be known first. These values of Prandtl-Glauert compressibility factor,  $\beta$ , and aspect ratio, AR, taper ratio,  $\lambda$ , has been calculated in sections A.7.1 and A.4.1.1.

From reference 9, eq. F-18, may be used for estimating higher figure of  $\beta \cdot AR$ , range from 6 to 30. From which we may obtain the increment of plane wing vortex induced drag coefficient due to additional lift,  $\delta$ , is

$$\delta = (0.0015 + 0.016(\lambda - 0.4)^2)(\beta AR - 4.5) \quad (A.7.19)$$

When we replace the value of taper ratio,  $\lambda$ , Prandtl-Glauert compressibility factor,  $\beta$ , and aspect ratio, AR, into eq. A.7.19. After calculation, the increment of plane wing vortex induced drag coefficient due to additional lift,  $\delta$ , is 0.0133.

Again from reference 27, p:174

$$e = \frac{1}{1 + \delta + K\pi AR} \quad (A.7.20)$$

Where K is the constant of proportionality giving the rate of  $C_d$  with  $C_l^2$ , for some particular laminar flow airfoil this value is found to be 0.0038. So, after substituting these variables into eq. A.7.20, the Oswald efficiency becomes 0.838.

By using reference 10, eq. 12.49, for straight wing aircraft, the Oswald efficiency is:

$$e = 1.78(1 - 0.045AR^{0.68}) - 0.64 \quad (A.7.21)$$

Since the aspect ratio is 15, after calculation the Oswald efficiency is 0.635. This figure, compare to above estimation, is quite low which is due to the effect of high aspect ratio.

Comparing the above values, it is hard to say which method is good to estimate high aspect ratio Oswald efficiency. From reference 10, p:298, it mentioned that a reasonable value of Oswald efficiency should between 0.7 to 0.85. Finally, a conservative 0.8 is the value has been selected as Oswald efficiency for performing conceptual design.



## A.8 Aircraft component drag estimation

In this paragraph, the following method, listed in reference 6, chap. 4, may be used to estimate aircraft component drag.

### A.8.1 Drag breakdown procedure

According to reference 6, chap. 4, the total airplane drag is written as:

$$D = \frac{1}{2} \rho V^2 S C_D \quad (\text{A.8.1})$$

There are four variables in eq. A.8.1,  $C_D$  is the total airplane drag coefficient,  $\rho$  is the air density,  $V$  is the free stream velocity and  $S$  is the wing area. Three of them have been given in a previous section, only the value of  $C_D$  is unknown.

From reference 6, eq. 4.4, the total airplane drag coefficient can be broken down to:

$$C_D = C_{Dw} + C_{Di} + C_{Dn} + C_{Dl} + C_{Dm} \quad (\text{A.8.2})$$

For HALE, in preliminary design phase, we only dealing with drag coefficient of wing, fuselage, empennage, nacelle and landing gear. The miscellaneous drag coefficient,  $C_{Dm}$  is so small that it can be ignored.

Since the total aircraft drag is fluctuating with aircraft altitude, weight and velocity, it's not easy to list every drag value during the whole flight. In this section, we only chose cruise phase as an example ( assume  $W_{TO}$  is 10,400 lbs, flight velocity,  $M$ , is 0.55, altitude is 60000 ft ) to calculate total aircraft drag.

#### A.8.1.1 Wing drag coefficient prediction

In subsonic flight, according to reference 6, eq. 4.5, the wing drag coefficient can be divided into zero-lift drag and lift drag.

$$C_{Dw} = C_{D0w} + C_{Diw} \quad (\text{A.8.3})$$

##### A.8.1.1.1 Wing drag coefficient due to zero drag

According to reference 6, eq. 4.6, the subsonic wing zero-lift drag coefficient can be calculated as :

$$C_{D0w} = R_{w_f} R_{LS} C_{fw} \left( 1 + L'(\%) + 100(\%)^4 \right) \frac{S_{wetw}}{S} \quad (\text{A.8.4})$$

There are seven parameters in eq. A.8.4. Only wing area,  $S$ , is given in section A.4.1.1.

The other six parameters, such as wing/fuselage interference factor,  $R_{wf}$ , lifting surface correction factor,  $R_{Ls}$ , turbulent flat plate friction coefficient of the wing,  $C_{fw}$ , airfoil thickness location parameter,  $L'$ , wing thickness ratio,  $t/c$ , and wetted area of the wing,  $S_{wetw}$ , should be calculated.

**a. Wing/fuselage interference factor,  $R_{wf}$ .**

From reference 6, fig 4.1, we generate three equations to calculate  $R_{wf}$ . Before calculating, the fuselage Reynolds number should be given.

Assume the cruise altitude is 60000 ft ( $\rho$  is 0.000224 slugs/ft<sup>3</sup>),  $l$  is the characteristic length, for fuselage is 35.7 ft, velocity is 532.44 fps and dynamic viscosity,  $\mu$ , is 0.000000297 lb-s/ft<sup>2</sup>. Substituting these variables into eq. A.7.4, the fuselage Reynolds number,  $R_{Nfus}$ , is 14,336,081.

According to reference 6, fig 4.1, there are 3 different Mach number to calculate wing/fuselage interference factor,  $R_{wf}$ , stated in following pseudo program:

if (M .LT. 0.25) then

$$R_{wf} = -9E - 17R_{Nfus}^2 + 2E - 09R_{Nfus} + 1.0634$$

else if (M .GT. 0.25 and  $R_{Nfus}$  .LT.  $2 \times 10^7$ ) then

$$\alpha = 4.176E - 15M^3 - 7.689E - 15M^2 - 4.578E - 15M - 8.83E - 16$$

$$\beta = 1.6727E - 8M^3 + 3.45327E - 8M^2 - 2.3274E - 8M + 7.5365E - 9$$

$$\gamma = -0.8792M^3 + 1.3681M^2 - 0.8941M + 1.2069$$

$$R_{wf} = \alpha R_{Nfus}^2 + \beta R_{Nfus} + \gamma$$

else if (M .GT. 0.25 and  $R_{Nfus}$  .GT.  $2 \times 10^7$ ) then

$$\alpha = 2.054E - 24M^2 - 3.838E - 24M - 1.511E - 24$$

$$\beta = 4.699E - 16M^2 - 9.07E - 16M + 3.677E - 16$$

$$\gamma = 3.32E - 8M^2 - 6.866E - 8M + 2.926E - 8$$

$$\tau = 0.2258254M^2 - 1.0046005M + 1.5595657$$

$$R_{wf} = \alpha R_{Nfus}^3 + \beta R_{Nfus}^2 + \gamma R_{Nfus} + \tau$$

end if

In this example, Reynolds number is  $1.43 \times 10^7$ , and Mach number is 0.55, greater than 0.25. After calculation, the value of wing/fuselage interference factor,  $R_{wf}$ , is 1.0178.

**b. Lifting surface correction factor,  $R_{Ls}$ .**

From reference 6, fig 4.2, before we calculate the value of lifting surface correction

factor. There are two variables, Mach number,  $M$ , and maximum thickness ratio wing chord sweep angle,  $\Lambda_{\%_{\max}}$ , has to be given first. For HALE, during cruise its Mach number,  $M$ , is 0.55 and maximum thickness ratio wing chord sweep angle  $(\%)_{\max}$  is 0 degrees. Thus, substituting these values into eq. A.8.5 to eq. A.8.9, then the value of lifting surface correction factor,  $R_{Ls}$ , is

$$\alpha = -32.035M^2 + 34.849M - 5.5989 \quad (\text{A.8.5})$$

$$\beta = 83.386M^2 - 90.716M + 13.224 \quad (\text{A.8.6})$$

$$\gamma = -71.713M^2 + 78.064M - 9.9117 \quad (\text{A.8.7})$$

$$\tau = 20.94M^2 - 22.454M + 3.3816 \quad (\text{A.8.8})$$

$$R_{LS} = \alpha \cos \Lambda_{\%_{\max}}^3 + \beta \cos \Lambda_{\%_{\max}}^2 + \gamma \cos \Lambda_{\%_{\max}} + \tau \quad (\text{A.8.9})$$

After calculation, the value of  $R_{Ls}$  is 1.128.

### c. Turbulent flat plate friction coefficient of the wing, $C_{fw}$ .

Before calculating turbulent flat plate friction coefficient of the wing,  $C_{fw}$ , there is one variable, wing Reynolds number, should know first. From section A.7.1, the wing Reynolds number is 1,982,550.

According to reference 29, if the airfoil leading edge angle is less than 10 degree, and Reynolds number of airflow over airfoil is less than 12 million, then the airflow over airfoil can maintain laminar flow until it reaches maximum airfoil thickness ratio point.

Since the Reynolds number over airfoil is 1.98 million which is less than 12 million, and wing leading edge sweep angle is 2 degrees which is less than 10 degree. So, it could maintain laminar flow as long as it reaches maximum airfoil thickness ratio point.

According to reference 28, the maximum thickness ratio point in both MS-0317 and -0313 airfoil is at 0.375 chord length. So, the transition point is at 0.375 chord length.

According to reference 6, fig 4.3 and reference 32, p:331-334, the turbulent flat plate friction coefficient of the wing,  $C_{fw}$ , can be computed by using eq. A.8.10.

$$C_{fw} = 0.04R_{N_{wing}}^{-0.1623} \quad (\text{A.8.10})$$

Substituting the Reynolds number into eq. A.8.10, the value of  $C_{fw}$  is 0.0038. And with following pseudo program, the turbulent flat plate friction coefficient of the wing can be modified as:

if ( $R_N$  .GT.  $12 \cdot 10^7$ ) then

$$C_{fw} = C_{fwt}$$

else

$$x_0 = \left( 0.664 \left( \frac{x_t}{c} \right)^{\frac{1}{2}} R_{Nwing}^{-0.5} \frac{1}{0.225} \right)^{\frac{2}{5}} R_{Nwing}^{\frac{1}{5}}$$

$$C_{fw} = C_{fwt} \left( 1 - (x_{tc} - x_0) \right)^{\frac{2}{5}}$$

end if

For HALE, the Reynolds number is 1.98 million less than 12 million, so the equivalent turbulent flat plate friction coefficient of the wing,  $C_{fw}$ , is 0.0029.

**d. Airfoil thickness location parameter,  $L'$ .**

From reference 6 fig 4.4, for the maximum thickness ratio point in both MS-0317 and -0313 airfoil is at 0.375 chord length great than 0.3 chord length. Therefore, airfoil thickness location parameter,  $L'$ , is 1.2

**e. Thickness ratio,  $t/c$ .**

Since we assume the wing for HALE is MS-0317 at root section and MS-0313 at tip section. So, wing thickness ratio at root is 0.17 and at tip is 0.13. For whole wing thickness ratio, we take mean value of root and tip thickness ratio.

$$\frac{t}{c} = \frac{\left( \frac{t}{c} \right)_{tip} + \left( \frac{t}{c} \right)_{root}}{2} \quad (A.8.11)$$

Therefore, the thickness ratio ( $t/c$ ) for whole wing is 0.15.

**f. Wetted area of the wing,  $S_{wetw}$ .**

In order to calculate wing wetted area, there are two variables, wing /body adjacent area,  $S_{wb}$ , and wing net area,  $S_{net}$ , should be given first.

$$S_{wb} = C_r d \quad (A.8.12)$$

Since wing root chord,  $C_r$ , is 6.348 ft and fuselage diameter,  $d$ , is 3 ft, the value of  $S_{wb}$  is 19.044 ft<sup>2</sup>. And wing area,  $S$ , is 340 ft<sup>2</sup>, therefore the net area of wing, from eq. A.8.13, is 320.956 ft<sup>2</sup>.

$$S_{net} = S - S_{wb} \quad (A.8.13)$$

Wing taper ratio,  $\lambda$ , is 0.5, and the ratio of  $t/c$  ratios at tip and root,  $\tau$ , is calculated by

eq. A.4.8, its value is 0.765.

According to reference 9, eq. B.11, the value of Wetted area of the wing,  $S_{wetw}$ , may be calculated. Substituting these variables into eq. A.8.14.

$$S_{wetw} = 2S_{net} \left( 1 + 0.25 \left( \frac{t}{c} \right), \frac{1 + \tau\lambda}{1 + \lambda} \right) \quad (A.8.14)$$

$$= 667 \text{ ft}^2$$

Finally when we substitute these variables (a to f) into equation A.8.14, then the subsonic wing zero-lift drag coefficient,  $C_{D0w}$ , is 0.00805.

#### A.8.1.1.2 Wing drag coefficient due to lift, $C_{Dlw}$

For wing drag coefficient due to lift, from reference 6, eq. 4.8, may be used.

$$C_{Dlw} = \frac{C_{Lw}^2}{\pi A Re} + 2\pi C_{Lw} \varepsilon_t \nu + 4\pi \varepsilon_t^2 \omega \quad (A.8.15)$$

There are six parameters in eq. A.8.15. Since wing aspect ratio, AR, is given in section A.4.1.1, and Oswald efficiency factor, e, according to section A.7.4, is assumed to be 0.8. Still there are four unknown parameters.

##### a. Wing lift coefficient, $C_{Lw}$

From reference 6 eq 4.11, suggest in preliminary design wing lift coefficient,  $C_{Lw}$ , may be use 1.05 multiply airplane lift coefficient.

$$C_{Lw} = 1.05 C_L \quad (A.8.16)$$

From reference 6, eq.4.10, the airplane lift coefficient,  $C_L$ , is

$$C_L = \frac{2 * W_{TO}}{\rho * V^2 * S} \quad (A.8.17)$$

But in this example, we assume during cruise the airplane lift coefficient is 1 (this figure will correct at later calculation), when multiply by 1.05, the value of wing lift coefficient,  $C_{Lw}$ , is 1.05.

##### b. Wing twist angle, $\varepsilon_t$

Positive for wash-in, for HALE is (change to radian) -3/57.3 radian.

##### c. Induced drag factor due to linear twist, $\nu$

From reference 6, fig 4.9, there are 6 charts suitable for taper ratio from 0.1 to 0.6 to obtain the value of induced drag factor due to linear twist,  $\nu$ .

As the wing taper ratio of HALE is 0.5, we can take the chart of taper ratio is 0.5, then calculate it.

Thus, if taper ratio is 0.5, and  $\beta AR$  is 7.5, then

$$v = -5.6E-07(\beta AR)^3 + 1.38E-05(\beta AR)^2 + 5.54E-05\beta AR + 0.000195 \quad (A.8.18)$$

The value of  $v$  is 0.000576.

**d. Zero-lift drag factor due to linear twist,  $\omega$**

From reference 6, fig 4.10, there are 7 charts suitable for taper ratio from 0.1 to 0.75 to obtain the value of zero-lift drag factor due to linear twist,  $\omega$ .

Like we did in solving the value of induced drag factor due to linear twist,  $v$ , we can take the chart of taper ratio is 0.5, then calculate it.

Thus, if taper ratio is 0.5, and  $\beta AR$  is 7.5, then

$$\begin{aligned} \omega &= -5.7E-06(\beta AR)^2 + 0.00014\beta AR + 0.00139 \\ &= 0.002255 \end{aligned} \quad (A.8.19)$$

Since  $\beta = 0.835$ , then

$$\omega = \frac{0.002255}{\beta} = 0.0027$$

Substituting these variables from, a to d, into eq. A.8.15, after calculation, the value of wing drag coefficient due to lift,  $C_{Diw}$ , is 0.02934.

### A.8.1.2 Fuselage drag Coefficient

As mentioned in section A.8.1.1, like wing drag coefficient, the fuselage drag coefficient in subsonic flow also can be divided into zero-lift drag and lift drag.

$$C_{Dfus} = C_{D0fus} + C_{Difus} \quad (A.8.20)$$

#### A.8.1.2.1 Fuselage drag coefficient due to zero drag, $C_{D0fus}$

According to reference 6, eq. 4.30, the subsonic fuselage zero-lift drag coefficient may be calculated as :

$$C_{D0fus} = R_{wf} C_{ffus} \left( 1 + \frac{60}{(L/D)^3} + 0.0025 \frac{L}{D} \right) \frac{S_{wetfus}}{S} \quad (A.8.21)$$

In order to calculate wing drag coefficient due to lift, there are six parameters should be solved in eq. A.8.21. Since wing area,  $S$ , fuselage length,  $L$ , and fuselage diameter,  $D$ , is given in section A.4.1.1 and A.4.1.4, there are three parameters unknown.

**a. Wing/fuselage interference factor,  $R_{wf}$**

From reference 6, p:44, if for fuselage only,  $R_{wf}$  is assumed to be 1.

**b. Turbulent flat plate skin friction coefficient of the fuselage,  $C_{ffus}$ .**

As we calculated in eq. A.8.10, now the Reynolds number of fuselage is  $1.4 \times 10^7$ , so the turbulent flat plate skin friction coefficient of the fuselage,  $C_{ffus}$ , is 0.00276.

**c. Wetted area of fuselage**

According to reference 9, eq. B.6, wetted area of fuselage with cylindrical mid-sections ( $S_{wet fus}$ ) may be calculated. And  $\lambda_f$  is fineness ratio, its value, from eq. A.5.9, is 11.9.

$$\begin{aligned} S_{wet fus} &= \pi DL \left(1 - \frac{2}{\lambda_f}\right)^{\frac{2}{3}} \left(1 + \frac{1}{\lambda_f^2}\right) \\ &= 299.77 \text{ ft}^2 \end{aligned} \quad (A.8.22)$$

Substituting these variables, from a to c, into eq. A.8.22, after calculation, the value of fuselage zero-lift drag coefficient,  $C_{D0 fus}$  is 0.002637.

**A.8.1.2.2 Fuselage drag coefficient due to lift,  $C_{Difus}$**

For fuselage drag coefficient due to lift, the eq. 4.33 of reference 6, may be used..

$$C_{Difus} = 2\alpha^2 \frac{S_{bfus}}{S} + \eta C_{dc} \alpha^3 \frac{S_{pfus}}{S} \quad (A.8.23)$$

In order to calculate fuselage drag coefficient due to lift, six parameters in eq. A.8.23 need to be solved. Since wing area,  $S$ , is given in section A.4.1.1, there are five parameters unknown.

**a. Fuselage angle of attack,  $\alpha$**

$$\alpha = \frac{W/qS - C_{L0}}{C_{L\alpha}} \quad (A.8.24)$$

Before calculating the value of fuselage angle of attack,  $\alpha$ , there are two parameters should be given first. Zero-angle-of-attack lift coefficient,  $C_{L0}$ , and the aircraft lift curve slope,  $C_{L\alpha}$ .

**a.1 Zero-angle-of- attack lift coefficient,  $C_{L0}$ ,**

From reference 6, eq. 8.32, may be used.

$$C_{L0} = C_{L0wf} + C_{Lah} \eta_h \left( \frac{S_{TH}}{S} \right) (i_h - \varepsilon_{0h}) \quad (A.8.25)$$

In eq. A.8.25, if both tailplane incidence angle,  $i_h$ , and the magnitude downwash angle,  $\varepsilon_{0h}$ , are zero, then zero-angle-of- attack lift coefficient,  $C_{L0}$ , is equal to zero-angle-of- attack lift coefficient of the wing-fuselage combination,  $C_{L0wf}$ .

$$C_{L0} = C_{L0wf}$$

and zero-angle-of- attack lift coefficient of the wing-fuselage combination,  $C_{L0wf}$ , may be solved by using reference 6, eq. 8.33.

$$C_{L0wf} = (i_w - \alpha_{0Lw})C_{L\alpha wf} \quad (A.8.26)$$

From section A.7.2, wing-fuselage lift curve slope,  $C_{L\alpha wf}$ , is 6.082, and wing-body incidence angle,  $i_w$ , is 6.796 degrees. So, in eq. A.8.26 there is only one unknown parameter, the wing zero-lift angle of attack,  $\alpha_{0Lw}$ .

From reference 6, eq. 8.21, at subsonic speed, for wings with constant airfoil sections and linear twist distributions. The wing zero-lift angle of attack,  $\alpha_{0Lw}$ , may be computed by eq. A.8.27.

$$\alpha_{0Lw} = \left( \alpha_{0l} + \left( \frac{\Delta\alpha_0}{\varepsilon_t} \right) \varepsilon_t \right) \frac{(\alpha_{0l})_{atM}}{(\alpha_{0l})_{atM=0.3}} \quad (A.8.27)$$

In eq. A.8.27, there are four parameters that should be given first. From reference 28, the zero-lift angle of attack,  $\alpha_{0l}$ , of MS(1)-0317 and -0313 airfoil is -3 degrees, and the wing twist angle,  $\varepsilon_t$ , is assumed to be -3 degrees. Therefore, there are two parameters need to be calculated.

From reference 6, fig 8.41(b), if quarter wing chord angle is less than 40 degree and wing aspect ratio is great than 6. then the change in wing zero-lift angle of attack per degree of linear wing twist,  $\frac{\Delta\alpha_0}{\varepsilon_t}$ , can be calculated from below.

For HALE, the wing taper ratio,  $\lambda$ , is 0.5 and quarter wing chord angle  $\Lambda_{\frac{1}{4}}$  is 0 degrees.

$$\frac{\Delta\alpha_0}{\varepsilon_t} = -3.6E - 07\Lambda_{\frac{1}{4}}^2 + 0.000445\Lambda_{\frac{1}{4}} - 0.413 = -0.41$$

Now, about the value of  $\frac{(\alpha_0)_M}{(\alpha_0)_{M=0.3}}$ , from reference 6, fig 8.42,



if  $(M^* \cos(\Lambda_{c/4}) .LT. 0.5)$  then

$$\frac{(\alpha_0)_M}{(\alpha_0)_{M=0.3}} = 1$$

else if  $((M^* \cos(\Lambda_{c/4})) .GT. 0.5$  and  $t/c .EQ. 0.14)$  then

$$\begin{aligned} \frac{(\alpha_0)_M}{(\alpha_0)_{M=0.3}} &= -84.4 \left( M \cos(\Lambda_{c/4}) \right)^3 + 117.2 \left( M \cos(\Lambda_{c/4}) \right)^2 - 54.1 M \cos(\Lambda_{c/4}) + 9.295 \\ &= 0.9468 \end{aligned}$$

else if  $((M^* \cos(\Lambda_{c/4})) .GT. 0.5$  and  $t/c .EQ. 0.16)$  then

$$\begin{aligned} \frac{(\alpha_0)_M}{(\alpha_0)_{M=0.3}} &= -96.7 \left( M \cos(\Lambda_{c/4}) \right)^3 + 128.9 \left( M \cos(\Lambda_{c/4}) \right)^2 - 58.5 M \cos(\Lambda_{c/4}) + 9.97 \\ &= 0.7 \end{aligned}$$

end if

The thickness ratio of HALE is 15 percent, therefore take average value;

$$\frac{(\alpha_0)_M}{(\alpha_0)_{M=0.3}} = \frac{0.9468 + 0.7}{2} = 0.8235$$

Thus, substituting these variables into eq. A.8.27, then the wing zero-lift angle of attack,  $\alpha_{0Lw}$ , is -0.0253 (the more accurate calculation method is listed in paragraph A.17.1).

Substituting these values of wing zero-lift angle of attack,  $\alpha_{0Lw}$ , wing-fuselage lift curve slope,  $C_{L\alpha wf}$ , and wing-body incidence angle,  $i_w$ , into eq. A.8.26, then the value of zero-angle-of- attack lift coefficient,  $C_{L0}$ , is 0.875.

#### **a.2 Aircraft lift curve slope, $C_{L\alpha}$**

From section A.7.2, eq. A.7.14, the aircraft lift curve slope,  $C_{L\alpha}$ , is 6.666.

Assume the aircraft weight,  $W$ , is 10400 lbs, flight at 60000 ft, flight Mach number is 0.55, and wing area,  $S$ , is 340. ft<sup>2</sup>, so dynamic pressure,  $q$ , is 31.7 lbf/ft<sup>2</sup>.

Substituting these variables into eq. A.8.24, the fuselage angle of attack is 0.0146 rad.

#### **b. Fuselage base area ( $S_{bfus}$ )**

From reference 6, fig. 4.17, since there is no base area at the end of Tier-II Plus fuselage, so  $S_{bfus}$  is zero.

#### **c. Ratio of the drag of a finite cylinder to the drag of an infinite cylinder, $\eta$**

From reference 6, fig 4.19, following equation may be used.

$$\eta = -0.0005\left(\frac{L}{D}\right)^2 + 0.0214\frac{L}{D} + 0.5216 \quad (\text{A.8.28})$$

$$= 0.705$$

**d. Experimental steady state cross-flow drag coefficient a circular cylinder,  $C_{dc}$ .**

From reference 6, fig 4.20, if Mach number multiplied by  $\sin\alpha$  is less than 0.28 then  $C_{dc}$  is constant, 1.2. During cruise the Mach number of HALE is 0.55, and  $\sin\alpha$  is 0.0145. The result, 0.008, is less than 0.38. Therefore,  $C_{dc}$  is 1.2

**e. Fuselage platform area,  $S_{p\text{ fus}}$**

From reference 6, fig 4.17. In order to calculate the fuselage platform area, we divided HALE into three segment. first part, nose section. Second part middle body section. Last part tail section.

**e.1. Nose section**

According to the profile of HALE, nose section is assumed to be 0.25 multiplied by fuselage length. Therefore, the nose section area is

$$S_{pn} = 0.5\left(\pi 0.25L \frac{D}{2}\right) = 21 \text{ ft}^2$$

**e.2. The middle section area**

$$S_{pm} = 0.45LD = 48.2 \text{ ft}^2$$

**e.3. The tail section area**

Assuming the tail section is 0.3 multiplied by fuselage length. Therefore, the tail section area is

$$S_{pt} = 0.5\left(\pi 0.3L \frac{D}{2}\right) = 25.235 \text{ ft}^2$$

Thus, the fuselage planform area

$$S_{pfus} = S_{pn} + S_{pm} + S_{pt} = 94.44 \text{ ft}^2$$

Finally when we put these variables, from a to e, into eq. A.8.23, then fuselage drag coefficient due to lift,  $C_{Difus}$  is 0.0000007.

### A.8.1.3 Empennage drag coefficient

According to reference 6, chap. 4, in subsonic, total empennage drag coefficient,  $C_{Demp}$ , may be calculated by using eq. A.8.29, empennage zero-lift drag coefficient,  $C_{D0emp}$ , and empennage coefficient due to lift,  $C_{Diemp}$

$$C_{Demp} = C_{D0emp} + C_{Diemp} \quad (\text{A.8.29})$$

#### A.8.1.3.1 Empennage zero-lift drag coefficient, $C_{D0emp}$

According to reference 6, chap 4, in subsonic flight, the empennage zero-lift drag coefficient can be calculated by using the same equation to calculate the wing zero-lift drag coefficient. Only change wing area to tailplane area. Therefore, after substituting tailplane makes eq. A.8.4 becomes

$$C_{D0h} = R_{wfh} R_{Lsh} C_{fh} \left( 1 + L'_h \left( \frac{t}{c} \right)_h + 100 \left( \frac{t}{c} \right)_h^4 \right) \frac{S_{weth}}{S} \quad (A.8.30)$$

$$C_{D0v} = R_{wfv} R_{Lsv} C_{fv} \left( 1 + L'_v \left( \frac{t}{c} \right)_v + 100 \left( \frac{t}{c} \right)_v^4 \right) \frac{S_{wetv}}{S} \quad (A.8.31)$$

In order to calculate empennage zero-lift drag coefficient, there are six variables in eq. A.8.30, A.8.31 should be given first. Since the airfoil of tailplane is NACA 0012, then the airfoil thickness ratio  $(t/c)_v$ ,  $(t/c)_h$  is 0.12. So, there are five variables which still need to be computed.

##### a. Empennage/fuselage interference factor, $R_{wfh}$ , $R_{wfv}$

From reference 6, chap 4.4.1.1, this value is equal to 1.

##### b. Lifting surface correction factor, $R_L$ ,

Based on reference 6, fig 4.2, there are two variables, Mach number and maximum thickness ratio of tailplane sweep angle,  $\Lambda_{\frac{1}{c}\max}$ , has to be given first. For HALE, during cruise its Mach number is 0.55 and maximum thickness ratio horizontal and vertical tailplane chord sweep angle is 0 and 30 degrees.

From eq. A.8.5 to A.8.8, the value of  $\alpha$  is 3.88,  $\beta$  is -11.45,  $\gamma$  is 11.33 and  $\tau$  is -2.63. Substituting these values into eq. A.8.9, then  $R_{Lsh}$  and  $R_{Lsv}$  is 1.128 and 1.113.

##### c. Turbulent flat plate friction coefficient of tailplane, $C_f$

Before calculating, there is one variable which should be given first : empennage Reynolds number. During cruise, altitude is 60000 ft , density,  $\rho$ , is 0.000224 , Horizontal and vertical tailplane MAC is 3.53 and 3.43 ft, velocity is 532.44 fps and viscosity, dynamic viscosity,  $\mu$ , is 0.000000297 lb-s/ft<sup>2</sup>.

Substituting these values into eq. A.7.4, the Reynolds number at horizontal and vertical tailplane is  $1.41 \times 10^6$  and  $1.38 \times 10^6$ .

Since the Reynolds number over horizontal tailplane is around 1.4 million less than 12 million, and the leading edge sweep angle of horizontal tailplane is 0 degrees less than 10 degrees. And the leading edge sweep angle of vertical tailplane is 30 degree, and the Reynolds number over vertical tailplane is around 1.38 million less than 6.5 million. From reference 29, the airflow can maintain laminar flow as long as it reaches airfoil maximum thickness ratio point.

According to reference 19, the maximum thickness ratio point in NACA 0012 airfoil is at 0.3 chord length. So, the transition point is at 0.3 chord length.

According to eq. A.8.10, the turbulent flat plate friction coefficient of the horizontal and vertical tailplane,  $C_{fh}$  and  $C_{fv}$ , is 0.004015 and 0.00403.

if ( $R_{Nt} \cdot LT \cdot 12 \cdot 10^7$ ) then

$$x_0 = \left( 0.664 \left( \frac{x_t}{c} \right)^{\frac{1}{2}} R_{Nt}^{-0.5} \frac{1}{0.225} \right)^{\frac{5}{3}} R_{Nt}^{\frac{1}{3}}$$

$$C_{fh} = C_{fht} \left( 1 - (x_{tc} - x_0) \right)^{\frac{5}{6}}$$

else

$$C_{fh} = C_{fht}$$

end if

For HALE, the Reynolds number at tailplane is 1.4 million, far less than 12 million. Therefore, the turbulent flat plate friction coefficient of the horizontal and vertical tailplane,  $C_{fh}$  and  $C_{fv}$  is 0.00333 and 0.00335.

#### d. Airfoil thickness location parameter, $L'_h$ , $L'_v$

From reference 6, fig 4.4, if the airfoil maximum thickness ratio point is equal or greater than 0.3 chord length, its value is 1.2. For NACA 0012, the airfoil maximum thickness ratio point is equal 0.3. Therefore, the airfoil thickness location parameter,  $L'_h$  and  $L'_v$ , is 1.2

#### e. Wetted area of the empennage, $S_{we}$

The vertical tailplane is connected to the horizontal tailplane. Therefore, we assume there is no overlap area for these tailplane, the net area of vertical tailplane is equal to vertical tailplane area.

The horizontal tailplane is mounted at the fuselage tail section, so its net area should reduce the body and tail adjacent part. Thus, the net area of horizontal tailplane is 51.04.

The horizontal and vertical tailplane taper ratio,  $\lambda_h$  and  $\lambda_v$  is 1 and 0.5. The ratio of  $t/c$  ratios at tip and root,  $\tau_t$  and  $\tau_v$  is 1.

Substituting these variables into eq. A.8.14, the wetted area of the horizontal and vertical tailplane is 117.12 and 86.72 ft<sup>2</sup> respectively.

Substituting these values, from a to e, into eq. A.8.30 and A.8.31. After calculation, the horizontal and vertical zero-lift drag coefficient is 0.0015 and 0.000553 respectively. Since the number of the vertical tailplane is 2, then we should multiply 2

by above vertical zero-lift drag coefficient. Then the vertical zero-lift drag coefficient becomes 0.0011.

Finally, the empennage zero-lift drag coefficient,  $C_{D0emp}$ , is 0.0026.

#### A.8.1.3.2 Empennage drag coefficient due to lift, $C_{Diemp}$

Since the drag coefficient due to lift of vertical tailplane is very small, so its assumed equal to zero. Therefore, for the drag coefficient due to lift, we only consider the horizontal tailplane. According to reference 6, eq. 4.51, the drag coefficient due to lift,  $C_{Diemp}$  may be calculated by

$$C_{Diemp} = \left( \frac{C_{Lh}^2}{\pi A R_{TH} e_h} \right) \frac{S_{TH}}{S} \quad (A.8.32)$$

Before calculating there are 5 variables that should be given in advance. But from sections A.4.1.1 and A.4.1.5, the wing area, horizontal tailplane area and horizontal tailplane aspect ratio are given. Thus, only 2 variables remain unknown.

##### a. Lift coefficient of tailplane, $C_{Lh}$

From reference 6, eq. 4.46, the amount of lift coefficient on tailplane can be computed from :

$$C_{Lh} = C_{Lah} (\alpha_h - \alpha_{0Lh}) \quad (A.8.33)$$

$$\alpha_h = \alpha \left( 1 - \frac{d\varepsilon}{d\alpha} \right) \quad (A.8.34)$$

From eq.A.7.13, the down wash gradient at the horizontal tail,  $\frac{d\varepsilon}{d\alpha}$ , is 0.303, and from eq. A.8.24, the fuselage angle of attack,  $\alpha$ , is 0.0146 radian. Thus, substituting these variables into eq. A.8.34, the value of  $\alpha_h$  is 0.01018, and the zero-lift angle of attack,  $\alpha_{0Lh}$ , for NACA 0012 airfoil is zero. From eq. A.7.9, the tailplane lift curve slope,  $C_{Lah}$ , is 4.64.

Substituting these variables into eq. A.8.33, the lift coefficient of tailplane,  $C_{Lh}$ , is 0.047.

##### b. Oswald efficiency for empennage, $e_h$

From reference 9, eq. (G-54)

$$e_h = 1 - \frac{0.25}{\cos^2 \Lambda_H} \quad (A.8.35)$$

The horizontal tailplane leading edge sweep angle,  $\cos^2 \Lambda_H$ , is 0 degrees. Thus, the value of  $e_h$  is 0.75.

Substituting these variables into eq. A.8.32, the tailplane drag coefficient due to lift,  $C_{Diemp}$ , is 0.0000347.

#### A.8.1.4 Trim drag prediction, $C_{Dt}$

Trim drag is caused by lift generated on horizontal tailplane as a result to moment trim an airplane. Reference 7, chap. 4.10 states there are two types of trim drag, trim drag due to lift and trim drag due to profile drag.

$$C_{Dt} = C_{Dtl} + C_{Dtp} \quad (A.8.36)$$

##### A.8.1.4.1 Trim drag due to lift

Reference 7, eq. 4.83 may be used to calculate trim drag due to lift.

$$C_{Dtl} = \frac{C_{Lh}^2}{\pi A_h e_h} \frac{S}{S_h} \quad (A.8.37)$$

There are 5 variables in eq. A.8.37,  $C_{Lh}$  is the horizontal tail incremental lift coefficient required for trim,  $A_h$  is the horizontal tail aspect ratio,  $e_h$  is horizontal tail Oswald efficiency,  $S$  is wing area, and  $S_h$  is horizontal tail area. The unknown variable is  $C_{Lh}$ .

Reference 66 suggests that  $C_{Lh}$  may be calculated by;

$$C_{Lh} = \frac{C_{Lw}(H - H_0)}{l_T} MAC + \frac{C_{m0wb}}{l_T} MAC \quad (A.8.38)$$

In eq. A.8.38,  $H$  is aircraft most aft C.G. location, in this case its value is 0.36,  $H_0$  is wing body aerodynamic centre, its value is 0.227,  $l_T$  is the distance from aircraft most aft C.G. location to horizontal tail aerodynamic centre, its value is 13.77 ft,  $C_{Lw}$  is wing lift coefficient, its value is 1.05, and  $C_{m0wb}$  is wing body pitch moment coefficient at zero lift, its value is -0.083 (more accurate calculation is in paragraph A.17.1).

Substituting these variables into eq. A.8.38, the value of  $C_{Lh}$  is 0.02. When substitute this value into eq. A.8.37, the value of trim drag due to lift is 0.0002.

##### A.8.1.4.2 Trim drag due to profile drag, $C_{Dtp}$

In accordance with reference 7, chap. 4.10.2, the trim drag due to profile drag is caused by the deflection of an elevator to trim aircraft. Therefore, if there is no deflection of elevator to trim aircraft, the trim drag due to profile drag is zero. In preliminary design phase, we assume the elevator deflection angle is zero, therefore, the  $C_{Dtp}$  is 0.

The trim drag prediction,  $C_{Dt}$ , is equal to the trim drag due to lift, as 0.0002.

### A.8.1.5 Engine nacelle drag coefficient

According to reference 6, chap 4, in general the shape of a nacelle is like a fuselage. Therefore, it can be treated as a small fuselage, and use the same drag formula which is being used to calculate fuselage drag.

$$C_{Dnac} = C_{D0nac} + C_{Dinac} \quad (A.8.39)$$

#### A.8.1.5.1 Engine nacelle drag coefficient due to zero drag

According to reference 6, chap 4.5.1.1, it suggests that in subsonic flight the engine nacelle zero-lift drag coefficient may be treated as if it is a fuselage. Thus, if we replace the fuselage length and diameter to engine length and diameter, then the eq. A.8.21 becomes:

$$C_{D0nac} = R_{wf} C_{fnac} \left( 1 + \frac{60}{\left( \frac{l_{nac}}{d_{nac}} \right)^3} + 0.0025 \frac{l_{nac}}{d_{nac}} \right) \frac{S_{wetnac}}{S} \quad (A.8.40)$$

Before calculation, in eq. A.8.40, there are six variables should be given in advance. The engine nacelle length,  $l_{nac}$ , is 10 ft, maximum engine nacelle diameter,  $d_{nac}$ , is 3 ft. From paragraph A.8.1.2.1, the value of fuselage/nacelle interference factor,  $R_{wf}$ , is 1, and the value of wing area is given. Therefore, there are two variables which need to be calculated.

##### a. Turbulent flat plate skin friction coefficient of the nacelle, $C_{fnac}$

By using eq. A.8.10, the turbulent flat plate skin friction coefficient of the nacelle,  $C_{fnac}$ , can be obtained. Before calculating, there is one variable should be given, the engine nacelle Reynolds number. By using eq. A.7.4, engine nacelle Reynolds number,  $R_{Nnac}$ , is 4,012,422.76.

Therefore, substituting the engine nacelle Reynolds number,  $R_{Nnac}$ , into eq. A.8.10. After calculation, the value of turbulent flat plate skin friction coefficient of the nacelle ( $C_{fnac}$ ) is 0.00339.

##### b. Wetted area of nacelle, $S_{wetnac}$

The wetted area of nacelle,  $S_{wetnac}$ , may be obtained by substituting the fuselage diameter and length, and fuselage fineness ratio into eq. A.8.22.

The engine nacelle diameter,  $d_{nac}$ , is 3 ft, the engine nacelle length,  $l_{nac}$ , is 10 ft, and the from eq. A.5.9, the engine nacelle fineness ratio,  $\lambda_{nac}$ , is 3.33.

Substituting these variables into eq. A.8.22, the wetted area of nacelle,  $S_{wetnac}$ , as 55.75 ft<sup>2</sup> is obtained.

Substituting these variables, from a to b, into eq. A.8.37, after calculation, the value of engine nacelle zero-lift drag coefficient  $C_{D0nac}$  is 0.00146.

#### **A.8.1.5.2 Engine nacelle drag coefficient due to lift, $C_{Dinac}$**

The same formula to calculate the fuselage drag coefficient due to lift in paragraph A.8.1.2.2, eq. A.8.23, may be used to calculate the engine nacelle drag coefficient due to lift.

The wing area,  $S$ , has been given. Therefore, in eq. A.8.23, there are five unknown variables left.

##### **a. Nacelle angle of attack;**

Assume engine nacelle angle of attack,  $\alpha_{nac}$ , is 1.5 degree.

##### **b. Engine nacelle base area ( $S_{b\ fus}$ )**

The diameter of nacelle base area is very small, so the area of nacelle is 0.

##### **c. Ratio of the drag of a finite cylinder to the drag of an infinite cylinder, $\eta$**

From eq. A.8.28, its value,  $\eta$ , is 0.58.

##### **d. Experimental steady state cross-flow drag coefficient of a circular cylinder, $C_{dc}$**

As we calculated in section A.8.1.2.2, the value of  $C_{dc}$  is 1.2.

##### **e. Engine nacelle platform area ( $S_{p\ nac}$ )**

In order to calculate the nacelle platform area, we divided nacelle into two segment. first part, forward body. Second part tail section.

The proportion of forward nacelle section is 0.75 multiplied by the fuselage length, tail nacelle section is 0.25 multiplied by the fuselage length.

Therefore, the forward nacelle section area is

$$\begin{aligned} S_{pf} &= 0.75 l_f d_{nac} \\ &= 22.5 \text{ ft}^2 \end{aligned} \quad (\text{A.8.41})$$

the tail nacelle section area is

$$\begin{aligned} S_{pt} &= 0.5 \left( \pi 0.25 l_{nac} \frac{d_{nac}}{2} \right) \\ &= 5.89 \text{ ft}^2 \end{aligned} \quad (\text{A.8.42})$$

Therefore, the engine nacelle platform area is 28.39 ft<sup>2</sup>.

Substituting these variables, from a to e, into eq. A.8.23, the value of nacelle drag coefficient due to lift,  $C_{Dinac}$ , is 0.000000935.

#### **A.8.1.6 Nacelle fuselage interference drag coefficient, $C_{Dnp}$**

From reference 6, eq. 4.65, may be used to calculate the nacelle fuselage interference drag coefficient.



$$C_{Dnp} = F_{a2} \left( C_{Dn}' - 0.05 \right) \frac{S_n}{S} \quad (\text{A.8.43})$$

In eq. A.8.43, for fuselage nacelle intersections without local area ruling,  $F_{a2}$  is 1.  $S_n$  is the maximum frontal area of the nacelle, assume its  $3.93 \text{ ft}^2$ . From reference 6, fig. 4.42, the value of  $C_{Dnp}$  is 0.1124.

Substituting these values into eq. A.8.40, then the nacelle fuselage interference drag coefficient,  $C_{Dnp}$ , is 0.00072.

#### A.8.1.7 Landing gear drag coefficient, $C_{Dgear}$

The landing gear for HALE is retractable, so there is no landing gear drag during flight. But during take-off and landing, the landing gear has to be stay outside of the aircraft and landing gear drag has to be calculated.

According to reference 6, chap. 4, and reference 9 appendix F, assuming the undercarriages of HALE main wheel is using 30-5 tire and nose wheel is using 32-6 tire. The landing gear tire drag coefficient may be computed in eq.(A.8.44)

$$C_{Dgear} = 2C_{DMgear} \frac{S_{Mgear}}{S} + C_{DNgear} \frac{S_{Ngear}}{S} \quad (\text{A.8.44})$$

According to reference 9, fig f-18,  $C_{dMgear}$  is 0.35, for 30-5 tire, the tire area,  $S_{mgear}$ , is  $1.033 \text{ ft}^2$  and for 32-6 tire,  $C_{dNgear}$  is 0.31, the tire area,  $S_{ngear}$ , is  $1.367 \text{ ft}^2$ . The main landing gear has two tires. Substituting these variables into eq. A.8.44, the landing gear tire drag coefficient,  $C_{Dgear}$  is 0.00357.

Reference 66 suggests that the landing gear strut drag coefficient is around the tire drag coefficient multiplied by two, therefore, the landing gear strut drag coefficient is 0.00714.

The landing gear drag coefficient is equal to the landing gear strut drag coefficient add the tire drag coefficient, which is 0.0107.

#### A.8.1.8 Total drag coefficient

The aircraft drag , as mentioned earlier, is varied with aircraft velocity, weight and altitude. In this section, we only make an example of how to calculate aircraft drag.

In this example, we assume the aircraft is cruise at 60,000 ft, aircraft weight is 10,400 lbs, and flight Mach number is 0.55. After calculation, the zero-drag drag and drag coefficient due to lift is listed as:

a. The total airplane zero-drag coefficient is :

$$\begin{aligned} C_{D0} &= C_{D0w} + C_{D0fus} + C_{D0emp} + C_{D0nac} + C_{Dnp} \\ &= 0.0155 \end{aligned} \quad (A.8.45)$$

b. The total airplane drag coefficient due to lift is :

$$\begin{aligned} C_{Di} &= C_{Diw} + C_{Difus} + C_{Diemp} + C_{Dinac} + C_{Dil} \\ &= 0.0296 \end{aligned} \quad (A.8.46)$$

Thus, the total drag is :

$$\begin{aligned} C_D &= C_{D0} + C_{Di} \\ &= 0.045 \end{aligned} \quad (A.8.47)$$

## A.9 Stall analysis

According to reference 15, eq.15.1, the definition of level flight stall is that the all forces are in equilibrium as the aircraft moves at a constant speed and altitude. The flight speed in level flight may be calculated by :

$$V = \sqrt{\frac{2L}{\rho C_L S}} = \sqrt{\frac{2W}{\rho C_L S}} \quad (A.9.1)$$

If the value of  $C_L$  , in eq. A.9.1, is substituted by  $C_{Lmax}$  , then the stall speed may be obtained.

$$V_s = \sqrt{\frac{2W}{\rho C_{Lmax} S}} \quad (A.9.2)$$

Thus, if we want to calculate the aircraft level flight stall speed, from eq. A.9.2, apart from the aircraft weight, wing area, altitude etc., the maximum lift coefficient,  $C_{Lmax}$ , should be given as well.

According to the specification of HALE, after take-off , first climb to an altitude of 65,000 ft, then cruise 500 nautical miles to target area, and then loiter at 65,000 ft for 24 hours loiter.

In order to calculate the stall characteristics of the HALE, we assume the HALE starts to cruise is at 60,000 ft. Thus, the following condition is made, altitude is 60,000 ft, flight Mach number is 0.55.

According to reference 7, chap. 5, reference 9, appendix E and reference 16, 17. Following steps are taken to calculate the aircraft stall speed, at different altitude and different weight, of the HALE.

### A.9.1. Maximum lift coefficient of the wing, $(C_{Lmax})_w$

From reference 18, and reference 9, appendix E, the basic procedure for estimating the maximum lift coefficient of high aspect ratio straight wings is;

#### a. The total spanwise lift coefficient

The total spanwise lift coefficient may be divided into two parts - basic and additional lift coefficient.

$$C_l(y) = C_l(y)_{basic} + C_l(y)_{add} \quad (A.9.3)$$

and the basic lift coefficient may be calculated as :

$$C_l(y)_{basic} = 0.5((\alpha_{0Lw} - \alpha_{0L}) + \varepsilon(y))C_{l\alpha}(y, t) \quad (A.9.4)$$

and the additional lift coefficient may be calculated as:

$$C_l(y)_{add} = \frac{C_{l\alpha}(y)}{C_{l\alpha}C(y)} \left( \left( \frac{C(y)C_{l\alpha}(y, t)}{2C_{l\alpha}} \right) + \frac{2S}{\pi b} \left( 1 - \left( \frac{2y}{b} \right)^2 \right)^{\frac{1}{2}} \right) \quad (A.9.5)$$

Before obtaining the value of basic and additional lift coefficient, there are six variables should be given first.

#### a.1 Wing zero-lift angle of attack, $\alpha_{0Lw}$

From eq. A.8.27, the wing zero-lift angle of attack,  $\alpha_{0Lw}$ , is -0.0253.

#### a.2 Airfoil zero-lift angle of attack, $\alpha_{0L}$

The wing airfoil is belong to the MS(1)-series. From reference 28, the airfoil zero-lift angle of attack,  $\alpha_{0L}$ , is -3 degree.

#### a.3 Twist angle of a local section at y, out from centre line, $\varepsilon(y)$

$$\varepsilon(y) = \frac{-3y}{\frac{1}{2}b} \quad (A.9.6)$$

From eq. (A.9.6), the twist angle at airfoil local section is varied with different wing span location, y, and wingspan, b .

#### a.4 Local airfoil lift curve slope, $C_{l\alpha}(y, t)$

In order to simplify the calculation, in this example we assume local airfoil lift curve slope is equal to mean value of airfoil lift curve slope,  $C_{L\alpha}$  .

From eq. A.7.4, the mean value of Reynolds number of the wing is  $1.95 \cdot 10^6$ .

From eq. A.7.3, the theoretical airfoil lift curve slope in incompressible flow,  $C_{l\alpha 0T}$  , is 6.996.

From eq. A.7.5, the value of  $\frac{(C_{l\alpha})_{atM=0}}{C_{l\alpha 0T}}$  is 0.8424.

From eq. A.7.6, the airfoil lift curve slope (no compression),  $(C_{l\alpha})_{atM=0}$ , is 5.893.

From eq. A.7.7, the mean value of airfoil lift curve slope,  $(C_{l\alpha})_{atM}$ , is 7.056.

In order to simplify the calculation in eq. A.9.5, following reference 18, we assume local lift curve slope is equal to the mean lift curve slope of the whole wing ( $C_{l\alpha}(y) = C_{l\alpha}$ ).

**a.5 Local airfoil chord,  $C(y)$ , may be calculated as:**

$$C(y) = C_r - \tan \Lambda_{le} y + \tan \Lambda_{te} y \quad (A.9.7)$$

The leading edge sweep angle and the trailing edge sweep angle of The HALE is 2 degree backward and 3.09 degree forward respectively.

Substituting these variables, from a.1 to a.5, into eq. A.9.4 and eq. A.9.5, then the basic and additional lift coefficient can be obtained.

**b. Spanwise local airfoil maximum lift coefficient,  $C_{lmax}$**

Since the wing of the HALE is straight taper wing, so the spanwise local airfoil maximum lift coefficient, from wing root to wing tip, may be calculated as:

$$C_{lmax} = C_{lmaxr} - \frac{C_{lmaxr} - C_{lmaxt}}{b/2} y \quad (A.9.8)$$

In eq. A.9.8, the wing root maximum lift coefficient,  $C_{lmaxr}$ , and the wing tip maximum lift coefficient,  $C_{lmaxt}$ , should be given first.

From reference 28, fig 18 and fig 23, we can find the maximum lift coefficient is dependent on Reynolds number. So, in order to obtain maximum lift coefficient, the Reynolds number should know first.

Though the length of the wing root is 6.35 ft, and the wing tip is 3.175 ft. By using eq. A.7.4, then the Reynolds number at root and tip is 2547060 and 1273530 respectively.

From reference 28, fig 23, For MS(1)-0317 airfoil, we find two different maximum lift coefficient 1.6, 1.82, at the Reynolds number is equal to  $2 \times 10^6$  and  $4 \times 10^6$ .

Since the Reynolds number at root is  $2.54 \times 10^6$ , this value is in between  $2 \times 10^6$  and  $4 \times 10^6$ . Thus, we have to use interpolation methods to obtain its maximum lift coefficient.

$$C_{lmaxr} = 1.6 + \frac{1.82 - 1.6}{4 \times 10^6 - 2 \times 10^6} (R_{nr} - 2 \times 10^6) \quad (A.9.9)$$

After calculation, the value at the Reynolds number at wing root is 1.66.

From reference 28, fig 18, for MS(1)-0313 airfoil, the minimum Reynolds number is  $2.1 \times 10^6$  and its maximum lift coefficient is 1.7. Since the Reynolds number at wing tip is  $1.27 \times 10^6$  which is less than  $2.1 \times 10^6$ . From this figure, we decide to take a approximate value of 1.55 as the maximum lift coefficient,  $C_{lmaxt}$ , at tip.

When we substitute the maximum lift coefficient at wing root and tip into eq. A.9.8, then spanwise local airfoil maximum lift coefficient,  $C_{lmax}$ , can be obtained.

Now, from above calculation, the local airfoil maximum lift coefficient,  $C_{lmax}$ , the value of basic and additional lift coefficient all are obtained. Thus, the maximum lift coefficient of the wing,  $(C_{lmax})_w$ , can be calculated as:

$$(C_{Lmax})_w = \frac{C_{lmax} - C_{lbasic}}{C_{ladd}} \quad (A.9.10)$$

Since the values of spanwise local airfoil maximum lift coefficient,  $C_{lmax}$ , the basic lift coefficient,  $C_{lbasic}$ , and additional lift coefficient,  $C_{ladd}$ , vary along spanwise direction. Therefore, from eq. A.9.10, the value of maximum lift coefficient of the wing  $(C_{lmax})_w$  is also varies along spanwise direction.

After calculation, we put the value of spanwise local airfoil maximum lift coefficient  $C_{lmax}$ , the basic lift coefficient,  $C_{lbasic}$ , additional lift coefficient,  $C_{ladd}$ , and maximum lift coefficient of the wing  $(C_{lmax})_w$  into table A.9.1.

| semi-span ft | $C_{lbasic}$ | $C_{ladd}$ | $C_{lmax}$ | $C_{Lmaxw}$ |
|--------------|--------------|------------|------------|-------------|
| 1            | 0.09         | 0.984      | 1.657      | 1.592       |
| 3            | 0.08         | 0.997      | 1.651      | 1.576       |
| 5            | 0.07         | 1.008      | 1.645      | 1.562       |
| 7            | 0.059        | 1.019      | 1.639      | 1.55        |
| 9            | 0.049        | 1.029      | 1.632      | 1.539       |
| 11           | 0.038        | 1.037      | 1.626      | 1.531       |
| 13           | 0.028        | 1.044      | 1.62       | 1.525       |
| 15           | 0.018        | 1.049      | 1.614      | 1.522       |
| 17           | 0.007        | 1.051      | 1.608      | 1.522       |
| 19           | -0.003       | 1.051      | 1.602      | 1.527       |
| 21           | -0.013       | 1.047      | 1.595      | 1.536       |
| 23           | -0.024       | 1.039      | 1.589      | 1.553       |
| 25           | -0.034       | 1.025      | 1.583      | 1.578       |
| 27           | -0.044       | 1.002      | 1.577      | 1.617       |
| 29           | -0.055       | 0.969      | 1.571      | 1.677       |
| 31           | -0.065       | 0.919      | 1.565      | 1.774       |
| 33           | -0.075       | 0.839      | 1.558      | 1.947       |
| 35           | -0.086       | 0.685      | 1.552      | 2.389       |

Table A.9.1 Maximum lift coefficient illustration

According to reference 9, p:475-476, the maximum lift coefficient of the wing  $(C_{Lmax})_w$  may be obtained by using eq. A.9.10. And along the spanwise direction, in table A.9.1, from semi-span, 1 ft is at the wing root and 35 ft is at the wing tip, and the minimum value of  $(C_{Lmax})_w$  in table A.9.1 will become the maximum lift coefficient of the wing  $(C_{Lmax})_w$ .

From table A.9.1, the minimum value of  $C_{Lmax}$ , along semi-span is, at 40 ft, 1.522. Thus, the maximum lift coefficient of the wing  $(C_{Lmax})_w$  is 1.522.

### A.9.2 The aircraft maximum lift coefficient, $C_{Lmax}$

According to reference 6, eq. 8.50, the aircraft maximum lift coefficient may be calculated as:

$$C_{Lmax} = C_{Lmaxw} + C_{L\alpha T} \frac{S_h}{S} \left( \alpha_{CLmax} \left( 1 - \frac{d\epsilon}{d\alpha} - \epsilon_{0T} \right) + i_T \right) \quad (A.9.11)$$

In eq. A.9.11, there are six variables should be given. However, the value of maximum lift coefficient of the wing  $(C_{Lmax})_w$  has been given in section A.9.1, and the average angle of attack of wing airfoil MS(1)-0317 & -0313,  $\alpha_{CLmax}$ , according to reference 28, is 15 degree. The tailplane downwash angle for zero aircraft angle of attack,  $\epsilon_{0T}$ , and incidence angle of tailplane,  $i_T$ , are 0 degrees. From eq. A.7.9, the tailplane lift curve slope,  $C_{L\alpha T}$ , is 4.64. And from eq. A.7.13, the value of the down wash gradient at the horizontal tail,  $\left( \frac{d\epsilon}{d\alpha} \right)$ , is 0.303.

Substituting these variables into eq. A.9.11, then the value of aircraft maximum lift coefficient,  $C_{Lmax}$ , is 1.61.

### A.9.3 Calculate the aircraft stall speed

In order to calculate the stall characteristic of the aircraft, we assumes the aircraft maximum lift coefficient,  $C_{Lmax}$ , keeps constant through the calculation, but the aircraft weight and altitude are changeable. After calculation, by using eq. A.9.2, the stall speed at different altitude and weight are obtained. Table A.9.2 are the results.

| Altitude, ft | Weight, *100 lbs | True $V_{stall}$ , fps |
|--------------|------------------|------------------------|
| 0            | 110              | 130.036                |
| 0            | 90               | 117.622                |
| 0            | 70               | 103.733                |
| 0            | 50               | 87.67                  |

|       |     |         |
|-------|-----|---------|
| 10000 | 110 | 151.32  |
| 10000 | 90  | 136.874 |
| 10000 | 70  | 120.711 |
| 10000 | 50  | 102.02  |
| 20000 | 110 | 178.147 |
| 20000 | 90  | 161.14  |
| 20000 | 70  | 142.112 |
| 20000 | 50  | 120.106 |
| 30000 | 110 | 212.594 |
| 30000 | 90  | 192.299 |
| 30000 | 70  | 169.592 |
| 30000 | 50  | 143.331 |
| 40000 | 110 | 262.088 |
| 40000 | 90  | 237.067 |
| 40000 | 70  | 209.074 |
| 40000 | 50  | 176.7   |
| 50000 | 110 | 333.285 |
| 50000 | 90  | 301.467 |
| 50000 | 70  | 265.869 |
| 50000 | 50  | 224.7   |
| 60000 | 110 | 423.822 |
| 60000 | 90  | 383.362 |
| 60000 | 70  | 338.093 |
| 60000 | 50  | 285.741 |

Table A.9.2 Aircraft stall characteristics

## A.10 Engine simulation analysis

From reference 12, it mentioned that the propulsion system of Teledyne YQM-98A is using Garret ATF3 YF104-GA-100, and Boeing YQM-94A is using GE TF34-GE-100 turbofan engine.

And from reference 12, the basic engine data such as engine diameter, length, weight, mass flow rate, total pressure ratio and bypass ratio is listed. But this data are not detailed enough to analyse aircraft climb and cruise performance.

For calculating the aircraft performance, we still need more detailed engine data such as specific engine fuel consumption and engine net thrust at different altitude and

velocity, in another words - an engine map. But, due to some classified justification, the engine map cannot be acquired.

Lacking real engine data, from reference 12, we collect some available engine data which is relevant to the engine we need, and listed in table A.10.1.

| Engine type | Mass flow<br>lb/s | Total pressure<br>ratio | Bypass<br>ratio    | Net thrust lbs<br>(S.L. static) | Engine weight<br>lbs |
|-------------|-------------------|-------------------------|--------------------|---------------------------------|----------------------|
| YF-GA-100   | 162               | TO :21<br>Cruise : 25   | 2.8 at<br>take-off | 5050                            | 900                  |
| TF34-GE-100 | ≈ 330             | 21                      | 6.2                | 9065                            | 1440                 |

Table A.10.1 The engine data

According to table A.10.1, we select the similar engine data which is the mass flow rate, 130 lb/s, the total pressure ratio, 16, the bypass ratio, 3.28, the net thrust at sea level, 5330, and the engine weight, 1000 lbs.

Apart from above assumed engine data, we didn't have further detailed engine map. In order to gain approximate engine data, according to reference 13, we tried to use different combination of fan and compressor pressure ratio and engine inlet temperature, TET, to run TURBOMATCH program. Finally, we find one data set that is more close to what we need, and thus, use it to simulate aircraft performance.

The following steps are the process of simulating different combination of engine parameters.

First, the total pressure ratio,  $P_t$  , and mass flow rate, are fixed. Then, under the different turbine inlet temperature, TET, goes with different fan and compressor pressure. Substituting into the TURBOMATCH program, After simulating, we obtain the results as in table A.10.2.

|                 | TET = 1480 °F, M=130 lb/s |        |        | TET=1500°F, M=130 lb/s |       |        |
|-----------------|---------------------------|--------|--------|------------------------|-------|--------|
| Fan Pressure    | 1.6                       | 1.8    | 2      | 1.6                    | 1.8   | 2      |
| Compressor      | 10                        | 8.889  | 8      | 10                     | 8.889 | 8      |
| sfc, lb/lb-h    | 0.492                     | 0.477  | 0.472  | 0.5                    | 0.484 | 0.478  |
| net thrust, lbs | 4984.2                    | 5135.3 | 5186.4 | 5039.3                 | 5200  | 5263.6 |

(a) Mass flow rate & total pressure is 130 lb/s and 16 respectively,  
with engine inlet temperature (TET) varies.



|                 | TET = 1480 °F, M=130 lb/s |        |        | TET=1500°F, M=130 lb/s |       |        |
|-----------------|---------------------------|--------|--------|------------------------|-------|--------|
| Fan Pressure    | 1.6                       | 1.8    | 2      | 1.6                    | 1.8   | 2      |
| Compressor      | 11.25                     | 10     | 9      | 11.25                  | 10    | 9      |
| sfc, lb/lb-h    | 0.48                      | 0.465  | 0.46   | 0.488                  | 0.472 | 0.466  |
| net thrust, lbs | 4975.7                    | 5126.3 | 5175.3 | 5032.5                 | 5193  | 5255.5 |

(b) Mass flow rate & total pressure is 130 lb/s and 18 respectively,  
with engine inlet temperature (TET) varies.

|                 | TET = 1480 °F, M=130 lb/s |        |        | TET=1500°F, M=130 lb/s |        |        |
|-----------------|---------------------------|--------|--------|------------------------|--------|--------|
| Fan Pressure    | 1.6                       | 1.8    | 2      | 1.6                    | 1.8    | 2      |
| Compressor      | 13.125                    | 11.667 | 10.5   | 13.125                 | 11.667 | 10.5   |
| sfc, lb/lb-h    | 0.464                     | 0.45   | 0.445  | 0.472                  | 0.457  | 0.451  |
| net thrust, lbs | 4953.4                    | 5100   | 5144.4 | 5102.9                 | 5171.9 | 5229.5 |

( c ) Mass flow rate & total pressure is 130 lb/s and 21 respectively,  
with engine inlet temperature (TET) varies.

|                 | TET = 1480 °F, M=140 lb/s |        |        | TET=1500°F, M=140 lb/s |        |        |
|-----------------|---------------------------|--------|--------|------------------------|--------|--------|
| Fan Pressure    | 1.6                       | 1.8    | 2      | 1.6                    | 1.8    | 2      |
| Compressor      | 13.125                    | 11.667 | 10.5   | 13.125                 | 11.667 | 10.5   |
| sfc, lb/lb-h    | 0.464                     | 0.45   | 0.445  | 0.472                  | 0.457  | 0.451  |
| net thrust, lbs | 5301.8                    | 5459.7 | 5506.2 | 5365.5                 | 5535.7 | 5597.3 |

( d ) Mass flow rate & total pressure is 140 lb/s and 21 respectively,  
with engine inlet temperature (TET) varies.

Table A.10.2 Engine design point results

In table A.10.2 ( a ), ( b ), ( c ), we found that if the total pressure ratio remains the same, when the fan pressure ratio goes up, the compressor pressure goes down, the specific fuel consumption will go down, and the static net thrust will go up. From this point of view, it seems good to have higher value of fan pressure. But, for practical problem, the single stage fan pressure ratio should not over certain value.

Besides, in table A.10.2, we also found that with the same total pressure ratio. The higher turbine inlet temperature the higher net thrust we will get, but the specific fuel consumption will increase. Since the HALE is going to stay at high altitude for a long time, so the fuel consumption should keep as low as possible.

After comparing these advantage and drawbacks from table A.10.2 ( a ), ( b ), ( c ), we decide to change another engine parameter, mass flow rate, in table A.10.2, ( d ), the

mass flow rate is 140 lb/s. From the results, it is easy to find the specific fuel consumption is the same with table A.10.2 ( c ), but the net thrust is increased a lot.

Finally, when referred to table A.10.1, the total pressure ratio of those two engines are higher than 21. Therefore, we decide to chose the one in table A.10.2, ( d ), the total pressure ratio is 21 - the fan pressure ratio is 1.8, and compressor pressure ratio is 11.667, and the engine inlet temperature is 1480°F to do engine simulation.

After simulating, the results we obtained is from design point sea-level, every 2,000 ft, to off design point 66,000 ft. Since the results is too large, we only pick some part of the engine data, table A.10.3.

| Height, ft | Mach number | Fuel flow, lb/sec | Thrust, lbf |
|------------|-------------|-------------------|-------------|
| 0          | 0           | 0.6818            | 5459.664    |
| 0          | 0.3         | 0.695             | 4191.979    |
| 6000       | 0.25        | 0.6057            | 4066.895    |
| 6000       | 0.35        | 0.619             | 3856.983    |
| 6000       | 0.4         | 0.6267            | 3772.021    |
| 12000      | 0.3         | 0.5208            | 3450.349    |
| 12000      | 0.35        | 0.5276            | 3381.736    |
| 12000      | 0.45        | 0.5433            | 3262.464    |
| 18000      | 0.3         | 0.4485            | 2833.401    |
| 18000      | 0.4         | 0.4598            | 2725.116    |
| 18000      | 0.5         | 0.4742            | 2589.312    |
| 24000      | 0.3         | 0.3744            | 2443.811    |
| 24000      | 0.4         | 0.3843            | 2358.362    |
| 24000      | 0.5         | 0.3975            | 2296.305    |
| 30000      | 0.3         | 0.3149            | 2005.947    |
| 30000      | 0.4         | 0.3238            | 1940.548    |
| 30000      | 0.5         | 0.3353            | 1883.969    |
| 36000      | 0.4         | 0.2686            | 1602.268    |
| 36000      | 0.5         | 0.2783            | 1555.373    |
| 36000      | 0.6         | 0.2904            | 1513.119    |
| 42000      | 0.4         | 0.2016            | 1200.944    |
| 42000      | 0.5         | 0.2084            | 1171.573    |
| 42000      | 0.6         | 0.2179            | 1136.019    |
| 48000      | 0.4         | 0.1511            | 900.0677    |
| 48000      | 0.5         | 0.1562            | 877.9723    |
| 48000      | 0.6         | 0.1633            | 851.2068    |
| 54000      | 0.45        | 0.1151            | 665.8903    |

|       |      |        |          |
|-------|------|--------|----------|
| 54000 | 0.55 | 0.1197 | 647.5724 |
| 54000 | 0.65 | 0.1253 | 627.1116 |
| 60000 | 0.45 | 0.0863 | 498.901  |
| 60000 | 0.55 | 0.0897 | 485.2549 |
| 60000 | 0.65 | 0.0939 | 469.8761 |
| 66000 | 0.45 | 0.0647 | 373.8573 |
| 66000 | 0.55 | 0.0672 | 363.7059 |
| 66000 | 0.65 | 0.0704 | 352.0576 |

Table A.10.3 Engine simulation results

## A.11 Climb performance

In section A.10, the engine data at different altitude and speed, such as the fuel flow, specific fuel consumption and net thrust, has been obtained. Thus, according to reference 15, p:268, if the climb speed is given as well, then the time, fuel and distance to climb can be calculated.

From the mission specification, after take-off, the HALE will climb from sea level to cruise altitude. According to reference 15, at this stage the rate of climb, as a function of altitude, should be determined. But how long should the HALE climb from sea level to cruise altitude? And what's the climb speed? No previous data was available. So, we had to assume a proper climb velocity.

In order to chose a proper climb velocity, according to reference 66, there are two requirements that should follow. First, no less than aircraft stall speed. Second, no less than minimum aircraft drag speed. Based on these two restrictions. We divide the climbing altitude into four segments, and in each segment we chose a constant indicated speed to climb, that is from sea-level to 22,000 ft, speed is 338 fps, from 22,000 to 34,000 ft, speed is 270.4 fps, from 34,000 to 44,000 ft, speed is 236.6 fps, and from 44,000 to 52,000 ft, speed is 210 fps. In order to assure, in each segment, the climb velocity is higher than the stall speed and minimum drag velocity, we verified as below.

### a. The stall speed

From section A.9, table A.9.2, the aircraft stall speed, at different weight and altitude, is given.

### b. The minimum drag speed

From reference 37, p:220 to 226. According to the definition of minimum drag velocity, it is the velocity which zero-lift drag and induced drag are equal.

$$C_{D0} = C_{Di} \quad (A.11.1)$$

Following is the pseudo program to calculate the minimum drag velocity, at every 2,000 ft (from 2,000 to 52,000 ft). Assume the aircraft weight is 10,000 lbs.

```

w = 10000.
do i = 0 , 26
  h = 2000 * i
  do j = 0 , 300
    v = 150 + j
    
$$C_i = \frac{2w}{\rho v^2 S}$$

    call drag subroutine
    a = Cdi
    b = Cd0
    diff = abs ( Cdi - Cd0 )
    if ( diff .LT. 0.0002) then
      Vmin = v
    else
      continue
    end if
  end do
end do

```

After calculation, the results are presented at table A.11.1.

| Altitude, ft | CD0    | CDL    | V <sub>dmin</sub> , fps | V <sub>e-climb</sub> , fps | V <sub>t-climb</sub> , fps |
|--------------|--------|--------|-------------------------|----------------------------|----------------------------|
| 0            | 0.0125 | 0.0125 | 195                     | 338                        | 338                        |
| 2000         | 0.0126 | 0.0124 | 201.19                  | 338                        | 348.18                     |
| 4000         | 0.0127 | 0.0126 | 206.56                  | 338                        | 358.66                     |
| 6000         | 0.0128 | 0.0126 | 213.13                  | 338                        | 369.69                     |
| 8000         | 0.0129 | 0.0128 | 218.91                  | 338                        | 381.22                     |
| 10000        | 0.0129 | 0.0127 | 225.91                  | 338                        | 393.29                     |
| 12000        | 0.013  | 0.013  | 232.15                  | 338                        | 405.92                     |
| 14000        | 0.0131 | 0.013  | 239.65                  | 338                        | 419.17                     |
| 16000        | 0.0132 | 0.013  | 247.40                  | 338                        | 433.05                     |
| 18000        | 0.0133 | 0.0131 | 255.45                  | 338                        | 447.66                     |
| 20000        | 0.0132 | 0.0132 | 263.80                  | 338                        | 462.95                     |
| 22000        | 0.0133 | 0.0133 | 272.47                  | 270.4                      | 383.12                     |
| 24000        | 0.0135 | 0.0134 | 281.49                  | 270.4                      | 396.71                     |
| 26000        | 0.0136 | 0.0135 | 290.88                  | 270.4                      | 410.92                     |
| 28000        | 0.0137 | 0.0137 | 300.67                  | 270.4                      | 425.92                     |

|       |        |        |        |       |        |
|-------|--------|--------|--------|-------|--------|
| 30000 | 0.0138 | 0.0137 | 311.88 | 270.4 | 441.64 |
| 32000 | 0.0139 | 0.0139 | 322.56 | 270.4 | 458.42 |
| 34000 | 0.014  | 0.0139 | 334.72 | 236.6 | 416.52 |
| 36000 | 0.0142 | 0.0142 | 346.41 | 236.6 | 432.84 |
| 38000 | 0.0144 | 0.0142 | 362.93 | 236.6 | 453.84 |
| 40000 | 0.0146 | 0.0145 | 379.17 | 236.6 | 476.10 |
| 42000 | 0.0148 | 0.0147 | 396.06 | 236.6 | 499.64 |
| 44000 | 0.015  | 0.0149 | 414.64 | 210   | 464.94 |
| 46000 | 0.0152 | 0.0151 | 432.94 | 210   | 487.53 |
| 48000 | 0.0154 | 0.0153 | 452.98 | 210   | 511.91 |
| 50000 | 0.0156 | 0.0155 | 473.82 | 210   | 536.63 |
| 52000 | 0.0156 | 0.0155 | 497.49 | 210   | 563.00 |

Table A.11.1 The minimum drag velocity VS climb velocity

By comparing these figures, in table A.11.1 and table A.9.2, the climb velocity is higher than the minimum drag speed, and the stall velocity. Thus, the climb speed meets our requirement.

### A.11.1 The climb segment

When the climb velocity is established, the following calculates the climb performance. From reference 15, eq. 15.18, if the aircraft climb with true airspeed, the aircraft rate of climb may be calculated.

$$RC = \frac{V_{cl}(T - D)}{W} \quad (A.11.2)$$

However, if the aircraft climb with equivalent airspeed, there is a factor, kinetic energy correction factor that should be included. Thus, from reference 15, eq. 15.21, the rate of climb, RC, below 36150 ft and above 36150 ft, may be modified by the following equations.

$$RC = \frac{V_{cl}(T - D)}{W} \frac{1}{1 + 0.567M^2} \quad (A.11.3)$$

$$RC = \frac{V_{cl}(T - D)}{W} \frac{1}{1 + 0.7M^2} \quad (A.11.4)$$

In this case, during climb, we convert equivalent airspeed into true airspeed. Therefore, eq. A.11.2 is used for calculating the rate of climb.

Reference 15, eq. 15.23, the time to climb,  $T_{cl}$ , may be computed by

$$T_{cl} = \sum_i^n \frac{\Delta h}{RC} \quad (A.11.5)$$

As mentioned before, the rate of climb is calculated at 2,000 ft intervals, from sea level to 52,000 ft. According to this, the total time to climb may be obtained.

After the value of the rate of climb and the time to climb in each segment is obtained, then from reference 15, eq. 15.24, the fuel to climb, FC, may be calculated.

$$FC = F_f T_{cl} \quad (\text{A.11.6})$$

Finally, from reference 15, eq. 15.25, the distance to climb, DC, may be calculated.

$$DC = \frac{V_{cl}}{RC} 2000 \quad (\text{A.11.7})$$

After calculation, we put the rate of climb, the time to climb, the fuel to climb and the distance to climb into table A.11.2.

| Altitude<br>,ft | Rate of Climb<br>, fps | Fuel consump.<br>, lb/sec | Time to<br>climb, sec | Distance to<br>climb, nm |
|-----------------|------------------------|---------------------------|-----------------------|--------------------------|
| 2000            | 110.72                 | 12.55                     | 18.06                 | 1.00                     |
| 4000            | 109.91                 | 12.15                     | 18.20                 | 1.04                     |
| 6000            | 114.24                 | 11.33                     | 17.51                 | 1.03                     |
| 8000            | 110.61                 | 11.16                     | 18.08                 | 1.10                     |
| 10000           | 111.31                 | 10.57                     | 17.97                 | 1.13                     |
| 12000           | 107.52                 | 10.40                     | 18.60                 | 1.20                     |
| 14000           | 104.09                 | 10.21                     | 19.22                 | 1.28                     |
| 16000           | 100.50                 | 10.05                     | 19.90                 | 1.37                     |
| 18000           | 96.86                  | 9.91                      | 20.65                 | 1.47                     |
| 20000           | 85.95                  | 10.80                     | 23.27                 | 1.71                     |
| 22000           | 82.82                  | 10.64                     | 24.15                 | 1.84                     |
| 24000           | 74.30                  | 10.92                     | 26.92                 | 1.70                     |
| 26000           | 71.57                  | 10.70                     | 27.94                 | 1.82                     |
| 28000           | 65.23                  | 11.31                     | 30.66                 | 2.07                     |
| 30000           | 62.78                  | 11.09                     | 31.86                 | 2.23                     |
| 32000           | 60.28                  | 10.90                     | 33.18                 | 2.41                     |
| 34000           | 54.62                  | 11.53                     | 36.61                 | 2.76                     |
| 36000           | 52.12                  | 11.38                     | 38.37                 | 3.01                     |
| 38000           | 47.81                  | 11.41                     | 41.83                 | 2.98                     |
| 40000           | 43.08                  | 11.60                     | 46.42                 | 3.47                     |
| 42000           | 38.69                  | 11.81                     | 51.69                 | 4.05                     |
| 44000           | 34.31                  | 12.22                     | 58.29                 | 4.80                     |
| 46000           | 29.91                  | 12.92                     | 66.87                 | 5.77                     |

|       |       |       |        |      |
|-------|-------|-------|--------|------|
| 48000 | 25.47 | 13.54 | 78.52  | 6.31 |
| 50000 | 21.83 | 14.51 | 91.60  | 7.73 |
| 52000 | 18.25 | 15.93 | 109.57 | 9.70 |

Table A.11.2 The climb performance

From table A.11.2, the rate of climb decreases, the time to climb increases and the fuel consumption increases with altitude. This is due to the engine net thrust decrease when the air density reduces.

After calculation, the aircraft takes about 17 minutes to climb from sea level to 52000 ft, consume about 301 lbs fuel and the climb distance about 75 nautical miles.

During climb, the aircraft component drag is calculated and listed in table A.11.3. From this table, it is not difficult to tell that the wing drag coefficient is the largest compared to all component drag coefficient. And by analysing the wing drag coefficient, from the beginning, the zero drag coefficient is higher than the drag coefficient due to lift. However, with the altitude and higher velocity, the drag coefficient due to lift keeps increasing, and eventually becomes greater than the zero drag coefficient.

| Alt., ft | $C_{d0w}$ | $C_{diw}$ | $C_{d0f}$ | $C_{d0e}$ | $C_{die}$ | $C_{d0n}$ | $C_{Dn}$ | $C_D$  |
|----------|-----------|-----------|-----------|-----------|-----------|-----------|----------|--------|
| 2000     | 0.0071    | 0.0017    | 0.0019    | 0.0018    | 0.0016    | 0.0011    | 0.0007   | 0.0160 |
| 4000     | 0.0072    | 0.0017    | 0.0019    | 0.0018    | 0.0016    | 0.0011    | 0.0007   | 0.0161 |
| 6000     | 0.0072    | 0.0017    | 0.0019    | 0.0018    | 0.0016    | 0.0011    | 0.0007   | 0.0162 |
| 8000     | 0.0073    | 0.0017    | 0.0019    | 0.0018    | 0.0016    | 0.0011    | 0.0007   | 0.0163 |
| 10000    | 0.0053    | 0.0017    | 0.0019    | 0.0018    | 0.0016    | 0.0011    | 0.0007   | 0.0144 |
| 12000    | 0.0054    | 0.0017    | 0.0020    | 0.0018    | 0.0016    | 0.0011    | 0.0007   | 0.0145 |
| 14000    | 0.0054    | 0.0017    | 0.0020    | 0.0018    | 0.0017    | 0.0011    | 0.0007   | 0.0146 |
| 16000    | 0.0055    | 0.0017    | 0.0020    | 0.0019    | 0.0017    | 0.0011    | 0.0007   | 0.0147 |
| 18000    | 0.0056    | 0.0017    | 0.0020    | 0.0019    | 0.0017    | 0.0011    | 0.0007   | 0.0148 |
| 20000    | 0.0056    | 0.0017    | 0.0020    | 0.0019    | 0.0017    | 0.0011    | 0.0007   | 0.0149 |
| 22000    | 0.0057    | 0.0017    | 0.0020    | 0.0019    | 0.0017    | 0.0011    | 0.0007   | 0.0151 |
| 24000    | 0.0059    | 0.0038    | 0.0021    | 0.0020    | 0.0010    | 0.0012    | 0.0007   | 0.0167 |
| 26000    | 0.0060    | 0.0038    | 0.0021    | 0.0020    | 0.0011    | 0.0012    | 0.0007   | 0.0169 |
| 28000    | 0.0060    | 0.0038    | 0.0021    | 0.0020    | 0.0011    | 0.0012    | 0.0007   | 0.0170 |
| 30000    | 0.0061    | 0.0038    | 0.0022    | 0.0020    | 0.0011    | 0.0012    | 0.0007   | 0.0171 |
| 32000    | 0.0062    | 0.0038    | 0.0022    | 0.0020    | 0.0011    | 0.0012    | 0.0007   | 0.0173 |
| 34000    | 0.0063    | 0.0038    | 0.0022    | 0.0020    | 0.0011    | 0.0012    | 0.0007   | 0.0174 |

|       |        |        |        |        |        |        |        |        |
|-------|--------|--------|--------|--------|--------|--------|--------|--------|
| 36000 | 0.0064 | 0.0038 | 0.0022 | 0.0021 | 0.0012 | 0.0012 | 0.0007 | 0.0176 |
| 38000 | 0.0065 | 0.0062 | 0.0023 | 0.0021 | 0.0007 | 0.0013 | 0.0007 | 0.0199 |
| 40000 | 0.0067 | 0.0062 | 0.0023 | 0.0021 | 0.0007 | 0.0013 | 0.0007 | 0.0201 |
| 42000 | 0.0068 | 0.0062 | 0.0023 | 0.0022 | 0.0008 | 0.0013 | 0.0007 | 0.0203 |
| 44000 | 0.0069 | 0.0062 | 0.0024 | 0.0022 | 0.0007 | 0.0013 | 0.0007 | 0.0204 |
| 46000 | 0.0070 | 0.0062 | 0.0024 | 0.0022 | 0.0007 | 0.0013 | 0.0007 | 0.0206 |
| 48000 | 0.0073 | 0.0098 | 0.0025 | 0.0023 | 0.0004 | 0.0013 | 0.0007 | 0.0242 |
| 50000 | 0.0074 | 0.0098 | 0.0025 | 0.0023 | 0.0004 | 0.0013 | 0.0007 | 0.0244 |
| 52000 | 0.0075 | 0.0098 | 0.0025 | 0.0024 | 0.0004 | 0.0014 | 0.0007 | 0.0246 |

Table A.11.3 Component drag during climbing (value less than 0.0002, is ignored)

### A.11.2 The minimum drag velocity

The aircraft reaches 52,000 ft, from previous calculation, the fuel consumption is given, thus, the actual weight may be obtained. From section A.11, the minimum drag speed may be calculated from following pseudo program.

```

w = 10152.7
h = 52000.
do j = 0 , 300
    v = 150 + j
    
$$C_l = \frac{2w}{\rho v^2 S}$$

    call drag subroutine
    a = Cdi
    b = Cd0
    diff = abs ( Cdi - Cd0 )
    if ( diff .LT. 0.0002) then
        vmin = v
    else
        continue
    end if
end do

```

After calculation, the minimum drag speed, at 52,000 ft steady-state level flight, is 508.3 fps(true speed).

Substituting the minimum drag speed and aircraft weight into section A.8.1, the drag subroutine. After calculation, the lift drag ratio is 23.7.



## **A.12 Cruise performance**

According to reference 39, 40, there are three approximate methods for estimation of cruise range and endurance.

**a. Cruise air range at constant lift drag ratio.**

**b. Cruise at constant V or M.**

**c. Cruise air range at constant pressure height assuming a parabolic drag law.**

According to the mission listed in fig 2.1.1, the cruise segment of the HALE is 24 hours cruise-climb from 52,000 ft to 65,000 ft. Therefore, in this segment, first the HALE takes the method, cruise-climb, until the altitude reaches 65,000 ft, then we compare this period of time with 24 hours. If it already spends 24 hours then we say the cruise is finished. Otherwise, the remaining hours will be loitered at 65,000 ft.

In following paragraph, we will calculate the cruise performance.

### **A.12.1 Cruise-climb**

According to reference 39, the cruise-climb method is of limited operational use. But this cruise techniques provides for operating at or near the maximum range capability of the aeroplane.

The best range of this method is to cruise at aircraft minimum drag speed. For HALE at 52,000 ft, as calculated in section A.11.2, the minimum drag speed is 508.3 fps (0.525 M). From the HALE's specification, the cruise speed is within 0.5 to 0.6 M. Obviously meets our requirement.

The condition of flight at cruise-climb segment, according to reference 39, is the lift drag ratio and velocity should keep constant. Then when the weight reduces, the lift coefficient and aircraft velocity are still the same. So, the aircraft altitude will automatically rise.

Although the minimum drag velocity is the best speed for endurance flight, from reference 30, p:326 to 328, if we fly at the minimum drag speed, then any decrease in speed could cause increase drag. Therefore, at speeds less than that for minimum drag, a turbo-jet aircraft suffers an instability of speed.

Thus in this calculation, we multiply the minimum drag speed by 1.03, then if any disturbance occur, the aircraft will not suffer speed instability. Therefore, the aircraft cruise-climb speed is 523.5 fps rather than the minimum drag speed.

Thus in this calculation, the minimum drag speed is multiplied by 1.03, in case if any disturbance occur, the aircraft will not suffer speed instability. Therefore, the aircraft cruise-climb speed is 523.5 fps rather than the minimum drag speed.

Following is the pseudo program about how to calculate during cruise-climb.

$$h = 52000.$$

$$w = 10152.7$$

$$v = 508.3 * 1.03$$

$$C_l = \frac{2W}{\rho V^2 S}$$

call drag subroutine to compute drag coefficient

$$L/D = \frac{C_L}{C_D}$$

do I = 1 , 24

$$D = \frac{W}{L/D}$$

call thrust subroutine to calculate net thrust ( $T_{net}$ )

call sfc subroutine to calculate specific fuel consumption

if ( $T_{net} \cdot GT \cdot D$ ) then \* compare  $T_{net}$  and D

continue

else

end of calculation

end if

$$sfc_{od} = sfc \left( 1 + 0.1 \left( \frac{T_{net}}{D} - 1 \right) \right)$$

$$f_{uel} = sfc_{od} D$$

$$W = W - f_{uel}$$

$$\rho = \frac{2W}{C_l V^2 S}$$

\*due to weight is reduced, and  $C_l$ , V and S is the same. So, the density will decrease, its means the altitude will increase.

$$h = 6341.6184 * -\log\left(\frac{\rho}{\rho_{SL}}\right) + \log(0.297076) + 11000$$

if ( h .lt. 64700 ) then

continue

else

go out of do loop

end if

end do

After cruise-climb for 21 hours, the HALE reaches to the altitude of 65,148 ft, so, there is 3 hours left for cruise at 65,000 ft.

We put the results of cruise climb in table A.12.1.

| Hours | Weight, lbs | Altitude, ft | Hours | Weight, lbs | Altitude, ft |
|-------|-------------|--------------|-------|-------------|--------------|
| 0     | 10193.3     | 52000        | 11    | 7322.199    | 58882.99     |
| 1     | 9891.362    | 52625.61     | 12    | 7105.028    | 59509.41     |
| 2     | 9598.318    | 53251.32     | 13    | 6894.271    | 60135.91     |
| 3     | 9313.945    | 53877.06     | 14    | 6689.72     | 60762.56     |
| 4     | 9038.028    | 54502.73     | 15    | 6491.046    | 61389.82     |
| 5     | 8770.294    | 55128.37     | 16    | 6298.104    | 62017.64     |
| 6     | 8510.524    | 55753.94     | 17    | 6110.761    | 62645.92     |
| 7     | 8258.518    | 56379.33     | 18    | 5929.288    | 63273.15     |
| 8     | 8013.927    | 57004.84     | 19    | 5753.537    | 63899.19     |
| 9     | 7776.463    | 57630.67     | 20    | 5583.363    | 64523.85     |
| 10    | 7545.949    | 58256.73     | 21    | 5418.377    | 65147.92     |

Table A.12.1 The cruise-climb performance

During cruise-climb, the aircraft drag component distribution is listed in table A.12.2 (for simplicity, the component drag coefficient less than 0.0001, not listed).

| Altitude, ft | $C_{Dow}$ | $C_{Diw}$ | $C_{Dof}$ | $C_{Doe}$ | $C_{Don}$ | $C_{Dn}$ | $C_D$  |
|--------------|-----------|-----------|-----------|-----------|-----------|----------|--------|
| 52000        | 0.0077    | 0.0131    | 0.0026    | 0.0024    | 0.0014    | 0.0007   | 0.0279 |
| 52625.6      | 0.0077    | 0.0131    | 0.0026    | 0.0024    | 0.0014    | 0.0007   | 0.0279 |
| 53251.3      | 0.0077    | 0.0131    | 0.0026    | 0.0024    | 0.0014    | 0.0007   | 0.0280 |
| 53877.1      | 0.0076    | 0.0131    | 0.0025    | 0.0024    | 0.0014    | 0.0007   | 0.0279 |
| 54502.7      | 0.0077    | 0.0131    | 0.0026    | 0.0025    | 0.0014    | 0.0007   | 0.0279 |
| 55128.4      | 0.0077    | 0.0131    | 0.0026    | 0.0025    | 0.0014    | 0.0007   | 0.0280 |
| 55753.9      | 0.0078    | 0.0131    | 0.0026    | 0.0025    | 0.0014    | 0.0007   | 0.0281 |
| 56379.3      | 0.0078    | 0.0131    | 0.0026    | 0.0025    | 0.0014    | 0.0007   | 0.0281 |
| 57004.8      | 0.0078    | 0.0131    | 0.0026    | 0.0025    | 0.0014    | 0.0007   | 0.0282 |
| 57630.7      | 0.0079    | 0.0131    | 0.0026    | 0.0025    | 0.0014    | 0.0007   | 0.0283 |
| 58256.7      | 0.0079    | 0.0131    | 0.0026    | 0.0026    | 0.0014    | 0.0007   | 0.0284 |
| 58883.       | 0.0079    | 0.0131    | 0.0026    | 0.0026    | 0.0014    | 0.0007   | 0.0285 |
| 59509.4      | 0.0080    | 0.0131    | 0.0026    | 0.0026    | 0.0015    | 0.0007   | 0.0285 |
| 60135.9      | 0.0080    | 0.0131    | 0.0026    | 0.0026    | 0.0015    | 0.0007   | 0.0286 |

|          |        |        |        |        |        |        |        |
|----------|--------|--------|--------|--------|--------|--------|--------|
| 60762.6  | 0.0081 | 0.0131 | 0.0027 | 0.0026 | 0.0015 | 0.0007 | 0.0287 |
| 61389.8  | 0.0081 | 0.0131 | 0.0027 | 0.0026 | 0.0015 | 0.0007 | 0.0288 |
| 62017.6  | 0.0082 | 0.0131 | 0.0027 | 0.0026 | 0.0015 | 0.0007 | 0.0289 |
| 62645.9  | 0.0082 | 0.0131 | 0.0027 | 0.0027 | 0.0015 | 0.0007 | 0.0290 |
| 63273.2  | 0.0082 | 0.0131 | 0.0027 | 0.0027 | 0.0015 | 0.0007 | 0.0291 |
| 63899.19 | 0.0083 | 0.0131 | 0.0027 | 0.0027 | 0.0015 | 0.0007 | 0.0292 |
| 64523.85 | 0.0083 | 0.0131 | 0.0027 | 0.0027 | 0.0015 | 0.0007 | 0.0293 |
| 65147.92 | 0.0084 | 0.0131 | 0.0027 | 0.0027 | 0.0015 | 0.0007 | 0.0294 |

Table A.12.2 The component drag distribution during cruise-climb segment

### A.12.2 Cruise at constant V or M

When the altitude has reached 65,000 ft, following the requirement, there is 24 hours for cruise. From previous section, there is 21 hours spent, therefore, there are 3 hours difference from our cruise requirement. At this paragraph, we select the method of cruising at constant V or M, and flight at the same altitude, 65,000 ft, to cruise for another 3 hours.

Following is the pseudo program to calculate the constant pressure altitude cruise.

h = 65000.

w = 5402.

v = 508.3\*1.03

do i = 1, 3

$$C_i = \frac{2W}{\rho V^2 S}$$

call drag subroutine to compute drag coefficient

$$L/D = \frac{C_L}{C_D}$$

$$D = \frac{W}{L/D}$$

call thrust subroutine to calculate net thrust ( $T_{net}$ )

call sfc subroutine to calculate specific fuel consumption

if ( $T_{net} > D$ ) then \* compare  $T_{net}$  and D

continue

else

```

end of calculation
end if

$$sfc_{od} = sfc \left( 1 + 0.1 \left( \frac{T_{net}}{D} - 1 \right) \right)$$


$$f_{uel} = sfc_{od} D$$


$$W = W - f_{uel}$$

end do

```

After calculation, the results are presented into table A.12.3.

| Hours | Weight, lbs | Altitude, ft |
|-------|-------------|--------------|
| 22    | 5250.784    | 65000        |
| 23    | 5086.908    | 65000        |
| 24    | 4926.544    | 65000        |

Table A.12.3 The cruise performance

As we did in previous section, we put the aircraft drag component distribution in cruise listed in table A.12.4 ( in order not to be complex, the component drag coefficient less than 0.0001, not listed).

| Altitude, ft | $C_{D0w}$ | $C_{Diw}$ | $C_{D0f}$ | $C_{D0e}$ | $C_{Die}$ | $C_{D0n}$ | $C_{Dn}$ | $C_D$  |
|--------------|-----------|-----------|-----------|-----------|-----------|-----------|----------|--------|
| 65000        | 0.0084    | 0.0131    | 0.0027    | 0.0027    | 0.0002    | 0.0015    | 0.0007   | 0.0294 |
| 65000        | 0.0084    | 0.0123    | 0.0027    | 0.0027    | 0.0002    | 0.0015    | 0.0007   | 0.0287 |
| 65000        | 0.0084    | 0.0116    | 0.0027    | 0.0027    | 0.0002    | 0.0015    | 0.0007   | 0.0280 |

Table A.12.4 The component drag distribution during cruise segment

## A.13 Cruise return and descend to sea level

At the end of cruise, following is cruise return from 65,000 ft to 52,000 ft. According to the mission requirement, during this segment the return cruise distance is 500 nautical miles. Assuming the aircraft descend 13,000 ft in 2 hours, equivalent to the descend rate as 1.8 fps. Assuming at first hour the aircraft speed is 0.54M and the second hour reduces to 0.5 M, with the descend rate as 1.8 fps, after calculation, the results are presented at table A.13.1.

| Hours | Weight, lbs | Height, ft | $L/D_{\max}$ | Rate of descend, fps | Distance, nm |
|-------|-------------|------------|--------------|----------------------|--------------|
| 1     | 4798.07     | 58176.35   | 22.18        | -1.9                 | 309.98       |
| 2     | 4663.55     | 51608.37   | 19.93        | -1.82                | 606.11       |

Table A.13.1 Cruise return performance

Following is descending from 51,608 ft to sea level. Assuming the descend segment is the same as in climb, first, from 51,608 to 44,000 ft, with constant airspeed 0.5 M. Second, from 44,000 to 34,000 ft, with constant airspeed 0.4 M. Third, from 34,000 to 22,000, with constant airspeed 0.3 M. Fourth, from 22,000 ft to sea-level with constant airspeed 0.25 M.

According to engine source data, at same speed and same altitude, the fuel flow at thrittle idle is one seventh the fuel flow at maximum throttle. Therefore, by taking the average fuel flow at each segment and divided by 7, and multiply the descend time. After calculation, in each segment the fuel consumption is 5.55, 8.15, 11.87 and 18.7 lbs respectively. Therefore, the total fuel consumption at descending is 44.28 lbs, and the aircraft weight becomes 4619.27 lbs.

## A.14 The aircraft weight comparison

So far, the aircraft has finished the “mission” calculation, and from the previous section, the final aircraft weight is obtained.

From section A.5.12, the aircraft empty weight is 4343.5 lbs, and the weight of payload is 200 lbs. So, the empty weight plus payload should be 4543.5 lbs. The final aircraft weight subtract the aircraft empty weight plus payload weight, the aircraft still has more than 75.77 lbs fuel. That means we may reduce some fuel in the beginning. Therefore, we go back into an iteration to reduce the fuel and calculation again, until it meet our requirement, the weight difference is within 10 pounds.

Following is the pseudo program showing us how to do this calculation.

```

aircraft take-off weight estimation
go to 2
1   W = W ± dif
2   •   * go through the whole process, but the wing
    •   body incidence.
    •
    W = W24   * after 24 hours cruise

```

```

W1 = Wemp + Wpay
dif = W - W1
if ( abs(dif) .lt. 10 ) then
    "this is our answer"
else
    go to 1
end if

```

| Wing load | W <sub>TO</sub> , lbs | W <sub>final</sub> , lbs | W <sub>emp</sub> +W <sub>pay</sub> , lbs | Fuel diff., lbs |
|-----------|-----------------------|--------------------------|------------------------------------------|-----------------|
| 30        | 10489.71              | 4619.27                  | 4543.47                                  | -75.8           |
| 30        | 10413.9               | 4588.18                  | 4543.47                                  | -44.71          |
| 30        | 10369.2               | 4564.52                  | 4543.47                                  | -21.05          |
| 30        | 10348.2               | 4554.01                  | 4543.47                                  | -10.54          |
| 30        | 10337.6               | 4547.8                   | 4543.47                                  | -4.33           |

Table A.14.1 The fuel difference calculation

Table A.14.1, is the results, after 5 iterations, finally the weight difference is 4.33 lbs, therefore, meets the requirement. The mission fuel weight, from first iteration to the end, is reduced around 152.11 lbs, and meets the mission requirement.

In table A.14.1, the negative sign in the fuel difference column means that the fuel is more than required, therefore, at the next iteration the excess fuel will be deducted.

## A.15 Take-off performance

Take-off runway distance includes take-off field length and take-off ground length.

### A.15.1 Take-off field length

From reference 7, eq. 5.6, the take-off field length, S<sub>TO</sub>, may be calculated as

$$S_{TO} = f_{TO} h_{TO} \left( \frac{1}{\gamma_{LOF}} + \frac{\left( \frac{V_3}{V_{STO}} \right)^2 \left( \frac{W}{S} \right)_{TO} \left( \left( \left( \frac{T}{W} \right)_{TO} - \mu \right)^{-1} + 1.414 \right)}{h_{TO} \rho g C_{L_{maxTO}} (1 + 1.414 \gamma_{LOF})} \right) \quad (A.15.1)$$

in eq. A.15.1, (T/W)<sub>to</sub> is the mean thrust-to-weight ratio taken at a speed of 0.707V<sub>LOF</sub>. And may be calculated as

$$T = 0.75 T_{TO} \frac{5 + \lambda_T}{4 + \lambda_T} \quad (A.15.2)$$

in eq. A.15.2, λ<sub>T</sub> is engine by-pass ratio, 3.5, and T<sub>TO</sub> is sea level static net thrust

, 5459.664 lbf.

Substituting these variables into eq. A.15.2, then the value of  $T$  is 4663.57 lbf.

The take-off weight  $W_{TO}$  assumes to be 10,000 lbs. From section A.9.2, eq. A.9.11, the value of aircraft maximum lift coefficient,  $C_{L\max}$ , is 1.61. Then from section A.9, eq. A.9.2, the aircraft sea-level stall speed,  $V_s$ , is 120.54 fps.

According to reference 7, p:122, the take-off speed is 1.2 multiply by aircraft sea-level stall speed, so,  $V_{LOf}$ , is 144.65 fps.

By using the drag subroutine, the total drag coefficient may be obtained ( $C_{D0}$  is 0.0244), and  $\mu_g$  is ground friction coefficient, from reference 7, p:40, for concrete or asphalt is 0.02.

$$\mu' = \mu_g + 0.72 \left( \frac{C_{D0}}{C_{L\max TO}} \right) \quad (A.15.3)$$

Substituting these variables into eq. A.15.3, the value of  $\mu'$  is 0.0323.

$$\gamma_{LOf} = 0.9 \left( \frac{T}{W} \right)_{TO} - \frac{0.3}{AR^{1/2}} \quad (A.15.4)$$

Substituting these values into eq. A.15.4, then the value of  $\gamma_{LOf}$  is 0.326.

Besides, from reference 7, table 5.1, the value of  $\left( \frac{V_3}{V_{STO}} \right)$  is 1.3. And from chap 2.4.1, the wing loading is 30. According to reference 7, table 5.1,  $f_{TO}$  is 1.15, obstacle height  $h_{TO}$  is 35 ft.

Substituting these variables into eq. A.15.1, after calculation, the take-off field length,  $S_{TO}$ , is 1307.37 ft.

## A.15.2 Take-off ground distance

Take-off ground distance can be computed by

$$S_{TOG} = \frac{V_{LOf}^2 / 2g}{\left( \frac{T}{W} \right)_{TO} - \mu'} \quad (A.15.5)$$

Substituting these variables into eq. A.15.5, the take-off ground distance is

$$S_{TOG} = 780.7 \text{ ft}$$

## A.16 Landing performance

Like take-off performance, landing distance can be divided into two parts, the distance from the obstacle height to the point of touchdown,  $S_{air}$ , and the landing ground run to zero speed on the runway,  $S_{LG}$ .



### A.16.1 The distance of $S_{air}$

Reference 7, eq. 5.82 may be used to calculate the distance of  $S_{air}$ .

$$S_{air} = \frac{1}{\bar{\gamma}} \left( \frac{V_{app}^2 - V_{TD}^2}{2g} + h_L \right) \quad (A.16.1)$$

Reference 7, fig. 5.20, the obstacle height,  $h_L$ , is 50 ft.

$$\bar{\gamma} = \left( \frac{D - T}{W} \right)_{ave} \quad (A.16.2)$$

From reference 7, p:164, an average value of  $\bar{\gamma}$  is 0.1. For landing, we assume the aircraft weight is 5,000 lbs, so the stall velocity from eq. A.9.2 now becomes 85.24 fps.

Assume the approach velocity is 1.3 multiplied by the stall velocity, then the approach velocity is 110.8 fps.

From reference 7, eq. 5.87, the touchdown speed may be calculated.

$$V_{TD} = V_{app} \left( 1 - \frac{\bar{\gamma}^2}{\Delta n} \right)^{1/2} \quad (A.16.3)$$

According to reference 7, assume  $\Delta n$  as 0.1 is a reasonable average. Substituting these variables into eq. A.16.3, the touchdown speed is 105.13 fps.

Substituting these variables into eq. A.16.1, then the distance of  $S_{air}$  is 690.8 ft.

### A.16.2 Landing ground run $S_{LG}$

Reference 7, eq. 5.82 may be used to calculate the distance of  $S_{LG}$ .

$$S_{LG} = \frac{V_{TD}^2}{2\bar{a}} \quad (A.16.4)$$

where  $\bar{a}$  is the average deceleration for the ground run. Reference 7, p:164, assume the value of,  $\bar{a}/g$ , for turbofan aircraft is 0.55. However, reference 66 suggests that in preliminary design, the value of 0.4 is more practical. Therefore, we take 0.4 as its value.

Substituting these variables into eq. A.16.4, the distance of  $S_{LG}$  is 429.36 ft.

Therefore, the total landing distance,  $S_{air}$  plus  $S_{LG}$ , is 1120.2 ft.

## A.17 Sensitivity analysis

This program is based on the assumption that the wing loading at take-off as 30 psf, aspect ratio as 15, and the thrust weight ratio as 0.4589. Based on these assumption, we did the aircraft conceptual design.

However, whether these assumption is the best solution or whether there is an other choice, still needs to be verified. In this section we will change both the aspect ratio and the wing loading to see the effect to the aircraft.

### A.17.1 Wing loading

First, we keep the same aspect ratio as 15, and take the wing loading as a variable, range from 28 to 38. Substituting this wing loading into our program, after calculation, we obtain the following iteration results, and put the wing loading VS aircraft take-off weight into table A.17.1 and the wing loading VS wing area and fuel tank volume into table A.17.2 respectively.

Table A.17.1, take-off weight is the final iteration results of different wing loading, fuel difference is negative means the fuel is more than mission required, and in accordance with the requirement that the absolute fuel difference value should within 10 pounds then the iteration being done.

From these results, it is easy to tell that when the wing loading increases, the aircraft take-off gross weight decreases. However, this situation keeps going until the wing loading is up to 36, when the aircraft take-off weight will increase again, in this case may be the wing loading as 36 is a good answer of minimum take-off weight.

| Wing loading | Take-off weight, lbs | Fuel diff. , lbs | $W_{emp} + W_{pay}$ , lbs | Wing/body incidence, deg. | Lift drag ratio |
|--------------|----------------------|------------------|---------------------------|---------------------------|-----------------|
| 28           | 10621.46             | -7.918           | 4802.952                  | 4.064                     | 24.394          |
| 30           | 10337.61             | -4.335           | 4543.467                  | 4.164                     | 23.959          |
| 32           | 10240.1              | -3.765           | 4410.448                  | 4.211                     | 23.649          |
| 34           | 10193.66             | 5.538            | 4338.907                  | 4.246                     | 23.549          |
| 36           | 10175.22             | 6.755            | 4274.451                  | 4.292                     | 23.328          |
| 38           | 10201.74             | 4.5              | 4216.026                  | 4.35                      | 23.107          |

Table A.17.1 Wing loading VS Aircraft take-off weight

| Wing loading | Wing area, ft <sup>2</sup> | Wingspan, ft | Mission fuel, lbs | Wing fuel tank, lbs | Fuselage fuel tank, lbs |
|--------------|----------------------------|--------------|-------------------|---------------------|-------------------------|
| 28           | 385.714                    | 76.064       | 5818.511          | 6240.222            | -421.711                |
| 30           | 340                        | 71.414       | 5794.137          | 5164.4              | 629.738                 |
| 32           | 312.5                      | 68.465       | 5829.651          | 4550.682            | 1278.969                |
| 34           | 294.118                    | 66.421       | 5854.755          | 4155.115            | 1699.639                |
| 36           | 277.778                    | 64.55        | 5900.764          | 3813.71             | 2087.054                |
| 38           | 263.158                    | 62.828       | 5985.71           | 3516.625            | 2469.085                |

Table A.17.2 Wing loading VS Wing area and fuel tank

From the results listed in table A.17.2, we may find that the higher the wing loading is, the less the wing area will be. And with the wing area decreases, the wing fuel tank will not be enough for carrying the whole mission fuel. The final column, if the value of fuselage fuel tank is negative means the wing is big enough to accommodate the mission fuel, and there is no need to put fuel into fuselage. Therefore, as the wing loading increases with the wing area decreases, we have to put more and more fuel into fuselage fuel tank.

On this basis, the table A.17.1 results, the wing loading 36 is the minimum take-off gross weight which is not match the need of the minimum fuel in fuselage. Therefore, an optimum solution should take different requirements into consideration.

### A.17.2 Aspect ratio

Again, if we fixed the wing loading, say 30, and let the aspect ratio change, range from 14 to 16, what is the impact to the aircraft take-off gross weight?

| T.O. weight, lbs | Fuel diff. , lbs | $W_{emp} + W_{pay}$ , lbs | Lift drag ratio |
|------------------|------------------|---------------------------|-----------------|
| 10249.82         | 66.09            | 4420.17                   | 23.14           |
| 10315.91         | 39.58            | 4420.17                   | 23.14           |
| 10355.49         | 23.28            | 4420.17                   | 23.14           |
| 10378.77         | 15.21            | 4420.17                   | 23.14           |
| 10393.98         | 7.47             | 4420.17                   | 23.14           |

(a) Aspect ratio as 14

| T.O. weight, lbs | Fuel diff. , lbs | W <sub>emp</sub> + W <sub>pay</sub> ,<br>lbs | Lift drag ratio |
|------------------|------------------|----------------------------------------------|-----------------|
| 10489.71         | -75.80           | 4543.47                                      | 23.96           |
| 10413.91         | -44.71           | 4543.47                                      | 23.96           |
| 10369.2          | -21.05           | 4543.47                                      | 23.96           |
| 10348.15         | -10.54           | 4543.47                                      | 23.96           |
| 10337.61         | -4.34            | 4543.47                                      | 23.96           |

(b) Aspect ratio as 15

| T.O. weight, lbs | Fuel diff. , lbs | W <sub>emp</sub> + W <sub>pay</sub> ,<br>lbs | Lift drag ratio |
|------------------|------------------|----------------------------------------------|-----------------|
| 10903.39         | -246.58          | 4723.96                                      | 24.82           |
| 10656.81         | -138.33          | 4723.96                                      | 24.82           |
| 10518.48         | -75.84           | 4723.96                                      | 24.82           |
| 10442.63         | -43.19           | 4723.96                                      | 24.82           |
| 10399.44         | -23.79           | 4723.96                                      | 24.82           |
| 10375.65         | -14.48           | 4723.96                                      | 24.82           |
| 10361.17         | -6.61            | 4723.96                                      | 24.82           |

(c) Aspect ratio as 16.

Table A.17.3 Aspect ratio VS Aircraft take-off gross weight

From table A.17.3 (a), (b), (c), compared with the previous paragraph results, the aspect ratio changes, if on the basis of the aspect ratio as 15, then no matter the aspect ratio is 14 or 16, the take-off gross weight change is not more than 0.5 percent. Therefore, the aspect ratio changes did not have much impact to aircraft take-off gross weight.

### A.17.3 Discussion

From above analysis, we would like to discuss why the wing loading changes have much impact on the aircraft weight rather than the aspect ratio changes to the aircraft take-off weight.

In order to simplify the demonstration, we only present at different wing loading, at cruise segment, the components drag coefficient and the total drag coefficient (drag coefficient less than 0.0005, didn't show in this table) in table A.17.4. Though, during cruise, there are 24 hours, for simplicity, instead of taking 24 hours, in each wing loading, we take the first 3 hours of cruise-climb, and first 2 hours of constant altitude loiter as an example to explore the difference.

|              | $C_{D0w}$ | $C_{Diw}$ | $C_{D0f}$ | $C_{D0e}$ | $C_{D0n}$ | $C_{Dn}$ | $C_D$  |
|--------------|-----------|-----------|-----------|-----------|-----------|----------|--------|
| Cruise-climb | 0.0077    | 0.0128    | 0.0024    | 0.0024    | 0.0012    | 0.0006   | 0.0271 |
| “            | 0.0077    | 0.0128    | 0.0024    | 0.0024    | 0.0012    | 0.0006   | 0.0272 |
| “            | 0.0076    | 0.0128    | 0.0024    | 0.0024    | 0.0012    | 0.0006   | 0.0271 |
| Loiter       | 0.0084    | 0.0128    | 0.0026    | 0.0027    | 0.0014    | 0.0006   | 0.0287 |
| “            | 0.0084    | 0.0120    | 0.0026    | 0.0027    | 0.0014    | 0.0006   | 0.0280 |

(a) Wing loading as 28

|              | $C_{D0w}$ | $C_{Diw}$ | $C_{D0f}$ | $C_{D0e}$ | $C_{D0n}$ | $C_{Dn}$ | $C_D$  |
|--------------|-----------|-----------|-----------|-----------|-----------|----------|--------|
| Cruise-climb | 0.0077    | 0.0129    | 0.0026    | 0.0024    | 0.0014    | 0.0007   | 0.0277 |
| “            | 0.0076    | 0.0129    | 0.0025    | 0.0024    | 0.0014    | 0.0007   | 0.0277 |
| “            | 0.0076    | 0.0129    | 0.0025    | 0.0024    | 0.0014    | 0.0007   | 0.0277 |
| Loiter       | 0.0084    | 0.0129    | 0.0027    | 0.0028    | 0.0015    | 0.0007   | 0.0293 |
| “            | 0.0084    | 0.0122    | 0.0027    | 0.0028    | 0.0015    | 0.0007   | 0.0285 |

(b) Wing loading as 30

|              | $C_{D0w}$ | $C_{Diw}$ | $C_{D0f}$ | $C_{D0e}$ | $C_{D0n}$ | $C_{Dn}$ | $C_D$  |
|--------------|-----------|-----------|-----------|-----------|-----------|----------|--------|
| Cruise-climb | 0.0077    | 0.0132    | 0.0027    | 0.0024    | 0.0015    | 0.0008   | 0.0283 |
| “            | 0.0076    | 0.0132    | 0.0026    | 0.0024    | 0.0015    | 0.0008   | 0.0282 |
| “            | 0.0077    | 0.0132    | 0.0026    | 0.0025    | 0.0015    | 0.0008   | 0.0282 |
| Loiter       | 0.0084    | 0.0132    | 0.0028    | 0.0027    | 0.0016    | 0.0008   | 0.0298 |
| “            | 0.0084    | 0.0124    | 0.0028    | 0.0027    | 0.0016    | 0.0008   | 0.0290 |

(c) Wing loading as 32

|              | $C_{D0w}$ | $C_{Diw}$ | $C_{D0f}$ | $C_{D0e}$ | $C_{D0n}$ | $C_{Dn}$ | $C_D$  |
|--------------|-----------|-----------|-----------|-----------|-----------|----------|--------|
| Cruise-climb | 0.0077    | 0.0134    | 0.0027    | 0.0024    | 0.0016    | 0.0008   | 0.0287 |
| “            | 0.0076    | 0.0134    | 0.0027    | 0.0024    | 0.0016    | 0.0008   | 0.0286 |
| “            | 0.0077    | 0.0134    | 0.0027    | 0.0025    | 0.0016    | 0.0008   | 0.0287 |
| Loiter       | 0.0084    | 0.0134    | 0.0029    | 0.0028    | 0.0017    | 0.0008   | 0.0303 |
| “            | 0.0084    | 0.0125    | 0.0029    | 0.0028    | 0.0017    | 0.0008   | 0.0294 |

(d) Wing loading as 34

|              | $C_{D0w}$ | $C_{Diw}$ | $C_{D0f}$ | $C_{D0e}$ | $C_{D0n}$ | $C_{Dn}$ | $C_D$  |
|--------------|-----------|-----------|-----------|-----------|-----------|----------|--------|
| Cruise-climb | 0.0077    | 0.0136    | 0.0028    | 0.0024    | 0.0017    | 0.0009   | 0.0291 |
| “            | 0.0076    | 0.0136    | 0.0028    | 0.0024    | 0.0017    | 0.0009   | 0.0290 |
| “            | 0.0077    | 0.0136    | 0.0028    | 0.0025    | 0.0017    | 0.0009   | 0.0291 |
| Loiter       | 0.0085    | 0.0136    | 0.0030    | 0.0028    | 0.0018    | 0.0009   | 0.0307 |
| “            | 0.0085    | 0.0127    | 0.0030    | 0.0028    | 0.0018    | 0.0009   | 0.0299 |

(e) Wing loading as 36

|              | $C_{D0w}$ | $C_{Diw}$ | $C_{D0f}$ | $C_{D0e}$ | $C_{D0n}$ | $C_{Dn}$ | $C_D$  |
|--------------|-----------|-----------|-----------|-----------|-----------|----------|--------|
| Cruise-climb | 0.0077    | 0.0138    | 0.0029    | 0.0024    | 0.0017    | 0.0009   | 0.0295 |
| “            | 0.0076    | 0.0138    | 0.0028    | 0.0025    | 0.0018    | 0.0009   | 0.0294 |
| “            | 0.0077    | 0.0138    | 0.0029    | 0.0025    | 0.0018    | 0.0009   | 0.0295 |
| Loiter       | 0.0084    | 0.0138    | 0.0031    | 0.0028    | 0.0019    | 0.0009   | 0.0310 |
| “            | 0.0084    | 0.0129    | 0.0031    | 0.0028    | 0.0019    | 0.0009   | 0.0302 |

(f) Wing loading as 38

Table A.17.4 Wing loading VS Drag coefficient

In table A.17.4, there is a trend of the total drag coefficient is slightly increased as the wing load increases. However, the last segment of cruise is including cruise-climb and constant altitude cruise these two parts. In order to demonstrate more clear, we divide table A.17.4 into two parts, and take the average value into table A.17.5.

| Wing loading | $C_{D0w}$ | $C_{Diw}$ | $C_{D0f}$ | $C_{D0e}$ | $C_{D0n}$ | $C_{Dn}$ | $C_D$  |
|--------------|-----------|-----------|-----------|-----------|-----------|----------|--------|
| 28           | 0.0077    | 0.0128    | 0.0024    | 0.0024    | 0.0012    | 0.0006   | 0.0272 |
| 30           | 0.0076    | 0.0129    | 0.0026    | 0.0024    | 0.0014    | 0.0007   | 0.0277 |
| 32           | 0.0077    | 0.0132    | 0.0026    | 0.0024    | 0.0015    | 0.0008   | 0.0282 |
| 34           | 0.0077    | 0.0134    | 0.0027    | 0.0024    | 0.0016    | 0.0008   | 0.0287 |
| 36           | 0.0077    | 0.0136    | 0.0028    | 0.0024    | 0.0017    | 0.0009   | 0.0291 |
| 38           | 0.0077    | 0.0138    | 0.0029    | 0.0025    | 0.0018    | 0.0009   | 0.0295 |

(a) Cruise-climb segment

| Wing loading | $C_{D0w}$ | $C_{Diw}$ | $C_{D0f}$ | $C_{D0e}$ | $C_{D0n}$ | $C_{Dn}$ | $C_D$  |
|--------------|-----------|-----------|-----------|-----------|-----------|----------|--------|
| 28           | 0.0084    | 0.0124    | 0.0026    | 0.0027    | 0.0014    | 0.0006   | 0.0283 |
| 30           | 0.0084    | 0.0126    | 0.0027    | 0.0028    | 0.0015    | 0.0007   | 0.0289 |
| 32           | 0.0084    | 0.0128    | 0.0028    | 0.0027    | 0.0016    | 0.0008   | 0.0294 |
| 34           | 0.0084    | 0.0130    | 0.0029    | 0.0028    | 0.0017    | 0.0008   | 0.0298 |
| 36           | 0.0085    | 0.0132    | 0.0030    | 0.0028    | 0.0018    | 0.0009   | 0.0303 |
| 38           | 0.0084    | 0.0133    | 0.0031    | 0.0028    | 0.0019    | 0.0009   | 0.0306 |

(b) Constant velocity segment

Table A.17.5 Average drag coefficient at different wing loading

In table A.17.5 (a) and (b), the wing zero and lift drag coefficient are increase as the wing loading increase. But the proportion of the wing zero drag coefficient to total drag coefficient is reduced as the wing loading increase, and the proportion of the lift drag coefficient is constant or slightly different as the wing loading increase.

| Wing loading | Velocity, fps | Wing, ft <sup>2</sup> | Density, lb-s <sup>2</sup> /ft <sup>4</sup> | Cd     | Drag, lbf |
|--------------|---------------|-----------------------|---------------------------------------------|--------|-----------|
| 28           | 497.77        | 385.7                 | 0.00018                                     | 0.0283 | 239.352   |
| 30           | 521.46        | 340                   | 0.00018                                     | 0.0289 | 236.462   |
| 32           | 538.97        | 312.5                 | 0.00018                                     | 0.0294 | 236.195   |
| 34           | 551.8         | 294                   | 0.00018                                     | 0.0298 | 236.086   |
| 36           | 565.75        | 277.8                 | 0.00018                                     | 0.0303 | 238.433   |
| 38           | 580.17        | 263.2                 | 0.00018                                     | 0.0306 | 239.917   |

Table A.17.6 Wing loading VS Drag force

In table A.17.6, when the wing loading increases, the drag force first decreases but as wing loading goes up in between 34 to 36, then the drag force will increase again.

## A.18 Stability analysis

In accordance with reference 45, the following steps were taken to estimate the size of horizontal stabiliser/control surface of the conceptual design, and to assure the aircraft stability.

### A.18.1 Trim in cruise

Referring to fig. 2.1.1, the cruise segment is within point *d* and *e*. The HALE on station cruise is 24 hours. At the end of cruise, the aircraft weight will have halved. In order to obtain reasonable results for the angle of incidence of the horizontal surface,  $\alpha_T$ , during cruise for trim. The computations will consider both extreme conditions during cruise, the initial and final cruise point.

From reference 45, eq 2, trim in cruise can be computed by;

$$\frac{S_T}{S} = \frac{G \left( \frac{TZ_T}{MAC} + qSC_{M0} \right) + Mg(H - H_0)}{qS \left( \frac{MgC_{LaT}}{qSC_{LaW}} \left( -\frac{d\varepsilon}{d\alpha} \right) + (C_{LaT}(\alpha_T + \alpha_w) + C_{i\eta}\eta) \right) \frac{l_T}{MAC}} \quad (\text{A.18.1})$$

If we want to calculate the value of angle of incidence of horizontal surface,  $\alpha_T$ , the eq. A.18.1 may be rewritten as eq. A.18.2.

Before calculation, the value of *G*, and elevator deflection,  $\eta$ , are assumed as 1 and 0 degrees respectively. Then the value of angle of incidence of horizontal surface,  $\alpha_T$ , may be calculated.

$$\alpha_T = \frac{\frac{TZ_T}{MAC} + qSC_{m0WB} + Mg(H - H_0)}{C_{LaT} \frac{S_T}{S} qS \frac{l_T}{MAC}} - \frac{Mg}{qSC_{LaW}} \left( 1 - \frac{d\varepsilon}{d\alpha} \right) - \alpha_w \quad (\text{A.18.2})$$

In order to obtain  $\alpha_T$ , there are 13 variables in eq. A.18.2 needs to be solved. Following is the calculation.

#### a. Distance of thrust line below centre of gravity, $Z_T$ .

The thrust line of the HALE is above centre of gravity. According to reference 45, it is negative (-3 ft).

#### b. Mean aerodynamic chord, MAC

From section A.4.1.1, eq. A.4.6, the mean aerodynamic chord is 4.94 ft.

#### c. Wing area, *S*

From section A.4.1.1, eq. A.4.3, the value of wing area is 340 ft<sup>2</sup>.



**d. Wing body pitch moment coefficient at zero lift,  $C_{m0WB}$**

From reference 46, the wing body pitch moment coefficient at zero lift  $C_{m0WB}$  can be calculated from following steps.

**d.1 Estimation of wing alone zero lift angle of attack,  $\alpha_{0rW}$**

From reference 47, eq. 3.2, wing alone zero lift angle of attack  $\alpha_{0rW}$  is comprised of two parts,

$$\alpha_{0rW} = \alpha_{0r1} + \alpha_{0r2} \quad (A.18.3)$$

$\alpha_{0r1}$  is the contribution due to reference camber line, and  $\alpha_{0r2}$  is the contribution due to effect of wing twist.

**d.1.1 Calculate the value of  $\alpha_{0r1}$**

From reference 44, the definition of camber distribution in chap. 2.4.

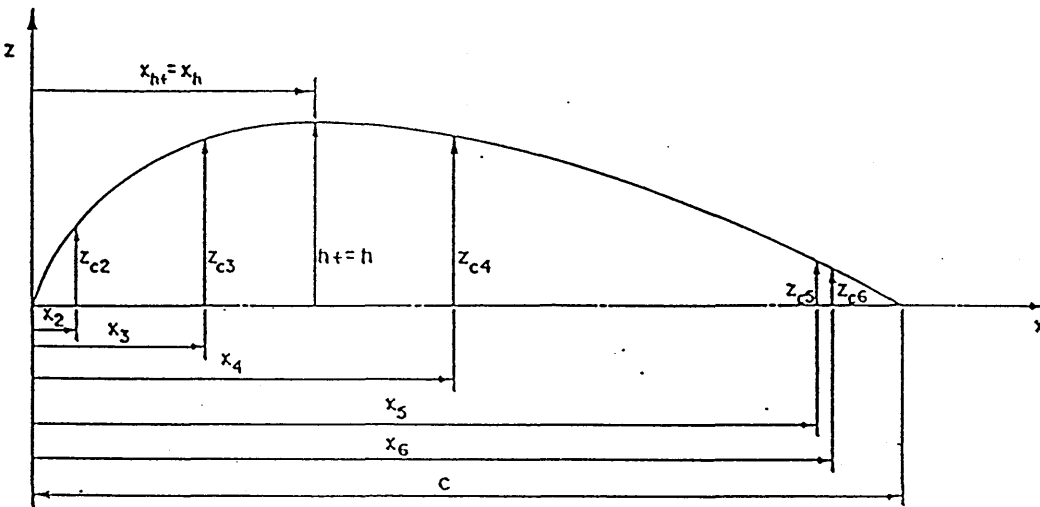


Fig A.18.1 The camber distribution

In fig A.18.1, the subscript  $i$  of  $x_i$  relates to:

|         |          |         |         |
|---------|----------|---------|---------|
| $i = 1$ | $0.005c$ | $i = 4$ | $0.5c$  |
| $i = 2$ | $0.05c$  | $i = 5$ | $0.9c$  |
| $i = 3$ | $0.2c$   | $i = 6$ | $0.92c$ |

From section A.7.1, table A.7.1, the camber line distribution of airfoil MS(1)-0317 is:

$$\frac{Z_{c3}}{Z_{c4}} = \frac{0.01383}{0.01472} = 0.9395$$

$$\frac{Z_{C6}}{Z_{C4}} = \frac{0.00792}{0.01472} = 0.538$$

$$\frac{Z_{C4}}{C} = \frac{0.01472}{1} = 0.01472$$

From reference 44, fig. 2, if  $\frac{Z_{C6}}{Z_{C4}}$  is 0.538, then  $F_2$  equals to -1.748, and the zero lift incidence is:

$$\alpha_{0th} = \left( F_2 - 0.23 \frac{Z_{C3}}{Z_{C4}} \right) 100 \frac{Z_{C4}}{C} \quad (\text{A.18.4})$$

After calculation, the value of zero lift incidence is -2.89 degrees.

Also, from section A.7.1, table A.7.1, the camber line for airfoil MS(1)-0313;

$$\frac{Z_{C3}}{Z_{C4}} = \frac{0.01374}{0.01512} = 0.9087$$

$$\frac{Z_{C6}}{Z_{C4}} = \frac{0.00617}{0.01512} = 0.408$$

$$\frac{Z_{C4}}{C} = \frac{0.01512}{1} = 0.01512$$

From reference 44, fig. 2, if  $\frac{Z_{C6}}{Z_{C4}}$  is 0.408, then  $F_2$  equals to -1.418, and the zero lift

incidence may be computed by eq. A.17.4. After calculation, the value of zero lift incidence is -2.46.

Since the airfoil at the wing root is MS(1)-0317, and at the wing tip is MS(1)-0313. In order to simplify the computation, the average value was taken between the two airfoil zero lift incidence. This average value is called the wing airfoil zero lift incidence:

$$\alpha_{0th} = \frac{-2.89 - 2.46}{2} = -2.675$$

The above value is obtained at the condition of incompressible flow (assume  $M \sim 0$ ). From reference 47, the zero lift angle of attack of the airfoil at low speed may be calculated by eq. 3.8.

$$\alpha_{0l} = 0.87 \alpha_{0th} \quad (\text{A.18.5})$$

Thus, the value of wing airfoil zero lift incidence is -2.328 degrees.

#### d.1.2 Calculate the value of $\alpha_{0r2}$

From reference 47, eq. 3.10

$$\frac{\delta}{\delta_i} = \frac{\eta\lambda}{1 - \eta(1 - \lambda)} \quad (\text{A.18.6})$$

$\delta$  is local camber dependent twist relative to wing root, at the wing tip  $\delta_i$  is -3 degrees.  $\eta$  is spanwise distance from root as fraction of semi-span, for HALE,  $\eta$  is 1.

$\lambda$  is the wing taper ratio, from section A.4.1.1, the value of  $\lambda$  is 0.5.

Substituting these variables into eq. A.18.6, then the value of  $\delta$  is -3 degrees.

In order to solve the value of  $\alpha_{or2}$ , the following equations were generated:

First, from section A.4.1.1, the wing aspect ratio is 15, and quarter wing chord sweep angle ( $\tan\Lambda_{1/4}$ ) is 0 degrees, so the value of  $AR \tan \Lambda_{1/4}$  is 0. Then from reference 47, fig. 1,

$$\begin{aligned} \alpha &= 0.0008\lambda - 0.0006 \\ \beta &= -0.0052\lambda + 0.0068 \\ \gamma &= -0.0856\lambda - 0.3727 \end{aligned}$$

Then

$$\frac{\alpha_{or2}}{\delta} = \alpha \left( A \tan \Lambda_{1/4} \right)^2 + \beta \left( A \tan \Lambda_{1/4} \right) + \gamma \quad (\text{A.18.7})$$

After calculation, the value of  $\alpha_{or2}$  is 1.2415.

We now have the values of  $\alpha_{or1}$  and  $\alpha_{or2}$ . Substituting these values into eq. A.18.3, then the value of  $\alpha_{orW}$  is -1.0815.

#### d.2 Estimate of body effect on zero lift angle of attack, $\Delta\alpha_{or}$

From reference 47, eq. 3.8,

$$\frac{K_2}{K_1} = \frac{1 + 0.7w/b}{1.03 + 2.15w/b} \quad (\text{A.18.8})$$

from section A.4.1, the width of fuselage is 3 ft and the wing span is 71.41 ft. So,

$$\frac{K_2}{K_1} = 0.9188$$

and from reference 47, eq.3.8,

$$\Delta\alpha_{or} = \left( 1 - \frac{K_2}{K_1} \right) (i_w - \alpha_{or1}) \quad (\text{A.18.9})$$

From appendix C.3, the wing body incidence,  $i_w$  is 4.266 degrees, and from eq. A.18.5, the value of wing airfoil zero lift incidence,  $\alpha_{or1}$  is -2.328 degree. Substituting these

variables into eq. A.18.9, then the value of body effect on zero lift angle of attack,  $\Delta\alpha_{0r}$ , is 0.535.

#### d.3 Estimation of wing body zero lift angle of attack, $\alpha_{0rWB}$

From reference 47, The value of wing body zero lift angle of attack is combining the results of d.1 and d.2.

$$\begin{aligned}\alpha_{0rWB} &= \alpha_{0rW} + \Delta\alpha_{0r} \\ &= -0.546 \text{ deg.}\end{aligned}\tag{A.18.10}$$

#### d.4 Estimation of wing alone pitching moment coefficient at zero lift, $C_{m0W}$

From reference 48, eq. 3.1, the zero lift pitching moment coefficient can be written as

$$C_{m0W} = C_{m01} + C_{m02}\tag{A.18.11}$$

and  $C_{m01}$ ,  $C_{m02}$  is the effect of platform and twist respectively.

##### d.4.1 Evaluation of $C_{m01}$ , platform effect

From section A.7.1, table A.7.1, the camber line distribution for airfoil MS(1)-0317

$$\begin{aligned}\frac{Z_{c2}}{h+} &= \frac{0.01014}{0.015295} = 0.663 \\ \frac{Z_{c5}}{h+} &= \frac{0.00992}{0.015295} = 0.6486 \\ \frac{h+}{C} &= 0.015295\end{aligned}$$

and the camber line distribution for airfoil MS(1)-0313 as:

$$\begin{aligned}\frac{Z_{c2}}{h+} &= \frac{0.00999}{0.015794} = 0.6325 \\ \frac{Z_{c5}}{h+} &= \frac{0.008272}{0.015794} = 0.5237 \\ \frac{h+}{C} &= 0.015794\end{aligned}$$

In order to simply the computation process, the author take average value of MS(1)-0317 and -0313.

$$\begin{aligned}\frac{Z_{c2}}{h+} &= \frac{0.663 + 0.6325}{2} = 0.6478 \\ \frac{Z_{c5}}{h+} &= \frac{0.6486 + 0.5237}{2} = 0.586 \\ \frac{h+}{C} &= \frac{0.015794 + 0.015295}{2} = 0.0155\end{aligned}$$

Then substitute those values into reference 44, fig. 3, and calculated by following equations:

$$\alpha = -262.9 \left( \frac{Z_{c2}}{h+} \right)^5 + 674.6 \left( \frac{Z_{c2}}{h+} \right)^4 - 604.1 \left( \frac{Z_{c2}}{h+} \right)^3 + 217.7 \left( \frac{Z_{c2}}{h+} \right)^2 - 25.1 \left( \frac{Z_{c2}}{h+} \right) + 5.9$$

$$\beta = 194 \left( \frac{Z_{c2}}{h+} \right)^5 - 496.5 \left( \frac{Z_{c2}}{h+} \right)^4 + 444.2 \left( \frac{Z_{c2}}{h+} \right)^3 + 160.5 \left( \frac{Z_{c2}}{h+} \right)^2 + 18.2 \left( \frac{Z_{c2}}{h+} \right) - 12.1$$

$$\gamma = -35.2 \left( \frac{Z_{c2}}{h+} \right)^5 + 89.6 \left( \frac{Z_{c2}}{h+} \right)^4 - 80 \left( \frac{Z_{c2}}{h+} \right)^3 + 29 \left( \frac{Z_{c2}}{h+} \right)^2 - 2.2 \left( \frac{Z_{c2}}{h+} \right) + 0.0814$$

Then the value of  $\frac{C_{m0i}}{h+/c}$  is

$$\begin{aligned} \frac{C_{m0i}}{h+/c} &= \alpha \left( \frac{Z_{c5}}{h+} \right)^2 + \beta \frac{Z_{c5}}{h+} + \gamma \\ &= -4.4 \end{aligned} \quad (A.18.12)$$

Therefore, the value of incompressible pitching moment coefficient at zero total wing lift,  $C_{m0i}$  is -0.0685.

From reference 48, fig. 1, the empirical correction factor, F is

$$\begin{aligned} F &= 2.143 C_{m0i}^2 + 1.68 C_{m0i} + 0.96 \\ &= 0.8585. \end{aligned} \quad (A.18.13)$$

Since the wing is straight taper and camber line is unchanged across the span, then from reference 48, eq. 3.11,  $C_{m0il}$  is

$$C_{m0il} = \frac{2AR}{2AR+1} \cos \Lambda_{\frac{1}{2}} F C_{m0i} \quad (A.18.14)$$

Since  $\cos \Lambda_{\frac{1}{2}}$  is 0 degrees, then substitute these variables into eq. A.18.14, the value of  $C_{m0il}$  is -0.05689. From reference 48, eq. 3.31, the effect of Mach number is;

$$C_{m01} = \frac{2AR+1}{2\beta AR+1} C_{m0il} = -0.0662 \quad (A.18.15)$$

#### d.4.2 Evaluation of $C_{m0i2}$ , linear twist

For straight and swept wing, the value of K is

if (  $\lambda$  .GT. 0.5) then

$$K = 0.017$$

else

$$K = 0.019$$

end if

From reference 48, eq.3.22,

$$\frac{C_{m0i2}}{\delta} = -\frac{K}{8} \frac{(1+\lambda)(1+2\lambda)}{(1+\lambda+\lambda^2)} \frac{A^2}{A+10} \tan \Lambda_{\frac{1}{2}} \quad (\text{A.18.16})$$

$$= 0.$$

From eq. 3.22, Mach number effect :

$$C_{m02} = \frac{A+10}{\beta A+10} C_{m0i2} \quad (\text{A.18.17})$$

$$= 0.$$

Substituting the values of  $C_{m01}$ ,  $C_{m02}$  into eq. A.18.11, the zero lift pitching moment coefficient is -0.0662.

#### d.5 Estimation of body effect pitching moment coefficient at zero lift, $\Delta C_{m0}$

From reference 46, eq. 4.1, the value of body effect pitching moment coefficient at zero lift,

$$\Delta C_{m0} = C_{m0b} + \Delta_z C_{m0} + \Delta_s C_{m0} \quad (\text{A.18.18})$$

The value of  $C_{m0b}$  can be calculated by following equations (reference 46, fig. 2.)

Assume  $X = \frac{w^2}{S_B}$ ,  $w$  is maximum body width, 3 ft, and  $S_B$  is platform area of body,

94.478 ft<sup>2</sup>. Then

$$\alpha = 61456X^5 - 51912.7X^4 + 16319.2X^3 - 2359.3X^2 + 157.8X - 3.46$$

$$\beta = -58722.7X^5 + 50342X^4 - 16094.7X^3 + 2368.7X^2 - 164.9X + 3.4$$

$$\gamma = 8832X^5 - 7430.7X^4 + 2314.3X^3 - 317.3X^2 + 9.4X - 0.47$$

Assuming  $Y = \frac{S_{Bn} I_{Bn}}{S_B L}$ ,  $S_{Bn}$  is platform area of body forward of lateral line through quarter chord point of MAC, 40.8 ft<sup>2</sup>,  $I_{Bn}$  is the distance from body nose to quarter chord point of MAC, 15.52 ft and  $L$  is the length of body, 35.7 ft.

$$\frac{10^3 C_{m0b} S_w c}{\psi S_B L} = \alpha Y^2 + \beta Y + \gamma \quad (\text{A.18.19})$$

$$= -1.18 \text{ deg}^{-1}$$

Because  $S_w$  is planform of gross area, 320.95 ft<sup>2</sup>,  $\psi$  is the angle between zero lift plane of wing and airflow direction for zero body alone pitching moment, 5.35 degrees  
Substituting these variables into eq. A.18.19, then the value of  $C_{m0b}$  is -0.0134.

From reference 46, eq. 4.2,  $Z$  is vertical displacement of quarter chord point of  $C_r$  above mid body position, -0.5 ft, and  $h$  is maximum height of body, 3 ft. So the contribution due to wing height is

$$\begin{aligned}\Delta_z C_{m0} &= 0.01 \frac{Z}{h} \\ &= -0.0017\end{aligned}\tag{A.18.20}$$

From reference 46, eq. 4.3, the contribution due to wing sweep is

$$\begin{aligned}\Delta_s C_{m0} &= -0.053 \left( C_{m0b} \delta_t \frac{w}{b} AR \tan \Lambda_{1/4} \right)^{0.3} \\ &= 0.\end{aligned}\tag{A.18.21}$$

Substituting these three variables into eq. A. 18.18, the value of body effect pitching moment coefficient at zero lift,  $\Delta C_{m0}$  is -0.015.

Finally, the value of wing body pitching moment coefficient at zero lift,  $C_{m0WB}$  is the results of steps d.4 and d.5,

$$\begin{aligned}C_{m0WB} &= C_{m0W} + \Delta C_{m0} \\ &= -0.0813\end{aligned}\tag{A.18.22}$$

**e. Location of horizontal surface lift after of centre of gravity,  $l_T$**

From reference 45,  $l_T$  is the distance from the location of aircraft most after C.G. to the location of tailplane aerodynamic centre. For HALE is 13.77 ft.

**f. Wing body incidence,  $\alpha_w$**

From section A.7, the wing body incidence  $\alpha_w$  is -4.266 degrees.

**g. Location of the aircraft centre of gravity**

From wing leading edge and divided by the MAC,  $H$  is 0.354.

From above computation, 7 variables of eq. A.18.2 has been solved, but there are 6 variables still remaining unknown, and these variables are varied with different cruise conditions.

**A. At the beginning of the cruise,  $C$**

At the beginning of cruise, the aircraft altitude is 52,000 ft, weight is 9,104 lbs and velocity is 502.9 fps. The rest of variables in eq. A.18.2 are

a. Thrust: Thrust required at this moment is 392.15 lbf.

b. Dynamic pressure:

$$\begin{aligned}q &= \frac{1}{2} \rho V^2 \\ &= 40.34 \text{ lb/ft}^{-2}\end{aligned}\tag{A.18.23}$$

- c. Wing body lift curve slope,  $C_{L_{awf}}$  is 6.01.
- d. Tailplane lift curve slope,  $C_{L_{at}}$  is 4.61.
- e. Down wash gradient at the horizontal tail,  $\frac{d\varepsilon}{d\alpha}$  is 0.315.
- f. Wing body aerodynamic centre,  $H_0$ .

From reference 9, eq. E-36, E-37, Wing body aerodynamic centre can be computed by

$$H_0 = 0.25 - \frac{1.8}{C_{L_{awf}}} \frac{b_f h_f l_{fn}}{Sc} \quad (\text{A.18.24})$$

$$= 0.2272$$

According to reference 45, we take the aircraft C.G. within the range  $\pm 0.04$ , and assumes  $s_r/s$  is 0.15, then  $s_r/s$  within the range  $\pm 0.05$ , to calculate the trim in cruise,  $\alpha_T$ .

Substituting these variables into eq. A.18.2, the following results are obtained.

| $H$   | $s_r/s$ | $\alpha_T$ |
|-------|---------|------------|
| 0.314 | 0.15    | -1.291     |
| 0.314 | 0.2     | -0.985     |
| 0.314 | 0.25    | -0.801     |
| 0.314 | 0.3     | -0.679     |
| 0.354 | 0.15    | -0.502     |
| 0.354 | 0.2     | -0.393     |
| 0.354 | 0.25    | -0.328     |
| 0.354 | 0.3     | -0.284     |
| 0.394 | 0.15    | 0.287      |
| 0.394 | 0.2     | 0.199      |
| 0.394 | 0.25    | 0.146      |
| 0.394 | 0.3     | 0.111      |

Table A.18.1 Trim in beginning cruise,  $\alpha_T$

Since the range of aircraft C.G. movement is small. So we can take an average value of aircraft C.G., compare to Mean Aerodynamic Chord, is at 0.354, then the elevator angle,  $\alpha_T$  is -0.376 degrees.



**B. At the end of cruise,  $e$ .**

At the end of cruise, the aircraft altitude is 65,000 ft , weight is 4,550 lbs and velocity is 502.9 fps. The rest of variables in eq. A.18.2 are

- a. Thrust: Thrust required at this moment is 210.9 lbf.
- b. Dynamic pressure: from eq. A.18.23,  $q$  is 22.17 lb/ft<sup>2</sup>
- c. Wing body lift curve slope,  $C_{L_{\alpha wf}}$  is 5.86.
- d. Tailplane lift curve slope,  $C_{L_{\alpha T}}$  is 4.514.
- e. Down wash gradient at the horizontal tail,  $\frac{d\varepsilon}{d\alpha}$  is 0.3152.
- f. Wing body aerodynamic centre. From eq. A.18.24,  $H_0$  is 0.2266.

| $H$   | $s_T/s$ | $\alpha_T$ |
|-------|---------|------------|
| 0.314 | 0.15    | -1.161     |
| 0.314 | 0.2     | -0.814     |
| 0.314 | 0.25    | -0.606     |
| 0.314 | 0.3     | -0.468     |
| 0.354 | 0.15    | -0.428     |
| 0.354 | 0.2     | -0.265     |
| 0.354 | 0.25    | -0.167     |
| 0.354 | 0.3     | -0.101     |
| 0.394 | 0.15    | 0.305      |
| 0.394 | 0.2     | 0.285      |
| 0.394 | 0.25    | 0.273      |
| 0.394 | 0.3     | 0.265      |

Table A.18.2 Trim in final cruise,  $\alpha_T$ .

As with previous section, we take an average value of  $\alpha_T$  at aircraft C.G. is 0.354, then trim in cruise,  $\alpha_T$  at the end of cruise is -0.24 degrees.

**A.18.2 Trim on approach to land**

Eq. A.18.1 is rewritten to calculate trim on approach to landing, but this time the value of  $\alpha_T$  has been obtained from section A.18.1, but  $\eta$  is a variable.

$$C_{IT}\eta = \frac{\frac{TZ_T}{c} + qSC_{m_{OWB}} + Mg(H - H_0)}{C_{LaT} \frac{S_T}{S} qS \frac{l_T}{c}} - \frac{Mg}{qSC_{LaW}} \left(1 - \frac{d\varepsilon}{d\alpha}\right) - C_{LaT}(\alpha_T + \alpha_w) \quad (A.18.25)$$

As the same with section A.18.1, the value of  $G$  is assumed to be 1, and the approach thrust is 240 lbf. According to reference 66, the value of lift coefficient due to control surface deflection,  $C_{IT}$  can be calculated by:

$$\frac{C_{IT}}{C_{LaT}} = 1.22 \frac{C_f^{0.5}}{C} + \frac{0.1}{\beta AR_T} \quad (A.18.26)$$

First assume the proportion of control surface chord to wing chord,  $c/c$ , is 0.3. From section A.7, eq. A.7.1, the Prandtl-Glauert compressibility factor,  $\beta$  is 0.8545 and the tailplane aspect ratio is 5.19. Substitute these variables into eq. A.18.26, then the value of lift coefficient due to control surface deflection,  $\frac{C_{IT}}{C_{LaT}}$  is 0.691.

The value of elevator deflection,  $\eta$  is calculated as follows :

#### A.18.2.1 When $\alpha_T$ is -0.377 and tail lift curve slope, $C_{LaT}$ is 5.97.

From reference 66, the value of  $K_1, K_2$  is :

$$K_1 = \left( 0.15 + 2.2 \left( \frac{t}{c} \right)_T \frac{1}{\cos \Lambda} \left( 1 - 1.6 \frac{C_f}{C} \right) \right) \quad (A.18.27)$$

The value of tailplane thickness ratio,  $(t/c)_T$  is 0.12, and tailplane sweep angle,  $\Lambda$  is 0 degrees. Substituting these variables into eq. A.18.27, then the value of  $K_1$  is 0.2153. From reference 49, the following equations may be used to calculate  $\eta$ .

Since the Reynolds number at tailplane,  $R_N$  is  $1.97 \times 10^6$  less than  $6 \times 10^6$ . According to reference 66, the value of  $K_2$  is 1. Then, from reference 49, eq.1, the value of  $C_{IT}$  may be calculated.

$$\begin{aligned} C_{IT} &= \frac{C_{IT}}{C_{LaT}} C_{LaT} (1 - K_1 K_2) \\ &= 3.237 \end{aligned} \quad (A.18.28)$$

Substituting these variables into eq. A.18.25, and takes the aircraft C.G. within the range  $\pm 0.04$ , and takes  $s_r/s$  as 0.15, 0.2, 0.25 and 0.3, to calculate trim on approach to land, the value of elevator deflection,  $\eta$ . The results obtained are :

| $H$   | $s_z/s$ | $\eta$ |
|-------|---------|--------|
| 0.314 | 0.15    | -3.409 |
| 0.314 | 0.2     | -3.101 |
| 0.314 | 0.25    | -2.916 |
| 0.314 | 0.3     | -2.793 |
| 0.354 | 0.15    | -1.808 |
| 0.354 | 0.2     | -1.9   |
| 0.354 | 0.25    | -1.955 |
| 0.354 | 0.3     | -1.992 |
| 0.394 | 0.15    | -0.207 |
| 0.394 | 0.2     | -0.7   |
| 0.394 | 0.25    | -0.995 |
| 0.394 | 0.3     | -1.192 |

Table A.18.3 Trim on approach,  $\alpha_T$  is -0.377 degrees

#### A.18.2.2 When $\alpha_T$ is -0.24 and tail lift curve slope, $C_{l_{\alpha T}}$ is 5.773

With the same value of  $K_1, K_2$ , and  $\frac{C_{IT}}{C_{L\alpha T}}$ , as in section A.18.2.1. From eq. A.18.28, the value of lift coefficient due to control surface deflection,  $C_{IT}$  is 3.129.

| $H$   | $s_z/s$ | $\eta$ |
|-------|---------|--------|
| 0.314 | 0.15    | -3.88  |
| 0.314 | 0.2     | -3.567 |
| 0.314 | 0.25    | -3.38  |
| 0.314 | 0.3     | -3.255 |
| 0.354 | 0.15    | -2.224 |
| 0.354 | 0.2     | -2.325 |
| 0.354 | 0.25    | -2.386 |
| 0.354 | 0.3     | -2.427 |
| 0.394 | 0.15    | -0.568 |
| 0.394 | 0.2     | -1.084 |
| 0.394 | 0.25    | -1.393 |
| 0.394 | 0.3     | -1.599 |

Table A.18.4 Trim on approach,  $\alpha_T$  is -0.24 degrees

The results obtained in table A.18.4, is as we did in section A.18.2.1. Substituting these variables into eq. A.18.25, and assume +, - 0.04 of aircraft C.G. and  $s_{r/s}$  to be 0.15, 0.2, 0.25 and 0.3, then the value of elevator deflection,  $\eta$ , may be acquired.

### A.18.3 Cruise static stability

From reference 45, eq.3 can be used for calculating static stability,

$$\frac{S_T}{S} = \frac{G(K_n - H_0 + H)}{\left( \frac{C_{L\alpha T}}{C_{L\alpha Wf}} \left( 1 - \frac{d\varepsilon}{d\alpha} \right) \right) \frac{l_T}{c}} \quad (\text{A.18.29})$$

In order to solve the value of  $s_{r/s}$ , at a stick fixed static margin,  $K_n$  is 0.05 and 0.1. We assume the value of  $G$  is 1, and from section A.18.1 and A.18.2, the variables in eq. A.18.29 has been obtained. So, if we calculate the cruise static stability at both extrem conditions during cruise, then the following results may be obtained.

#### A.18.3.1 Initial cruise

After computation, the value of  $s_{r/s}$  is

| $H$    | $K_n$ | $s_{r/s}$ |
|--------|-------|-----------|
| 0.3137 | 0.05  | 0.0933    |
| 0.3537 | 0.05  | 0.1206    |
| 0.3937 | 0.05  | 0.148     |
| 0.3137 | 0.1   | 0.1275    |
| 0.3537 | 0.1   | 0.1548    |
| 0.3937 | 0.1   | 0.1821    |

Table A.18.5 Initial cruise static stability

#### A.18.3.2 Final cruise

After computation, the value of  $s_{r/s}$  is

| $H$    | $K_n$ | $s_{r/s}$ |
|--------|-------|-----------|
| 0.3137 | 0.05  | 0.0933    |
| 0.3537 | 0.05  | 0.1205    |
| 0.3937 | 0.05  | 0.1477    |
| 0.3137 | 0.1   | 0.1273    |
| 0.3537 | 0.1   | 0.1545    |
| 0.3937 | 0.1   | 0.1817    |

Table A.18.6 Final cruise static stability

#### A.18.4 Rotation at take off

Assume the elevator up deflection,  $\eta$  at take off is -25 degrees, and use eq. 1 in reference 45, the value of  $s_{r/s}$  at take off rotation can be calculated by :

$$\frac{S_T}{S} = \frac{\left( T \frac{h_T}{c} - qS(C_{m0WB} - C_{L\alpha wf} \alpha_w (H_G - H_0)) \right) + Mg \left( H_G - H + \frac{0.25K_B^2}{l_T^1 c} \right)}{qS \left( C_{L\alpha h} \left( \alpha_T - \frac{Mg}{qSC_{L\alpha wf}} \frac{d\varepsilon}{d\alpha} \right) + C_{i\eta} \eta \right) \left( H_G - \frac{l_T^1}{c} \right)} \quad (\text{A.18.30})$$

In eq. A.18.30, the take off thrust is 5460 lbf, the distance from thrust line above ground,  $h_T$  is 7.5 ft, the location of leading edge of MAC of the most after wheel position,  $H_G$  is 0.737, the wing body lift curve slope,  $C_{L\alpha wf}$  is 5.37, the take off weight is 9680.9 lbs, the down wash gradient at the horizontal tail,  $\frac{d\varepsilon}{d\alpha}$  is 0.277, the distance from wing leading

edge to tailplane aerodynamic centre,  $l_T^1$  is 15.5 ft, the wing body aerodynamic centre,  $H_0$ , from eq A.18.24, is 0.186, the pitch radius of gyration,  $K_B$  is 6.66, and the position of centre of gravity after of leading edge of MAC, from section A.18.1, H is 0.354.

The only parameter varied is the angle of incidence of horizontal surface,  $\alpha_T$ . From section A.18.1, we obtained two values of  $\alpha_T$ , and from eq. A.9.2, the aircraft stall velocity is 147.67 fps.

Before calculating, the tail volume coefficient, and the  $s_{r/s}$  of the HALE should solve first. According to reference 10, eq. 6.26, the tail volume coefficient may be computed.

$$\bar{V} = \frac{S_T}{S} \frac{l_T}{c} \quad (\text{A.18.31})$$

For HALE, the horizontal tailplane area is 64.64 ft<sup>2</sup> wing area is 340 ft<sup>2</sup>, wing mean aerodynamic chord is 4.94 ft, Substitute these variables into eq. A.18.31, then the tail volume coefficient,  $\bar{V}$ , of the HALE is 0.53. The value of  $s_r/s$  of HALE is 0.19.

In order to obtain a proper value of the tail volume coefficient, and  $s_r/s$ , compare to the HALE, we take different take-off speed and change the proportion of control surface chord to wing chord,  $c/c$ , into our calculation. Then put the results into following tables.

**A.  $\alpha_T$  is -0.376 degrees**

| take off velocity, fps         | $s_r/s$ | $\bar{V}$ | HALE $\bar{V}$ |
|--------------------------------|---------|-----------|----------------|
| 146.9 (1.2V <sub>stall</sub> ) | 0.278   | 0.777     | 0.53           |
| 159.2 (1.3V <sub>stall</sub> ) | 0.24    | 0.67      |                |
| 171.4 (1.4V <sub>stall</sub> ) | 0.21    | 0.579     |                |

(a)  $c/c$  is 0.3

| take off velocity, fps          | $s_r/s$ | $\bar{V}$ | HALE $\bar{V}$ |
|---------------------------------|---------|-----------|----------------|
| 146.9 (1.2V <sub>stall</sub> )  | 0.23    | 0.643     | 0.53           |
| 159.2 (1.3V <sub>stall</sub> )  | 0.198   | 0.55      |                |
| 165.3 (1.35V <sub>stall</sub> ) | 0.183   | 0.512     |                |

(b)  $c/c$  is 0.4

Table A.18.7, Tail volume coefficient for rotation

**B.  $\alpha_T$  is -0.24 degrees**

| take off velocity, fps         | $s_r/s$ | $\bar{V}$ | HALE $\bar{V}$ |
|--------------------------------|---------|-----------|----------------|
| 146.9 (1.2V <sub>stall</sub> ) | 0.288   | 0.8       | 0.53           |
| 159.2 (1.3V <sub>stall</sub> ) | 0.248   | 0.69      |                |
| 171.4 (1.4V <sub>stall</sub> ) | 0.215   | 0.6       |                |

(a)  $c/c$  is 0.3

| take off velocity, fps          | $s_r/s$ | $\bar{V}$ | HALE $\bar{V}$ |
|---------------------------------|---------|-----------|----------------|
| 146.9 (1.2V <sub>stall</sub> )  | 0.238   | 0.665     | 0.53           |
| 159.2 (1.3V <sub>stall</sub> )  | 0.2     | 0.57      |                |
| 165.3 (1.35V <sub>stall</sub> ) | 0.19    | 0.53      |                |

(b)  $c/c$  is 0.4

Table A.18.8 Tail volume coefficient for rotation

## A.18.5 Dynamic stability

According to reference 45, for the stick fixed condition with a rigid airframe the longitudinal equations of motion of the aircraft usually yield four roots. These forms in pairs to give, first, short period motion, and second, long period motion or “phugoid” mode. The following section detail these two motions.

### A.18.5.1 Short period oscillation

According to reference 45, eq.4 and eq.5, the short period damping ratio,  $\zeta_s$  and short period frequency,  $\omega_s$  can be computed.

$$\zeta_s = \frac{\frac{C_{l_{awf}}}{4} + \frac{C_{l_{at}}}{4} \left( \frac{l_T}{K_B} \right)^2 \left( 1 + \frac{d\varepsilon}{d\alpha} \right) \frac{S_T}{S}}{\left( \frac{MC_{l_{awf}}c}{2\rho SK_B^2} \left( H_0 - H + \left( 0.9 \frac{C_{l_{at}}}{C_{l_{awf}}} \frac{l_T}{c} \left( 1 - \frac{d\varepsilon}{d\alpha} \right) + \frac{\rho SC_{l_{at}} l_T^2}{2Mc} \right) \frac{S_T}{S} \right) \right)^{\frac{1}{2}}} \quad (\text{A.18.32})$$

$$\omega_s = \rho \frac{SV}{2\pi M} \left( \frac{MC_{l_{awf}}c}{2\rho SK_B^2} \left( H_0 - H + \left( 0.9 \frac{C_{l_{at}}}{C_{l_{awf}}} \frac{l_T}{c} \left( 1 - \frac{d\varepsilon}{d\alpha} \right) + \rho \frac{SC_{l_{at}} l_T^2}{2Mc} \right) \frac{S_T}{S} \right) \right)^{\frac{1}{2}} \quad (\text{A.18.33})$$

We take two extreme conditions, initial and final cruise to facilitate calculations.

#### A. Initial cruise, $\alpha_T$ is -0.376

According to reference 45, we take +, - 0.04 of aircraft C.G. and assume  $s_r/s$  is 0.15, then take +, - 0.05 of  $s_r/s$ , to calculate the short period damping ratio,  $\zeta_s$  and short period frequency.

After calculation, by using eq. A.18.32 and A.18.33, the short period damping ratio,  $\zeta_s$  and short period frequency,  $\omega_s$  are obtained.

| $H$   | $s_r/s$ | $\zeta_s$ |
|-------|---------|-----------|
| 0.314 | 0.15    | 0.24      |
| 0.314 | 0.2     | 0.223     |
| 0.314 | 0.25    | 0.218     |
| 0.314 | 0.3     | 0.218     |
| 0.354 | 0.15    | 0.297     |
| 0.354 | 0.2     | 0.252     |
| 0.354 | 0.25    | 0.238     |

|       |      |       |
|-------|------|-------|
| 0.354 | 0.3  | 0.233 |
| 0.394 | 0.15 | 0.431 |
| 0.394 | 0.2  | 0.297 |
| 0.394 | 0.25 | 0.264 |
| 0.394 | 0.3  | 0.252 |

( a ) Damping ratio

| $H$   | $\omega_s$ | $\zeta_s$ |
|-------|------------|-----------|
| 0.314 | 0.385      | 0.24      |
| 0.314 | 0.484      | 0.223     |
| 0.314 | 0.566      | 0.218     |
| 0.314 | 0.638      | 0.218     |
| 0.354 | 0.312      | 0.297     |
| 0.354 | 0.428      | 0.252     |
| 0.354 | 0.519      | 0.238     |
| 0.354 | 0.597      | 0.233     |
| 0.394 | 0.215      | 0.431     |
| 0.394 | 0.364      | 0.297     |
| 0.394 | 0.467      | 0.264     |
| 0.394 | 0.552      | 0.252     |

( b ) Frequency

Table A.18.9 Short period oscillation

**B. Final cruise,  $\alpha_T$  is -0.24**

As last section, by using eq. A.18.32 and A.18.33, the short period damping ratio,  $\zeta_s$  and short period frequency,  $\omega_s$  are obtained.

| $H$   | $s_r/s$ | $\zeta_s$ |
|-------|---------|-----------|
| 0.314 | 0.15    | 0.252     |
| 0.314 | 0.2     | 0.233     |
| 0.314 | 0.25    | 0.228     |
| 0.314 | 0.3     | 0.228     |
| 0.354 | 0.15    | 0.31      |
| 0.354 | 0.2     | 0.264     |



|       |      |       |
|-------|------|-------|
| 0.354 | 0.25 | 0.249 |
| 0.354 | 0.3  | 0.244 |
| 0.394 | 0.15 | 0.448 |
| 0.394 | 0.2  | 0.31  |
| 0.394 | 0.25 | 0.276 |
| 0.394 | 0.3  | 0.263 |

( a ) Damping ratio

| $H$   | $\omega_s$ | $\zeta_s$ |
|-------|------------|-----------|
| 0.314 | 0.405      | 0.252     |
| 0.314 | 0.509      | 0.233     |
| 0.314 | 0.595      | 0.228     |
| 0.314 | 0.671      | 0.228     |
| 0.354 | 0.328      | 0.31      |
| 0.354 | 0.451      | 0.264     |
| 0.354 | 0.546      | 0.249     |
| 0.354 | 0.627      | 0.244     |
| 0.394 | 0.227      | 0.448     |
| 0.394 | 0.383      | 0.31      |
| 0.394 | 0.492      | 0.276     |
| 0.394 | 0.581      | 0.263     |

( b ) Frequency

Table A.18.10 Short period oscillation

### A.18.5.2 Phugoid in cruise

From reference 45, eq. A.16, the damping ratio in phugoid motion may be solved.

$$\zeta_L = F_1 C_{D0} + F_2 C_L^2 \quad (\text{A.18.34})$$

In eq. A.18.34, there are two variables,  $F_1$  and  $F_2$  should be given in advance. From reference 45, appendix A.4.2,  $F_1$  and  $F_2$  may be solve by following equations.

$$F_1 = 0.5 + \frac{\omega}{2(2\omega + C_{L\alpha\omega} \nu)} \quad (\text{A.18.35})$$

$$F_2 = \left( \frac{3\omega - C_{L\alpha\omega} \nu}{4\omega + 2C_{L\alpha\omega} \nu} \right) \frac{C_D - C_{D0}}{C_L^2} - \frac{C_{L\alpha\omega}}{2} \frac{\left( \omega - \nu^2 \left( 1 + \frac{d\varepsilon}{d\alpha} \right) \right)}{(2\omega + C_{L\alpha\omega} \nu)^2} \quad (\text{A.18.36})$$

Where

$$\omega = \frac{Mc}{\rho SK_B^2} \frac{C_{L\alpha wf}}{2} K_n \quad (\text{A.18.37})$$

$$\nu = \frac{C_{L\alpha T}}{2} \frac{S_T}{S} \left( \frac{l_T}{K_B} \right)^2 \quad (\text{A.18.38})$$

and from reference 45, eq.A.11, the static fixed margin,  $k_n$  is

$$K_n = H_0 - H + V_T \frac{C_{L\alpha T}}{C_{L\alpha wf}} \left( 1 - \frac{d\varepsilon}{d\alpha} \right) \quad (\text{A.18.39})$$

As before, we take two extreme points, during cruise, to calculate the phugoid motion.

Since the configuration of the HALE is fixed, so the value of  $s_T/s$  is 0.19, the location of horizontal surface lift after of centre of gravity,  $l_T$ , is 13.77 ft, the pitch radius of gyration,  $K_B$ , is 6.66 ft, and tail volume coefficient,  $V_T$  is 0.53.

#### A. Initial cruise

The value of wing body lift curve slope,  $C_{L\alpha wf}$  is 6.01, tailplane lift curve slope,  $C_{L\alpha T}$  is 4.6, aircraft mass is 282.7 lb-sec<sup>2</sup>/ft, density,  $\rho$  is 0.00032 slugs/ft<sup>3</sup>, down wash gradient at the horizontal tail,  $\frac{d\varepsilon}{d\alpha}$  is 0.315, from eq A.18.24, wing body aerodynamic centre,  $H_0$  is 0.227, the drag coefficient,  $C_D$  is 0.0277, the zero drag coefficient,  $C_{D0}$  is 0.0147, and lift coefficient,  $C_L$  is 0.662.

If we take two locations of aircraft C.G to calculate short period frequency phugoid, one is -0.04 from aircraft C.G., the other is at aircraft C.G.

##### a. -0.04 from aircraft C.G. ( $H$ is 0.314 )

From eq. A.18.39, the static fixed margin,  $k_n$ , is 0.192. Then from eq. A.18.38 and A.18.39, we can obtain the value of  $\omega$  and  $\nu$  is 257.86 and 2.896 respectively.

Substituting those values into eq. A.18.35, and A.18.36, then the value of  $F_1$  and  $F_2$  is 0.742 and 0.01845.

Finally, from eq. A.18.34, the value of damping ratio in the phugoid motion,  $\zeta_L$ , is 0.019.

##### b. At aircraft C.G. ( $H$ is 0.354 )

From eq. A.18.39, the static fixed margin,  $k_n$ , is 0.1517. Then from eq. A.18.38 and A.18.39, we can obtain the values of  $\omega$  and  $\nu$  as 204.05 and 2.896 respectively.

Substituting these variables into eq. A.18.35 and A.18.36, then the value of  $F_1$  and  $F_2$  is 0.7398 and 0.01755.

Finally, from eq. A.18.34, the value of damping ratio in the phugoid motion,  $\zeta_L$ , is 0.0186.

### **B. Final cruise**

The value of wing body lift curve slope,  $C_{L_{awf}}$  is 5.862, tailplane lift curve slope,  $C_{L_{at}}$  is 4.514, aircraft mass is 141.3 lb-sec<sup>2</sup>/ft, density,  $\rho$  is 0.00018 slugs/ft<sup>3</sup>, downwash gradient at the horizontal tail,  $\frac{d\epsilon}{d\alpha}$  is 0.3152, from eq. A.18.36, wing body aerodynamic centre,  $H_0$  is 0.2266, the drag coefficient,  $C_D$  is 0.028, the zero drag coefficient,  $C_{D0}$  is 0.0037, and lift coefficient,  $C_L$  is 0.59.

As we did in section A.18.5.1, two extreme case of aircraft C.G. are taken into consideration.

#### **a. -0.04 from aircraft C.G. ( $H$ is 0.314 )**

From eq. A.18.39, the static fixed margin,  $k_n$  is 0.192. Then from eq. A.18.37 and A.18.38, we can obtain the values of  $\omega$  and  $\nu$  is 224.13 and 2.84 respectively.

Substituting these values into eq. A.18.35, and A.18.36, then the value of  $F_1$  and  $F_2$  is 0.741 and 0.0467.

Finally, from eq. A.18.34, the value of damping ratio in the phugoid motion,  $\zeta_L$ , is 0.0189.

#### **b. At aircraft C.G. ( $H$ is 0.354 )**

From eq. A.18.39, the static fixed margin,  $k_n$ , is 0.152. Then from eq. A.18.38 and A.18.39, we can obtain the values of  $\omega$  and  $\nu$  as 177.5 and 2.84 respectively.

Substituting these variables into eq. A.18.35 and A.18.36, then the value of  $F_1$  and  $F_2$  is 0.739 and 0.0453.

Finally, from eq. A.18.34, the value of damping ratio in the phugoid motion,  $\zeta_L$ , is 0.0184.

## Appendix B

### Validation and application

This appendix contains the details behind the use of the program covered by chapter 2, in investigation of the Tier-II Plus aircraft. As explained in chapter 3, this can be used to some extent to validate the program.

#### B.1 Weight estimation

Before calculating the weight, the limit load factor,  $N_z$ , should be assumed first. In order to select a proper limit load factor for Tier-II Plus, we use the same methods as in appendix A.5. This method calculates the limit load factor at the condition when the aircraft first climbs through 20,000 ft and as it descends through 20,000 ft when the mission is finished.

##### Method a (reference 10)

###### a.1 Climb through 20,000 ft:

Assume the aircraft weight,  $W$ , to be 22,800 lbs, at 20,000 ft the density is 0.001267 lb-s<sup>2</sup>/ft<sup>4</sup>, aircraft lift curve slope,  $C_{L\alpha}$ , is 7.2375, aircraft true velocity,  $V_T$ , is 463 fps, mean aerodynamic chord is 4.99 ft, wing area,  $S$ , is 523.8 ft<sup>2</sup>, and standard vertical gust,  $U_{de}$ , is 30 fps. Substituting these variables into eq A.5.1 to eq A.5.6, after computation, the value of mass ratio,  $\mu$ , is 59.09, gust alleviation factor,  $K$ , is 0.808,

upward gust velocity,  $U$ , is 24.24 fps, the change in aircraft lift,  $\Delta L$ , is 26949.19 lbf. the change in load factor,  $\Delta n$ , is 1.182, and the limit load factor,  $N_z$ , is 2.182.

**a.2 Descent through 20,000 ft :**

Assume the aircraft weight to be 11,000 lbs, aircraft velocity and lift curve slope is the same. By using eq. A.5.1 to eq. A.5.6, the value of  $\mu$  is 28.51,  $K$  is 0.742,  $U$  is 22.26,  $\Delta n$  is 2.25. So, the limit load factor is 3.225.

**Method b (reference 27)**

**b.1 Climb through 20,000 ft :**

Assume the aircraft weight, density and lift curve slope to be the same as in method a, and the equivalent aircraft velocity,  $V_e$ , is 338 fps. Substituting these variables into eq. A.5.7, we obtain the value of the limit load factor is 2.07.

**b.2 Descent through 20,000 ft :**

Assume the aircraft weight to be 11,000 lbs, aircraft velocity is the same. After calculation the value of the limit load factor is 3.21.

Compare the result of limit load factor from both in method a and b. When aircraft climbs through 20,000 ft, the limit load factor is less than 3. When the aircraft descends through 20,000 ft, the limit load factor is higher than 3. As stated in appendix A.5, during climb, the aircraft weight is much higher than descend. The critical situation is at climb phase. Therefore, take 3 as limit load factor should be reasonable.

In accordance with chap 2, in preliminary design, general aviation method is used to compute aircraft weight.

Before computation, some parameters, from chap. 3.5, such as wing area, weight of fuel in wing, wing span, wing area, aspect ratio, taper ratio, tailplane area, fuselage length, aircraft design gross weight are given. The wing average thickness ratio,  $t/c$ , is 0.15, wing sweep at 25 percent M.A.C.,  $\Lambda$ , is 5 degrees, limit load factor,  $N_z$ , during cruise is 3.

By using eq. A.5.8, the fuselage wetted area,  $S_f$ , is 567.69 ft<sup>2</sup>. The tail arm length (wing quarter-MAC to tail quarter-MAC.  $L_t$ ) is 18.5 ft, limit landing load factor,  $N_l$ , is 2, landing design gross weight,  $W_l$ , is 12,000 lbs, length of main landing gear,  $L_m$ , is 46.43 inches, nose gear length,  $L_N$ , is 39.6 inches, engine weight,  $W_{en}$ , is 1,581 lbs, number of engines,  $N_{en}$ , is 1, integral tanks volume,  $V_i$ , is 8,172.63 lbs, maximum range is 3,000 nautical miles.

In accordance with fig. 3.2.1, the Tier-II Plus starts to cruise at 50,000 ft, and cruise speed is 0.6 M. Based on the above assumption, the aircraft component weight is to be estimated.

### **B.1.1 Wing group**

By using the eq. A.5.10, the wing weight is 3,850.2 lbs

### **B.1.2 Empennage group**

By using eq. A.5.11, the tailplane weight is 209.55 lbs

### **B.1.3 Fuselage group**

By using eq. A.5.13, the fuselage weight is 722.4 lbs

### **B.1.4 Main landing gear**

By using eq. A.5.14, the main landing gear weight is 382 lbs

### **B.1.5 Nose landing gear**

By using eq. A.5.15, the nose landing gear weight is 91.24 lbs

### **B.1.6 Install engine**

By using eq. A.5.16, the install engine weight is 2,291.88 lbs

### **B.1.7 Fuel system**

By using eq. A.5.17, the fuel system weight is 742.73 lbs

### **B.1.8 Flight control system**

By using eq. A.5.18, the flight control system weight is 490.48 lbs

### **B.1.9 Hydraulic system**

By using eq. A.5.19, the hydraulic system weight is 22.914 lbs

### **B.1.10 Miscellaneous**

By using eq. A.5.20, the miscellaneous weight is 611.99 lbs

After computation, we summarise the aircraft component weight into table B.1.1.

When comparing the result, gross weight, 24,919.154 lbs, to the gross weight provided by the specification, 22,914 lbs, the difference is 8.7 percent. This result is higher than the required 3 percent. Therefore, it is not acceptable.

| component                | weight (lbs) | percent total weight |
|--------------------------|--------------|----------------------|
| a. total structure       | 5255.39      |                      |
| b. total propulsion      | 3034.58      |                      |
| c. total fixed equipment | 1125.38      |                      |
| A. weight empty (a+b+c)  | 9415.35      | 0.378                |
| B. mission payload       | 2000         | 0.08                 |
| C. fuel                  | 13503.8      | 0.542                |
| gross weight (A+B+ C)    | 24919.154    |                      |

Table B.1.1 Aircraft component weight

The formulas which we used to estimate aircraft initial weight is based on the conventional metal material. If using composite material and advanced technology, the aircraft weight can be reduced. Reference 43 suggests if the aircraft is in “all-composite”, then significant weight saving can be achieved. Reference 38 stated that the wing is all composite and fuselage is standard aluminium with composite secondary structure of Tier-II Plus. Therefore, the value of 0.8, 0.7 and 0.88 has been taken to multiply by the wing weight, fly control system (assume fly by wire), and to other structure weight respectively. The results are listed in table B.1.2.

| Component              | Weight, lbs | Percent total weight |
|------------------------|-------------|----------------------|
| wing                   | 3080.162    |                      |
| tailplane              | 184.4       |                      |
| fuselage               | 635.71      |                      |
| main landing gear      | 336.15      |                      |
| nose landing gear      | 80.292      |                      |
| engine install         | 2016.86     |                      |
| fuel system            | 653.6       |                      |
| hydraulic system       | 22.9        |                      |
| flight control         | 343.34      |                      |
| inst.+avion.+elect.    | 530.95      |                      |
| A. weight empty        | 7884.38     | 0.337                |
| B. mission payload     | 2000        | 0.085                |
| C. fuel                | 13503.8     | 0.577                |
| gross weight ( A+B+C ) | 23388.2     |                      |

Table B.1.2 Modified aircraft component weight

When comparing the above result for gross weight, 23,388.2 lbs, to the gross weight provided by the specification, 22,914 lbs, the difference is 2 percent. Therefore, the results is acceptable.

## **B.2 Balance calculation**

After the aircraft component weight is obtained. According to reference 1, Chap 2, following is the aircraft C.G. movement estimation.

### **B.2.1 Component C.G. Locations**

From section B.1.10, the aircraft weight breakdown has been obtained. According to reference 2, Chap 10, each component C.G. can be estimated as follows (reference axis is 50 inches ahead of aircraft ) :

#### **B.2.1.1 Fuselage:**

By using eq. A.6.1, from the reference axis to the fuselage C.G. location,  $CG_{\text{fuse}}$ , is 257.79 inches.

#### **B.2.1.2 Wing :**

By using eq. A.6.2, from the reference axis to the aircraft wing C.G. location,  $CG_{\text{wing}}$ , is 337.32 inches.

#### **B.2.1.3 Tailplane :**

Since the profile of tailplane is like wing, by using eq. A.6.2, from the reference axis to the aircraft tailplane C.G. location,  $CG_{\text{tail}}$ , is 563.32 inches.

#### **B.2.1.4 Engine :**

By using eq. A.6.5, from the reference axis to the aircraft engine C.G. location,  $CG_{\text{engine}}$ , is 413.6 inches.

#### **B.2.1.5 Nose landing gear & Main landing gears' C.G.:**

The C.G. location of main and nose landing gear is at gear up location. From fig. 4.4.1 (Tier-II Plus three view diagram), landing gear C.G. location from reference axis to nose and main landing gear is around 180 and 357 inches respectively.

#### **B.2.1.6 Fixed equipment :**

Assume the C.G. location of fixed equipment to be 84 inches from aircraft nose. Therefore, from reference axis to fixed equipment, C.G. location is 134 inches.



### B.2.1.7 Payload C.G. :

Assume the payload is located at forward and aft. fuselage respectively. Therefore, the C.G. locations of payload from reference axis are 219.2 and 280.4 inches.

So far, each component C.G. location and component weight has been obtained, table B.2.1 is the results.

| TYPE OF COMPONENT | Wi (lbs) | Xi (in) | Wi*Xi       |
|-------------------|----------|---------|-------------|
| fuselage group    | 635.71   | 257.792 | 163880.95   |
| wing group        | 3080.162 | 337.32  | 1039000.246 |
| tailplane         | 184.4    | 563.32  | 103876.208  |
| engine group      | 2016.86  | 413.6   | 834173.296  |
| nose landing gear | 80.292   | 180     | 14452.56    |
| main landing gear | 336.15   | 357.5   | 120173.625  |
| fixed equipment   | 897.19   | 134     | 120223.46   |
| fuel weight       | 13503.8  | 337.32  | 4555102     |
| fuel tank weight  | 653.6    | 337.32  | 220472.4    |
| forward payload   | 800      | 219.2   | 175360      |
| aft. payload      | 1200     | 280.4   | 336480      |
| Take Off Weight   | 23388.2  |         | 7683195     |

Table B.2.1 Each component weight & C.G. location

### B.2.2 Aircraft C.G. movement

As we did in chap 2.6.1, following four scenarios has been taken, by using eq A.6.10 to eq. A.6.13 to calculate the C.G. movement. After calculation the following results are obtained.

- a. empty weight without payload (enp) is 331.83 inches
- b. empty weight with payload (ewp) is 315.23 inches
- c. take-off weight without payload (tnp) is 335.3 inches
- d. take-off weight with payload (twp) is 327.98 inches

Aircraft C.G. movement should within wing MAC. Because the range of wing MAC is from 313.36 to 373.258 inches. And from above calculation, the C.G. movement is from 315.23 to 335.3 inches. Therefore, the aircraft C.G. movement is within wing MAC.

### B.3 Wing body incidence

During cruise, according to requirement, the speed of Tier-II plus is 580.84 fps (0.6 M). As in appendix A.7, as long as the air flow over airfoil remains entirely subsonic, the free stream,  $M_\infty$ , on airfoil should be modified by a factor, the Prandtl-Glauert compressibility factor,  $\beta$ . By using eq. A.7.1, the value of  $\beta$  is 0.8.

Before calculation, the wing and tailplane basic data should be given first.

#### B.3.1 Wing

The wing airfoil of wing root as MS(1)-0317 and wing tip as MS(1)-0313 is the same as in appendix A.7.1, therefore, from table A.7.1, the airfoil data is obtained.

As the same procedure in appendix A.7.1, the theoretical airfoil lift curve slope,  $C_{l\alpha 0T}$ , in incompressible flow estimated is 6.996.

Some airfoil data, such as inner trailing edge angle ( $\tan(\frac{1}{2}\tau_a)$ ), outer trailing edge angle ( $\tau$ ) and the theoretical airfoil lift curve slope in incompressible flow is constant. Others, such as Reynolds number, airfoil lift curve slope, etc., are varied with speed and height. However, the following calculation is an example, and should bear in mind these values will vary as flight condition changes.

From reference 38, the cruise speed of Tier-II Plus is 0.6 M, if the altitude is 65,000 ft, by using eq. A.7.4, the Reynolds number on airfoil is 1,717,565.4 ( $\rho$  is 0.000176,  $l$  (mac) is 4.99 ft and dynamic viscosity,  $\mu$ , is 0.000000297 lb-s/ft<sup>2</sup>).

As stated in appendix A.7.1. If the aircraft leading edge sweep angle less than 10 degrees, the transition will occur when Reynolds number reach 12 million. For Tier-II Plus, its leading edge sweep angle is 7 degrees and the Reynolds number, during cruise, is  $1.7 \times 10^6$ , far less than 10 million. Therefore, the air flow on the airfoil can maintain laminar flow until reach the airfoil maximum thickness point. For both MS-0317 and MS-0313 airfoil, the thickness point is at 0.375 percent chord. That means the chordwise location of boundary layer transition ( $x/c$ ) point at wing is 0.375. Substituting above data into eq. A.7.5, the result of the airfoil lift curve slope in incompressible flow divided by the theoretical airfoil lift curve slope in incompressible

flow,  $\frac{(C_{l\alpha})_{aM=0}}{C_{l\alpha 0T}}$ , is 0.8368.

By using eq. A.7.6, the airfoil lift curve slope in incompressible flow,  $(C_{l\alpha})_{aM=0}$ , is 5.854.

By using eq. A.7.7, modified by Prandle-Grauert factor, airfoil lift curve slope in

compressor flow,  $(C_{l\alpha})_{atM}$ , is 7.3175.

By using eq. A.7.8, the value of constant K is 0.9316.

By using eq. A.7.9, the value of wing lift curve slope is 6.649.

By using eq. A.7.10 to eq. A.7.12, the value of constant  $K_A$ ,  $K_\lambda$ ,  $K_h$  is 0.0348, 1.28 and 1.4412 respectively.

Finally, by using eq. A.7.13, the value of down wash gradient at the horizontal tail is 0.21.

### B.3.2 Tailplane

The tailplane of Tier-II Plus is the same as HALE's tailplane, therefore, from table A.7.2, the tailplane airfoil data is obtained.

Following the same process in appendix A.7.2, the value of theoretical airfoil lift curve slope in incompressible flow,  $C_{l\alpha 0T}$ , is 6.877.

By using eq. A.7.4, the Reynolds number of tailplane during cruise is 1.5 million ( $l_T$  (tailplane MAC) is 4.328 ft and dynamic viscosity,  $\mu$ , is 0.000000297 lb-s/ft<sup>2</sup>).

The Mach number of air flow on the tailplane is less than 10 million and the tailplane leading edge sweep angle is 7 degrees, less than 10 degrees. Therefore, as the same in appendix A.7.2, by using eq. A.7.5, the value of the airfoil lift curve slope in incompressible flow divided by the theoretical airfoil lift curve slope in incompressible flow,  $\frac{(C_{l\alpha})_{atM=0}}{C_{l\alpha 0T}}$ , is 0.86.

By using eq. A.7.6, the value of tailplane airfoil lift curve slope is 5.919.

By using eq. A.7.7, modified by Prandle-Grauer factor, the value of tailplane airfoil lift curve slope,  $(C_{l\alpha})_{atM}$  is 7.399.

By using eq. A.7.8, the value of constant  $K_T$  is 0.942.

The tailplane of Tier-II Plus tilts 39 degrees from vertical line and there is no horizontal tailplane. Therefore, calculating the value of horizontal tailplane lift curve slope, we should take the project area of whole tailplane.

The tailplane area,  $S_T$ , and tailplane aspect ratio,  $A_T$  as

$$S_T = C_T * \frac{b_T}{2} - \tan(\phi_{le}) * \frac{b_T}{2} * \frac{b_T}{2} * 0.5 - \tan(\phi_{te}) * \frac{b_T}{2} * \frac{b_T}{2} * 0.5 \quad (B.3.1)$$

In eq. B.3.1,  $\phi_{le}$  is tailplane leading edge sweep angle, 7 degrees,  $\phi_{te}$  is tailplane trailing edge sweep forward angle, 5.37 degrees,  $C_T$  is tailplane root chord, 5.8 ft, and  $b_T$  is tailplane span, 31 ft.

Substituting these variables into eq. B.3.1, the value of tailplane area is 127.72 ft<sup>2</sup>.

By using eq. A.4.3, the value of tailplane aspect ratio,  $A_T$ , is 7.524.

The tailplane of Tier-II Plus tilts 39 degrees from the vertical axis, therefore, the project area,  $S_{TP}$  and projected tailplane aspect ratio,  $A_{TP}$ , multiplied by  $\sin(39)$ , becomes 80.378 ft<sup>2</sup> and 4.735 respectively.

Substituting these variables into eq. A.7.9, the value of tailplane lift curve slope,  $C_{L\alpha T}$ , is 4.583.

By using eq. A.7.16, the value of  $K_{wf}$  is 1.0006.

By using eq. A.7.15, the value of aircraft wing body lift curve slope,  $C_{L\alpha wf}$ , is 6.653.

By using eq. A.7.14, the value of aircraft lift curve slope,  $C_{L\alpha}$ , is 7.182.

### B.3.3 wing body incidence

In order to obtain a reasonable wing body incidence during cruise, two extreme cases has been taken, the initial cruise, and final cruise. Their weight assumed to be around 19,000 lbs, and 11,500 lbs respectively.

Substituting this two weight into eq. A.8.17, and the altitude assumed to be 65,000 ft, the lift coefficient at initial and final cruise as 1.22 and 0.739 is obtained.

Substituting these variables into eq. A.7.17, we obtain at initial and final cruise, the value of wing body incidence is 7.948 and 4.1 respectively.

Taking the average value of this two extreme cases, the value of wing body incidence angle during conceptual design is around 6 degrees.

### B.3.4 Oswald efficiency

It is hard to find one method to estimate the value of Oswald efficiency as aspect ratio higher than 20. Therefore, as in appendix A.7.4, two methods are used to estimate the Oswald efficiency.

a. The aspect ratio of Tier-II Plus is 25.78. Substituting its value into eq. A.2.6, after calculation, the value of Oswald efficiency is 0.804.

b. Wing root chord,  $C_r$ , is 6.875 ft, wing tip chord is 2.38 ft, the taper ratio,  $\lambda$ , is 0.35. Substituting these variables into eq. A.7.19, the increment of plane wing vortex induced drag coefficient due to additional lift,  $\delta$ , is 0.02483.

By using eq. A.7.20, the value of Oswald efficiency is 0.7504.

Since the results, from a and b, are not very reliable, reference 66 suggests, in terms of conservative, 0.78 is the figure not to far from estimation. Therefore, 0.78 has been taken as Oswald efficiency to perform conceptual design.

## B.4 Aircraft component drag estimation

In this paragraph, the same methods as stated in appendix A.8 is used to estimate aircraft component drag.

As in appendix A.8.1, for Tier-II Plus, in preliminary design phase, we only dealing with drag coefficient of wing, fuselage, empennage, nacelle, trim and landing gear. The total aircraft drag is varied with aircraft altitude, weight and velocity. Therefore, initial cruise has been taken as an example (the aircraft weight, flight speed, altitude are assumed to be 19,000 lbs, 0.6 M, 65,000 ft respectively) to demonstrate how to estimate total aircraft drag.

### B.4.1 Wing drag coefficient prediction

In subsonic, wing drag coefficient can be divided into zero-lift drag coefficient and lift drag coefficient as presented in eq. A.8.3.

#### B.4.1.1 Wing drag coefficient due to zero drag

As in appendix A.8.1.1.1, there are six variables in eq. A.8.4 should be given first.

##### a. Wing/fuselage interference factor, $R_{wf}$

During cruise, the altitude is 65,000 ft ( $\rho$  is 0.000176 slugs/ft<sup>3</sup>),  $l$  (fuselage) is 44.4 ft, velocity is 580.84 fps and dynamic viscosity,  $\mu$ , is 0.000000294 lb-s/ft<sup>2</sup>. Substituting these variables into eq. A.7.4, the fuselage Reynolds number,  $R_{Nfus}$ , is  $15 \times 10^6$ .

As in appendix A.8.1.1.1, the Reynolds number as  $1.53 \times 10^7$ , and Mach number as 0.6 are substituting into the pseudo program, after calculation, the value of wing/fuselage interference factor,  $R_{wf}$ , is 1.009.

##### b. Lifting surface correction factor, $R_L$ ,

As in appendix A.8.1.1.1, the Mach number and maximum thickness ratio wing chord sweep angle  $(t/c)_{max}$  of Tier-II Plus, during cruise, is 0.6 and 5 degrees respectively. Substituting these two variables into eq. A.8.5 to A.8.9, the value of lifting surface correction factor,  $R_L$ , is 1.1486.

##### c. Turbulent flat plate friction coefficient of the wing, $C_{fw}$

As in appendix A.8.1.1.1, during cruise,  $l$  (wing mac) is 4.99 ft, Substituting it into eq. A.7.4, the wing Reynolds number  $R_{Nwing}$  is 1,717,565.

The Reynolds number of airflow over airfoil is less than 12 million, and wing leading edge sweep angle is 7 degrees which less than 10 degrees. the airflow over airfoil can maintain laminar flow until it reaches maximum airfoil thickness ratio point, 0.375 chord length.

By using eq. A.8.10, we obtain the value of turbulent flat plate friction coefficient of the wing,  $C_{fw}$ , as 0.00389.

The wing Reynolds number  $R_{Nwing}$  is 1.7 million less than 12 million, by using the pseudo program in appendix A.8.1.1.1, after modification, the turbulent flat plate friction coefficient of the wing,  $C_{fw}$ , becomes 0.002988.

**d. Airfoil thickness location parameter,  $L'$**

As in appendix A.8.1.1.1, the airfoil thickness location parameter,  $L'$ , is 1.2

**e. Thickness ratio,  $t/c$**

As in appendix A.8.1.1.1, the thickness ratio ( $t/c$ ) for whole wing is 0.15.

**f. Wetted area of the wing,  $S_{wetw}$**

As in appendix A.8.1.1.1, the wing root chord,  $C_r$ , is 6.875 ft and fuselage diameter,  $d$ , is 4.72 ft. By using eq. A.8.12, wing/body adjacent area,  $S_{wb}$ , is 32.45 ft<sup>2</sup>. The wing area,  $S$ , is 523.8 ft<sup>2</sup>, by using eq. A.8.13, the value of wing net area,  $S_{net}$ , is 491.35 ft<sup>2</sup>

The wing taper ratio,  $\lambda$ , is 0.35, and the ratio of  $t/c$  ratios at tip and root,  $\tau$ , is 0.765.

Substituting these variables into eq. A.8.14, the value of Wetted area of the wing,  $S_{wetw}$ , is 1021.99 ft<sup>2</sup>.

Finally, substituting these variables, from a to f, into eq. A.8.4, the value of subsonic wing zero-lift drag coefficient,  $C_{D0w}$ , is 0.008316.

**B.4.1.2 Wing drag coefficient due to lift, ( $C_{Diw}$ )**

As in appendix A.8.1.1.2, there are four variables in eq. A.8.15 should be given first.

**a. Wing lift coefficient,  $C_{Lw}$**

The aspect ratio of Tier-II Plus is around 26, is much higher than conventional configuration aircraft. Reference 66 suggests using 1.03 multiplied by the value of airplane lift coefficient rather than using 1.05 (reference 6 suggested), as the wing lift coefficient.

By using eq. A.8.17, the airplane lift coefficient,  $C_L$ , is 1.224, after multiplied by 1.03, the value of wing lift coefficient,  $C_{Lw}$ , is 1.26.

**b. Wing twist angle,  $\epsilon_t$**

Positive for wash-in, for Tier-II Plus assumed to be -3/57.3 rad.

**c. Induced drag factor due to linear twist,  $v$**

The taper ratio of tier-II Plus is 0.35, therefore, taper ratio as 0.3 and 0.4 has been taken to calculate, then take the average value.

c.1 If taper ratio is 0.3, and  $\beta A = 20.6224$ , then

$$\begin{aligned} \nu_1 &= -9E - 19(\beta A)^2 + 0.00005\beta A - 0.0017 \\ &= -0.000669 \end{aligned} \quad (\text{B.4.1})$$

c.2 if taper ratio is 0.4,  $\beta A = 20.6224$ , then

$$\begin{aligned} \nu_2 &= 3.25E - 19(\beta A)^2 - 2.5E - 5\beta A + 0.0003 \\ &= -0.0002155 \end{aligned} \quad (\text{B.4.2})$$

The average value, from c.1 and c.2, is 0.000442.

#### **d. Zero-lift drag factor due to linear twist, $\omega$**

From reference 6, there are 7 charts for taper ratio range from 0.1 to 0.75 to calculate the value of zero-lift drag factor due to linear twist,  $\omega$ .

d.1 if taper ratio is 0.3,  $\beta A = 20.6224$

$$\begin{aligned} \beta\omega_1 &= -1.3E - 6(\beta A)^2 + 1.75E - 5\beta A + 0.00208 \\ &= 0.001909 \end{aligned} \quad (\text{B.4.3})$$

since  $\beta = 0.8$ , then

$$\omega_1 = \frac{0.001909}{\beta} = 0.002386$$

d.2 if taper ratio is 0.4,  $\beta A = 20.6224$

$$\begin{aligned} \beta\omega_2 &= -1.3E - 6(\beta A)^2 + 1.25E - 5\beta A + 0.00224 \\ \omega_2 &= \frac{0.002457}{\beta} \\ &= 0.002457 \end{aligned} \quad (\text{B.4.4})$$

The average value of zero-lift drag factor due to linear twist,  $\omega$ , from d.1 and d.2, is 0.002422.

Substituting these variables, from a to d, into eq. A.8.15, after calculation, the value of wing drag coefficient due to lift ( $C_{Diw}$ ) is 0.02553.

## **B.4.2 Fuselage drag Coefficient**

As in appendix A.8.1.2, eq. A.8.20 can be used to estimate the value of fuselage drag Coefficient.

### **B.4.2.1 Fuselage drag coefficient due to zero drag**

As stated in appendix A.8.1.2.1, in subsonic flight, eq. A.8.21 can be used to estimate the fuselage zero-lift drag coefficient.

Before calculating, there are three unknown variables.

**a. Wing/fuselage interference factor,  $R_{wf}$**

As in appendix A.8.1.2.1, for fuselage only,  $R_{wf}$  is 1.

**b. Turbulent flat plate skin friction coefficient of the fuselage,  $C_{ffus}$**

As in B.4.1.1.1, the fuselage Reynolds number,  $R_{Nfus}$ , is 15.3 million. By using eq. A.8.10, the turbulent flat plate skin friction coefficient of the fuselage is 0.002729.

**c. Finesse ratio and the wetted area of fuselage**

By using eq. A.5.9, the finesse ratio,  $\lambda_f$ , is 9.867, and by using eq. A.8.22, the wetted area of fuselage with cylindrical mid-sections,  $S_{wet fus}$ , is 567.7 ft<sup>2</sup>.

For Tier-II Plus, engine is install right on the fuselage, so the wetted area of fuselage should deduct these overlap area (assume engine nacelle diameter ( $e_d$ ) to be 4.72 ft), after calculation, its value becomes 484.6 ft<sup>2</sup>.

Substituting these variables, from a to c, into eq. A.8.20, after calculation, the value of fuselage zero-lift drag coefficient ( $C_{D0fus}$ ) is 0.002769.

**B.4.2.2 Fuselage drag coefficient due to lift ( $C_{Difus}$ )**

As stated in appendix A.8.1.2.1, in subsonic flight, eq. A.8.23 can be used to estimate the fuselage drag coefficient due to lift.

Before calculating, there are five unknown variables.

**a. Fuselage angle of attack,  $\alpha$**

As in appendix A.8.1.2.2, before calculating the value of fuselage angle of attack, two variables, zero-angle-of-attack lift coefficient,  $C_{L0}$ , and the aircraft lift curve slope, ( $C_{La}$ ), should be given first.

**a.1 Zero-angle-of-attack lift coefficient ( $C_{L0}$ )**

As stated in appendix A.8.1.2.2, both tailplane incidence angle, ( $i_h$ ), and the magnitude downwash angle, ( $\epsilon_{oh}$ ), are assumed to be zero, therefore, the zero-angle-of-attack lift coefficient ( $C_{L0}$ ) is equal to the zero-angle-of-attack lift coefficient of the wing-fuselage combination ( $C_{L0wf}$ ).

By using eq. A.8.26, the value of zero-angle-of-attack lift coefficient of the wing-fuselage combination can be estimated.

From paragraph B.3.2, the value of wing-fuselage lift curve slope ( $C_{Lawf}$ ) is 6.653, and wing-body incidence angle,  $i_w$ , is 6 degrees.



The taper ratio,  $\lambda$ , of Tier-II Plus is 0.35 and  $\Lambda_{c/4}$  is 5 degrees.

$$\alpha = 0.000004\lambda - 0.00000379 = -0.00000239$$

$$\beta = 0.00002\lambda + 0.00046893 = 0.000476$$

$$\gamma = -0.1224\lambda - 0.35171071 = -0.39455$$

Substituting these variables into eq. B.4.5, then the change in wing zero-lift angle of attack per degree of linear wing twist ( $\frac{\Delta\alpha_0}{\varepsilon_t}$ ) is

$$\frac{\Delta\alpha_0}{\varepsilon_t} = \alpha\Lambda_{c/4}^2 + \beta\Lambda_{c/4} + \gamma = -0.39176 \quad (\text{B.4.5})$$

As stated in appendix A.8.1.2.2, if the value of  $M^*\cos(\Lambda_{c/4})$  is great than 0.5, the thickness ratio as 0.14, the value of  $\frac{(\alpha_0)_M}{(\alpha_0)_{M=0.3}}$  is 0.8013, and the thickness ratio as

0.16, the value of  $\frac{(\alpha_0)_M}{(\alpha_0)_{M=0.3}}$  is 0.4063.

For Tier-II Plus, the value of  $M^*\cos(\Lambda_{c/4})$  is 0.6, greater than 0.5, and the thickness ratio is 0.15, thus, the average value of  $\frac{(\alpha_0)_M}{(\alpha_0)_{M=0.3}}$  is 0.6038.

Substituting these variables into eq. A.8.27, the wing zero-lift angle of attack,  $\alpha_{0Lw}$ , is -0.01923 (the more accurate calculation method is listed in paragraph A.17.1).

Substituting these variables into eq. A.8.26, the value of zero-angle-of-attack lift coefficient ( $C_{L0}$ ) is 0.8245.

#### a.2 Aircraft lift curve slope, ( $C_{L\alpha}$ )

From paragraph B.3, the aircraft lift curve slope is 7.182.

Assume the aircraft weight,  $W$ , to be 19,000 lbs, at 65,000 ft, flight Mach number is 0.6, and wing area,  $S$ , is 523.8 ft<sup>2</sup>. Substituting these variables into eq. A.8.24, the fuselage angle of attack is 0.0553 rad.

#### b. Fuselage base area, $S_{bfus}$

There is no base area at the end of Tier-II Plus fuselage, from reference 6 fig. 4.17, the value of  $S_{bfus}$  is zero.

#### c. Ratio of the drag of a finite cylinder to the drag of an infinite cylinder, $\eta$

By using eq. A.8.28, the value of the drag of a finite cylinder to the drag of an infinite cylinder is 0.6787.

**d. Experimental steady state cross-flow drag coefficient of a circular cylinder,  $C_{dc}$**

As in appendix A.8.1.2.2, d. the  $C_{dc}$  is constant, 1.2.

**e. Fuselage planform area,  $S_{p\ fus}$**

As stated in appendix A.8.1.2.2, e, the fuselage of Tier-II Plus is divided into three segment. first part, nose section. Second part middle body section. Last part tail section.

**e.1. Nose section**

According to the figure of Tier-II Plus, nose section is mixed by both parabola and cylinder section. We assume the length in parabola section is equal to 0.18 multiplied by the fuselage length and the cylinder section is equal to 0.2 multiplied by the fuselage length. Therefore, the nose section area is;

$$S_{pn} = 0.5 \left( \pi 0.18 l_f \frac{d_f}{2} \right) + d_f 0.2 l_f = 71.54 \text{ ft}^2$$

**e.2. Middle section**

$$S_{pm} = 0.42 l_f d_f = 88 \text{ ft}^2$$

**e.3. Tail section**

$$S_{pt} = 0.5 \left( \pi 0.2 l_f \frac{d_f}{2} \right) = 32.9 \text{ ft}^2$$

Therefore, the fuselage planform area is;

$$S_{pfus} = S_{pn} + S_{pm} + S_{pt} = 192.44 \text{ ft}^2$$

Finally, substituting these variables, from a to e, into eq. A.8.23, the fuselage drag coefficient due to lift ( $C_{Difus}$ ) is 0.000202.

### **B.4.3 Empennage drag coefficient**

As stated in appendix A.8.1.3, eq. A.8.29 can be used to estimate the value of empennage drag coefficient.

#### **B.4.3.1 Empennage zero-lift drag coefficient ( $C_{D0emp}$ )**

In subsonic flight, by using eq. A.8.4, the empennage zero-lift drag coefficient can be calculated, but substitute the wing by the tailplane.

Before calculating empennage zero-lift drag coefficient, there are five variables need to be given.

**a. Empennage/fuselage interference factor,  $R_{wfe}$**

As stated in appendix A.8.1.3.1 a.,  $R_{wfe}$  is equal to 1.

**b. Lifting surface correction factor,  $R_{L_{se}}$**

As stated in appendix A.8.1.3.1, b, for Tier-II Plus, during cruise its Mach number is 0.6 and maximum thickness ratio empennage chord sweep angle  $(\cos \Lambda_{\frac{1}{c}})_{\max}$  is 3 degrees. Substituting these variables into eq. A.8.5 to eq. A.8.8, the value of  $\alpha$  is 3.78,  $\beta$  is -11.2,  $\gamma$  is 11.1 and  $\tau$  is -2.55.

Substituting these variables into eq. A.8.9, the value of  $R_{L_{se}}$  is 1.1488.

**c. Turbulent flat plate friction coefficient of tailplane,  $(C_{fe})$**

As in appendix A.8.1.3.1, c., by using eq. A.7.4, during cruise the empennage Reynolds number is 1,489,871 (tailplane MAC ( $l_e$ ) is 4.328 ft).

By using eq. A.8.10, the value of turbulent flat plate friction coefficient of the empennage,  $(C_{fwt})$ , is 0.00398.

As stated in appendix A.8.1.1.1, if the Reynolds number is less than 12 million, and leading edge sweep angle less than 10 degrees, the turbulent flat plate friction coefficient can be further modified. For Tier-II Plus, Reynolds number at tailplane is 1.4 million, leading edge sweep angle is 3 degrees. Therefore, the turbulent flat plate friction coefficient of the wing,  $(C_{fwt})$ , becomes 0.003292.

**d. Airfoil thickness location parameter,  $L'_e$**

As in appendix A.8.1.3.1 d., the value of airfoil thickness location parameter is 1.2.

**e. Wetted area of the empennage,  $S_{we}$**

The tailplane tilts from vertical axis, if we assume there is no overlap area for empennage and tailplane, then the net area of tailplane is equal to tailplane area.

According to appendix B.3.2, tailplane area,  $S_T$ , is 127.722 ft<sup>2</sup>, tailplane taper ratio,  $\lambda_e$ , is 0.41 and ratio of thickness ratio at tip and root,  $\tau_e$ , is 1.

Substituting these variables into eq. A.8.14, the result of Wetted area of the empennage is 265 ft<sup>2</sup>.

Finally, substituting these variables, from a to e, into eq. A.8.4, after calculation, the value of empennage zero-lift drag coefficient,  $C_{D0emp}$ , is 0.002355.

**B.4.3.2 Empennage drag coefficient due to lift,  $C_{Diemp}$**

As stated in appendix A.8.1.3.2, eq. 8.32 can be used to estimate the empennage drag coefficient due to lift.

Before calculating, there are four variables unknown.

**a. Lift coefficient of tailplane,  $C_{LT}$**

By using eq. A.8.33, the amount of lift coefficient on tailplane can be calculated.

By using eq. A.7.13, the down wash gradient at the horizontal tail is 0.21 and from paragraph B.4.1.2.2, the fuselage angle of attack,  $\alpha$ , is 0.0553 radian.

Substituting these variables into eq. A.8.34, the value of  $\alpha_h$  is 0.04385, and zero-lift angle of attack,  $\alpha_{0Lh}$ , for NACA 0012 airfoil is zero, and from paragraph B.3.2, the tailplane lift curve slope,  $C_{L\alpha T}$ , is 4.583.

Substituting these variables into eq. A.8.33, the lift coefficient of tailplane,  $C_{LT}$ , is 0.2

**b. Aspect ratio of projected tailplane area,  $A_{TP}$ , the projected tailplane area,  $S_{TP}$**

From paragraph B.3.2, the projected tailplane area ( $S_{TP}$ ) and projected tailplane aspect ratio ( $A_{TP}$ ) are 80.378 ft<sup>2</sup> and 4.735 respectively.

**c. Oswald efficiency for empennage,  $e_h$**

Empennage leading edge sweep angle,  $(\cos^2 \Lambda_h)$ , is 7 degrees. Substituting this value into eq. A.8.35, the value of Oswald efficiency for empennage,  $e_h$ , is 0.75.

Substituting these variables, from a to c, into eq. A.8.32, the value of tailplane drag coefficient due to lift ( $C_{Diemp}$ ) becomes 0.0014.

**B.4.4 Trim drag prediction,  $C_{Dt}$**

As in appendix A.8.1.4, eq. A.8.36 can be used to calculate the trim drag coefficient.

**B.4.4.1 Trim drag due to lift**

As in appendix A.8.1.4.1, eq. A.8.37 may be used to calculate trim drag coefficient due to lift, and the unknown variable in eq. A.8.37 is the horizontal tail incremental lift coefficient required for trim,  $C_{Lh}$ .

As in appendix A.8.1.4.1, eq. A.8.38 may be used to calculate the value of  $C_{Lh}$ .

In eq. A.8.38, the value of  $H$  is 0.38,  $H_0$  is 0.2,  $l_T$  is 16.7 ft,  $C_{Lw}$  is 1.26, and  $C_{m0wb}$  is -0.086 (more accurate calculation is in paragraph A.17.1).

Substituting these variables into eq. A.8.38, the value of  $C_{Lh}$  is 0.0413.

Substituting this value into eq. A.8.37, the value of trim drag due to lift is 0.00063.

**B.4.4.2 Trim drag due to profile drag,  $C_{Dtp}$**

As the same reason presented in appendix A.8.1.4.2, this value is zero.

The trim drag prediction,  $C_{Dt}$ , is equal to the trim drag due to lift, as 0.00063.

### B.4.5 Engine nacelle drag coefficient

As stated in appendix A.8.1.5, eq. A.8.39 can be used to estimate engine nacelle drag coefficient.

#### B.4.5.1 Engine nacelle drag coefficient due to zero drag

As stated in appendix A.8.1.5.1, in subsonic flight, eq. A.8.40 can be used to estimate the value of the engine nacelle zero-lift drag coefficient.

In eq. A.8.40, the engine nacelle length,  $l_{nac}$ , is 17.5 ft, maximum engine nacelle diameter,  $d_{nac}$ , is 4.72 ft, the value of fuselage/nacelle interference factor,  $R_{wf}$ , is 1, and the value of wing area,  $S$ , is 523.8 ft<sup>2</sup>. Two variables still unknown.

##### a. Turbulent flat plate skin friction coefficient of the nacelle, $C_{fnac}$

As in appendix A.8.1.5.1, a. by using eq. A.7.4, the engine nacelle Reynolds number ( $R_{Nnac}$ ) is 6,023,525. By using eq. A.8.10, the turbulent flat plate skin friction coefficient of the nacelle,  $C_{fnac}$ , is 0.00317.

##### b. Wetted area of nacelle, $S_{wetnac}$

As in appendix A.8.1.5.1, b. by using eq. A.8.22, the value of wetted area of nacelle ( $S_{wetnac}$ ) is 165.97 ft<sup>2</sup> (assume nacelle finesse ratio,  $(\lambda_{nac})$ , nacelle diameter,  $(d_{nac})$ , and nacelle length  $(l_{nac})$  to be 3.71, 4.72 ft and 17.5 ft respectively).

The engine nacelle of Tier-II Plus is installed right on top of the fuselage, thus the wetted area of nacelle should deduct these overlap area, after calculation, the value of wetted area of nacelle becomes 83.37 ft<sup>2</sup>.

Substituting these variables, from a to b, into eq. A.8.37, the value of fuselage zero-lift drag coefficient,  $(C_{D0nac})$  is 0.0011.

#### B.4.5.2 Engine nacelle drag coefficient due to lift, $(C_{Dinac})$

As in appendix A.8.1.5.2, eq. A.8.23 may be used to calculate the value of engine nacelle drag coefficient due to lift.

In eq. A.8.23, apart from wing area, there are five variables unknown.

##### a. Nacelle angle of attack;

Assume engine nacelle angle of attack,  $\alpha_{nac}$ , is 1.5 degree.

##### b. Engine nacelle base area, $S_{b\ fus}$

The diameter of nacelle base area is assumed to be 2.88 ft, thus from reference 6, fig. 4.17, the area of nacelle is 6.535 ft<sup>2</sup>.

**c. Ratio of the drag of a finite cylinder to the drag of an infinite cylinder,  $\eta$**

By using eq. A.8.28, the value of  $\eta$  is 0.594.

**d. Experimental steady state cross-flow drag coefficient of a circular cylinder,  $C_{dc}$**

As in appendix A.8.1.5.2, d. the value of  $C_{dc}$  is 1.2.

**e. Engine nacelle planform area ( $S_{p\text{ nac}}$ )**

As in appendix A.8.1.5.2, e. we divided nacelle into two segment. From figure 3.4.1, the proportion of forward nacelle section is equal to 0.26 multiplied by the fuselage length, tail nacelle section is equal to 0.16 multiplied by the fuselage length.

By using 0.26 rather than 0.75 in eq. A.8.41, the forward nacelle section area is 54.488 ft<sup>2</sup>, and by using 0.16 rather than 0.25 in eq. A.8.42, the tail nacelle section area is 26.34 ft<sup>2</sup>, when add together, the engine nacelle planform area is 80.82 ft<sup>2</sup>.

Substituting these variables, from a to e, into eq. A.8.39, the value of nacelle drag coefficient due to lift, ( $C_{D\text{ nac}}$ ) is 0.0000187.

#### **B.4.6 Nacelle fuselage interference drag coefficient, $C_{Dnp}$**

The engine nacelle and fuselage is assumed adjacent to each other, and pylon is not existed. So, nacelle pylon drag coefficient is zero.

#### **B.4.7 Landing gear drag coefficient ( $C_{D\text{ gear}}$ )**

As in appendix A.8.1.6, the tire of main and nose landing gear for Tier-II Plus is the same tire as HALE's, therefore, by using eq. A.8.44, the landing gear tire drag can be calculated (the number of two main lading gear of Tier-II Plus is four, thus we use four instead of two in eq. A.8.44).

After calculation, the value of landing gear tire drag coefficient is 0.00357.

As stated in appendix A.8.1.7, the landing gear strut drag coefficient is around the tire drag coefficient multiplied by two, therefore, the landing gear strut drag coefficient is 0.00714. Adding together, the landing gear drag coefficient is 0.0107.

#### **B.4.8 Total drag coefficient**

As stated in appendix A.8.1.8, by using eq. A.8.45, the total airplane zero-drag coefficient of Tier-II Plus is 0.01454, and by using eq. A.8.46, the total airplane drag coefficient due to lift is 0.0278.

By adding together, the total airplane drag coefficient, at initial cruise, is 0.0423.

## B.5 Stall analysis

As stated in appendix A.9, following steps are taken to estimate the stall speed, at different altitude and weight, of Tier-II Plus.

### B.5.1. Maximum lift coefficient of the wing ( $C_{Lmax}$ )<sub>w</sub>

By using the same method stated in appendix A.9.1, the maximum lift coefficient of the wing may be estimated.

#### a. As in appendix A.9.1, a

The total spanwise lift coefficient can be divided into two parts - basic and additional lift coefficient.

Before estimating the value of basic and additional lift coefficient, there are six variables should be given first.

##### a.1 Wing zero-lift angle of attack, $\alpha_{0Lw}$

By using eq. A.8.27, the wing zero-lift angle of attack is -0.01923.

##### a.2 Airfoil zero-lift angle of attack, $\alpha_{0L}$

As in appendix A.9.1, a.2, the airfoil zero-lift angle of attack is -3 degrees.

##### a.3 Twist angle of a local section at y, out from center line, $\epsilon(y)$

By using eq. A.9.6, the twist angle at airfoil local section is varied with different wing span location, y.

##### a.4 Local airfoil lift curve slope, $C_{l\alpha}(y,t)$

As in appendix A.9.1, a.4, the local airfoil lift curve slope is assumed equal to mean value of airfoil lift curve slope,  $C_{L\alpha}$ .

By using eq. A.7.4, the mean value of Reynolds number on wing is 3,532,720 (50,000 ft,  $\rho$  is 0.00036 lb-s<sup>2</sup>/ft<sup>4</sup>,  $l$  (wing MAC) is 4.99 ft,  $V$  is 580.84 fps and  $\mu$  is 0.000000297 lb-s/ft<sup>2</sup>).

By using eq. A.7.3, the theoretical airfoil lift curve slope in incompressible flow ( $C_{l\alpha 0T}$ ) is 6.996.

By using eq. A.7.5, the value of  $\frac{(C_{l\alpha})_{atM=0}}{C_{l\alpha 0T}}$  is 0.8617.

By using eq. A.7.6, the airfoil lift curve slope (no compression),  $(C_{l\alpha})_{atM=0}$  is 6.028.

By using eq. A.7.7, the mean value of airfoil lift curve slope,  $(C_{l\alpha})_{atM}$  is 7.535.

As in appendix A.9.1, in order to simplify eq. A.9.5, the local lift curve slope is assumed equal to the mean lift curve slope of the whole wing ( $C_{l\alpha}(y) = C_{l\alpha}$ ).

a.5 By using eq. A.9.7, the local airfoil chord,  $C(y)$  can be calculated (the leading edge sweep angle and the trailing edge sweep angle of Tier-II Plus is 7 degrees backward and 2.365 degrees upward respectively).

Substituting these variables, from a.1 to a.5, into eq. A.9.4 and eq. A.9.5, then basic and additional lift coefficient can be obtained.

**b. Spanwise local airfoil maximum lift coefficient,  $C_{lmax}$**

As in appendix A.9.1, b. eq. A.9.8 can be used to estimate the spanwise local airfoil maximum lift coefficient.

The length of wing root ( $l$ ) is 6.875 ft, and wing tip ( $l$ ) is 2.38 ft. By using eq. A.7.4, Reynolds number at root and tip is 4,867,224 and 1,684,944 respectively.

From reference 28, fig 23, For MS(1)-0317 airfoil, the maximum lift coefficient is equal to 1.82 and 1.95, at Reynolds number is equal to  $4 \times 10^6$  and  $6 \times 10^6$  respectively.

By using eq. A.9.9, the maximum lift coefficient at wing root is 1.876.

From reference 28, fig. 18, For MS(1)-0313 airfoil, the lowest Reynolds number is  $2.1 \times 10^6$ , the value of maximum lift coefficient is 1.7. The Reynolds number at wing tip of Tier-II Plus is  $1.68 \times 10^6$  which is lower than  $2.1 \times 10^6$ , thus there is no value available. After carefully measurement, a approximate value 1.65 as the maximum lift coefficient has been taken.

By using eq. A.9.8, and eq. A.9.10, the spanwise local airfoil maximum lift coefficient, and the maximum lift coefficient of the wing, can be computed.

As in appendix A.9.1, the variables in eq. A.9.10 are varied along wingspan. Therefore, the value of maximum lift coefficient of the wing ( $C_{lmax}{}_w$ ) is also varied along spanwise direction, after calculation the results are presented in table B.5.1.

| semi-span ft | $C_{lbasic}$ | $C_{ladd}$ | $C_{lmax}$ | $C_{Lmaxw}$ |
|--------------|--------------|------------|------------|-------------|
| 0            | 0.125        | 0.917      | 1.876      | 1.909       |
| 4            | 0.111        | 0.937      | 1.861      | 1.867       |
| 8            | 0.098        | 0.957      | 1.845      | 1.826       |
| 12           | 0.084        | 0.976      | 1.83       | 1.788       |
| 16           | 0.07         | 0.995      | 1.814      | 1.752       |
| 20           | 0.057        | 1.014      | 1.798      | 1.718       |
| 24           | 0.043        | 1.031      | 1.783      | 1.687       |
| 28           | 0.03         | 1.047      | 1.767      | 1.659       |
| 32           | 0.016        | 1.061      | 1.752      | 1.635       |
| 36           | 0.003        | 1.071      | 1.736      | 1.618       |
| 40           | -0.011       | 1.076      | 1.721      | 1.61        |



|    |        |       |       |       |
|----|--------|-------|-------|-------|
| 44 | -0.025 | 1.07  | 1.705 | 1.617 |
| 48 | -0.038 | 1.046 | 1.689 | 1.652 |
| 52 | -0.052 | 0.985 | 1.674 | 1.751 |
| 56 | -0.065 | 0.831 | 1.658 | 2.075 |
| 58 | -0.072 | 0.578 | 1.65  | 2.978 |

Table B.5.1 Maximum lift coefficient illustration

As in appendix A.9.1, from table B.5.1, the minimum value of  $C_{L_{max}}$ , along semi-span is, at 40 ft, 1.61. Thus, the maximum lift coefficient of the wing ( $C_{L_{max}})_w$  is 1.61.

### B.5.2 Calculate airplane maximum lift coefficient ( $C_{L_{max}}$ )

As in appendix A.9.2, by using eq. A.9.11, the airplane maximum lift coefficient can be computed.

There are six variables in eq. A.9.11, and their values have been given from paragraph B.3.1, B.3.2. When substituting these variables into eq. A.9.11, the value of airplane maximum lift coefficient ( $C_{L_{max}}$ ) is 1.708.

### B.5.3 Stall speed

In order to calculate the stall characteristic, the airplane maximum lift coefficient is assumed to be constant as aircraft weight and height varies. By using eq. A.9.2, the stall speed at different Altitude and weight is showed in table B.5.2.

| Altitude,<br>ft | weight,<br>lbs | True $V_{stall}$ ,<br>knots | Altitude,<br>ft | weight,<br>lbs | True $V_{stall}$ ,<br>knots |
|-----------------|----------------|-----------------------------|-----------------|----------------|-----------------------------|
| 0               | 23000          | 87.003                      | 40000           | 23000          | 175.353                     |
| 0               | 20500          | 82.138                      | 40000           | 20500          | 165.549                     |
| 0               | 15500          | 71.422                      | 40000           | 15500          | 143.952                     |
| 0               | 13000          | 65.409                      | 40000           | 13000          | 131.832                     |
| 0               | 10500          | 58.785                      | 40000           | 10500          | 118.48                      |
| 10000           | 23000          | 101.243                     | 50000           | 23000          | 222.989                     |
| 10000           | 20500          | 95.582                      | 50000           | 20500          | 210.521                     |
| 10000           | 15500          | 83.112                      | 50000           | 15500          | 183.056                     |
| 10000           | 13000          | 76.115                      | 50000           | 13000          | 167.645                     |
| 10000           | 10500          | 68.406                      | 50000           | 10500          | 150.665                     |
| 20000           | 23000          | 119.192                     | 60000           | 23000          | 283.564                     |
| 20000           | 20500          | 112.527                     | 60000           | 20500          | 267.71                      |

|       |       |         |       |       |         |
|-------|-------|---------|-------|-------|---------|
| 20000 | 15500 | 97.847  | 60000 | 15500 | 232.784 |
| 20000 | 13000 | 89.609  | 60000 | 13000 | 213.186 |
| 20000 | 10500 | 80.533  | 60000 | 10500 | 191.594 |
| 30000 | 23000 | 142.239 | 65000 | 23000 | 319.768 |
| 30000 | 20500 | 134.287 | 65000 | 20500 | 301.89  |
| 30000 | 15500 | 116.767 | 65000 | 15500 | 262.505 |
| 30000 | 13000 | 106.937 | 65000 | 13000 | 240.405 |
| 30000 | 10500 | 96.106  | 65000 | 10500 | 216.056 |

Table B.5.2 Aircraft stall characteristics

## B.6 Engine analysis

Reference 26 stated that the propulsion system of Tier-II Plus is Allison AE3007. From reference 35, the basic engine data such as engine diameter, length, weight, mass flow rate, pressure ratio and bypass ratio is presented. However, these data is not enough to analyse aircraft climb and cruise performance. More detailed engine data such as specific engine fuel consumption and engine net thrust at different height and velocity is still needed.

As stated in appendix A.10, a true engine data map is hard to obtained. Therefore, an engine performance simulation program is an alternative to obtain approximate engine data map.

Reference 35 only supply with engine total pressure ratio, didn't state how to distribute total pressure ratio to fan and compressor. Besides, the turbine inlet temperature (TET) is not available. In order to gain reasonable engine data map, we tried to use different combinations of fan and compressor pressure ratio and TET to simulate. Finally, one combination which the result is close to real engine data has been chosen to perform engine simulation.

### B.6.1 Fixed turbine inlet temperature with different fan and compressor pressure ratio.

First we tried to fix turbine inlet temperature (TET), as 1440 °F, and substitute different pressure ratio into simulation program. Reference 35 states the bypass ratio of Allison AE3007 is 5, the total pressure is 23. Therefore, by distributing total pressure ratio evenly to fan and compressor, and run this engine program, following sea level static design point results, listed in table B.6.1, are obtained.

| Fan pressure | Comp. pressure | SFC lb/lb-hr | Sea Level Static State Net thrust, lbs |
|--------------|----------------|--------------|----------------------------------------|
| 1.5          | 15.333         | 0.427        | 6998                                   |
| 1.55         | 14.839         | 0.4212       | 7091.4                                 |
| 1.6          | 14.375         | 0.416        | 7174                                   |
| 1.65         | 13.939         | 0.4127       | 7230                                   |
| 1.7          | 13.529         | 0.411        | 7258                                   |

Table B.6.1 Sea level static thrust with different pressure ratio

In table B.6.1, we find that when the fan pressure ratio increases, the fuel flow and specific fuel consumption will decrease, the static net thrust will increase. From this point, it seems good to have higher value of fan pressure. But, when take practical situation into consideration, the single stage fan pressure ratio is usually not higher than 1.3. Reference 68 suggests, if the fan is wide chord (the fan of Allison AE3007 turbofan engine is featuring wide chord, clapperless blades). and it is a small turbofan engine, then it is possible to have a higher value of fan pressure ratio. Finally, we decide to select the result, which is close to real value of Allison AE3007, of fan pressure ratio is 1.6.

### B.6.2 Fixed pressure ratio with different turbine inlet temperature

From refernce 35, the sea level net static thrust of Allison AE3007 turbofan engine is 7150 lbf and specific fuel consumption is 0.39 lb/lb-hr. From table B.6.1, we select the fan pressure 1.6, compressor pressure 14.375, together with different turbine inlet temperature, again to run engine simulation program. Following are the results listed in table B.6.2.

| TET °F | Fuel flow lb/sec | SFC lb/lb-hr | Sea level Static State Net Thrust, lbf |
|--------|------------------|--------------|----------------------------------------|
| 1420   | 0.8027           | 0.41         | 7042                                   |
| 1430   | 0.816            | 0.4132       | 7109                                   |
| 1440   | 0.829            | 0.416        | 7174                                   |
| 1450   | 0.8426           | 0.42         | 7237                                   |
| 1460   | 0.856            | 0.4223       | 7298                                   |

Table B.6.2 Sea level static thrust with different turbine TET

From table B.6.2, we find that when the TET increases, the sea level static thrust and specific fuel consumption will increase as well. After comparing the data listed in table

B.6.1 and B.6.2, the engine result of TET is 1440 ° F, pressure ratio at fan is 1.6, and compressor pressure ratio is 14.375, this result is very close to Allison AE3007 turbofan engine. Thus, we chose this data to run engine simulation program, from design point (sea level static state) , every 2,000 ft, to off design point (66,000 ft), then the simulated engine data map is obtained.

The whole engine map data is too large, in table B.6.3, we only presented the engine data from 42,000 to 66,000 ft.

| Height, ft | Mach No. | Fuel flow, lb/sec | SFC, lb/lb-hr | Thrust, lbf |
|------------|----------|-------------------|---------------|-------------|
| 42000      | 0.45     | 0.2524            | 0.547532      | 1659.52     |
| 42000      | 0.55     | 0.2602            | 0.599166      | 1563.374    |
| 42000      | 0.65     | 0.2694            | 0.669227      | 1449.195    |
| 46000      | 0.45     | 0.2082            | 0.540382      | 1387.019    |
| 46000      | 0.55     | 0.2147            | 0.590376      | 1309.199    |
| 46000      | 0.65     | 0.2224            | 0.6561        | 1220.302    |
| 50000      | 0.45     | 0.1718            | 0.533617      | 1159.033    |
| 50000      | 0.55     | 0.1772            | 0.582513      | 1095.117    |
| 50000      | 0.65     | 0.1834            | 0.645913      | 1022.181    |
| 54000      | 0.45     | 0.1417            | 0.526635      | 968.6398    |
| 54000      | 0.55     | 0.1462            | 0.573866      | 917.1486    |
| 54000      | 0.65     | 0.1513            | 0.636464      | 855.7903    |
| 58000      | 0.45     | 0.1169            | 0.519889      | 809.4808    |
| 58000      | 0.55     | 0.1206            | 0.565834      | 767.2915    |
| 58000      | 0.65     | 0.1249            | 0.626509      | 717.6909    |
| 62000      | 0.45     | 0.0965            | 0.51381       | 676.1254    |
| 62000      | 0.55     | 0.0995            | 0.558177      | 641.7322    |
| 62000      | 0.65     | 0.103             | 0.616787      | 601.1796    |
| 66000      | 0.45     | 0.0796            | 0.507531      | 564.6157    |
| 66000      | 0.55     | 0.0821            | 0.550878      | 536.5249    |
| 66000      | 0.65     | 0.085             | 0.605974      | 504.9722    |

Table B.6.3 Simulated engine data

## B.7 Climb performance

As in appendix A.11, after the engine data map is available, and the climb speed is given as well, then the time, fuel and distance to climb can be calculated.

In order to choose a proper climb velocity, two requirements, higher than aircraft stall

speed, and minimum aircraft drag speed. As in appendix A.11, the climb altitude, from sea level to 50,000 ft, is divided into three segments, and in each segment, a constant indicated climb speed has been given. Following is the calculation of the stall speed and minimum drag velocity.

**a. Stall speed**

From paragraph B.5.2, table B.5.2, the aircraft stall speed, at different weight and altitude, is given.

**b. Minimum drag speed**

As in appendix A.11, b. by using the pseudo program to calculate the minimum drag velocity (weight is 22,000 lbs, from 2,000 to 50,000 ft, every 4,000 ft).

After calculation, the results of minimum drag velocity, stall velocity and climb velocity are presented in table B.7.1.

| Altitude<br>, ft | $C_{D0}$ | $C_{Di}$ | Min. Drag Vel.<br>(true), fps | Stall velocity<br>(true), fps | Climb velocity<br>(true) ,fps |
|------------------|----------|----------|-------------------------------|-------------------------------|-------------------------------|
| 2000             | 0.0094   | 0.0095   | 226                           | 148.11                        | 348.18                        |
| 6000             | 0.0097   | 0.0099   | 238                           | 157.3                         | 369.69                        |
| 10000            | 0.0099   | 0.01     | 252                           | 167.3                         | 393.29                        |
| 14000            | 0.0104   | 0.0105   | 266                           | 178.3                         | 419.17                        |
| 18000            | 0.0106   | 0.0108   | 282                           | 190.5                         | 447.66                        |
| 22000            | 0.0108   | 0.011    | 300                           | 203.8                         | 383.12                        |
| 26000            | 0.0111   | 0.0111   | 321                           | 218.7                         | 410.92                        |
| 30000            | 0.0113   | 0.0114   | 343                           | 235.1                         | 441.64                        |
| 34000            | 0.0115   | 0.0116   | 368                           | 253.4                         | 476.02                        |
| 38000            | 0.0119   | 0.012    | 398                           | 276.2                         | 453.84                        |
| 42000            | 0.0123   | 0.0125   | 434                           | 304.1                         | 499.64                        |
| 46000            | 0.0127   | 0.0129   | 474                           | 334.8                         | 549.29                        |
| 50000            | 0.0131   | 0.0132   | 519                           | 368.6                         | 604.60                        |

Table B.7.1 Climb velocity VS stall & minimum drag velocity

By comparing these figures, in table B.7.1, the climb speed is higher than stall and minimum drag speed.

**B.7.1 Climb performance**

As in appendix A.11.1, eq. A.11.2 can be used to calculate the rate of climb.

By using eq. A.11.5, A.11.6, and A.11.7, the time to climb, the fuel to climb (FC), the

distance to climb, DC, can be calculated.

After computation, the rate of climb, time to climb, fuel to climb and distance to climb are presented at table B.7.2.

| Altitude, ft | Rate of climb, fps | Time to climb, sec | Fuel to climb, lb/sec | Distance to climb, nm |
|--------------|--------------------|--------------------|-----------------------|-----------------------|
| 2000.10      | 55.55              | 36.01              | 22.64                 | 2.00                  |
| 4000.20      | 60.03              | 33.32              | 26.98                 | 1.91                  |
| 6000.29      | 60.90              | 32.84              | 25.48                 | 1.94                  |
| 8000.39      | 61.82              | 32.35              | 24.05                 | 1.97                  |
| 10000.49     | 62.22              | 32.15              | 22.88                 | 2.02                  |
| 12000.59     | 62.59              | 31.96              | 21.75                 | 2.07                  |
| 14000.68     | 61.70              | 32.42              | 21.00                 | 2.16                  |
| 16000.78     | 60.61              | 33.00              | 20.32                 | 2.28                  |
| 18000.88     | 46.92              | 42.63              | 24.87                 | 3.04                  |
| 20000.98     | 47.43              | 42.17              | 23.51                 | 3.11                  |
| 22001.07     | 47.91              | 41.75              | 22.22                 | 3.18                  |
| 24001.17     | 45.31              | 44.14              | 21.85                 | 2.78                  |
| 26001.27     | 44.52              | 44.92              | 21.03                 | 2.93                  |
| 28001.37     | 39.01              | 51.27              | 22.90                 | 3.47                  |
| 30001.46     | 38.42              | 52.05              | 22.05                 | 3.65                  |
| 32001.56     | 37.79              | 52.93              | 21.20                 | 3.85                  |
| 34001.66     | 33.39              | 59.90              | 22.85                 | 4.52                  |
| 36001.76     | 32.39              | 61.76              | 22.30                 | 4.84                  |
| 38001.86     | 31.35              | 63.79              | 21.76                 | 5.20                  |
| 40001.95     | 27.13              | 73.73              | 22.51                 | 5.51                  |
| 42002.05     | 22.11              | 90.46              | 25.46                 | 7.10                  |
| 44002.15     | 19.53              | 102.38             | 26.40                 | 8.43                  |
| 46002.24     | 16.50              | 121.21             | 28.60                 | 10.47                 |
| 48002.34     | 13.56              | 147.49             | 31.88                 | 13.36                 |
| 50002.44     | 10.74              | 186.23             | 36.91                 | 17.70                 |

Table B.7.2 Climb performance

From table B.7.2, the rate of climb will decrease, the time need to climb and the fuel consumption will increase as the altitude increases.

Finally, climb from sea level to 50,000 ft, the climb time is about 26 minutes, consumes around 603 lbs fuel and the climb distance is 119.46 nautical miles.

## B.8 Cruise performance

As in appendix A.12, there are three approximate methods for estimating of cruise range and endurance.

According to the mission listed in fig 3.2.1, the cruise segment of Tier-II Plus is divided into three segments

**a. Cruise-climb ingress, from 50,000 ft to 65,000 ft.**

**b. Loiter at 65,000 ft for 24 hours.**

**c. Egress, descend from 65,000 ft to 50,000 ft.**

The cruise performance will be explored in the following sections.

### B.8.1 Cruise-climb segment

As in appendix A.12.1, during cruise-climb, the lift drag ratio and velocity is assumed to be constant. When the aircraft weight reduces, the lift coefficient and aircraft velocity are still the same, thus the aircraft altitude will automatically rise.

After cruise-climb 9 hours, Tier-II Plus reaches to 53,728 ft, there is a big difference from the requirement to reach 65,000 ft.

Obviously, the cruise-climb method can not meet Tier-II Plus requirement. Therefore, an alternative method of using variable climb speed, from 0.54 to 0.6 M, and constant rate of climb has been taken, to reach the altitude of 65,000 ft.

Since the requirement is in 9 hours, the aircraft should ascend 15,000 ft, that means every hour the aircraft should climb 1,666.7 ft. In another word, the rate of climb should be 0.463 fps. First we select the rate of climb is 0.46 fps, after 9 hours the aircraft should reach 65,000 ft. Following pseudo program is used to calculate constant rate of climb, approximately 0.46 fps.

```
Vcl = 580.84
```

```
do I = 1 , 9
```

```
    call drag subroutine to compute drag coefficient
```

```
    call thrust subroutine to obtain net thrust
```

```
do J = 1 , 500
```

$$RC = \frac{V_{cl}(T - J - D)}{W}$$

```
    if ( RC.LT.0.46 ) then
```

```
        out of do loop
```

```

else
    continue
end if
end do
 $T_{net} = T - J$  ! the thrust need to maintain aircraft rate of climb
0.46 fps being obtained.
 $sfc_{od} = sfc \left( 1 + 0.1 \left( \frac{T_{net}}{T} - 1 \right) \right)$  !since aircraft is not flying at max. net
thrust, so, sfcod is the off-design value of sfc.
 $f_{uel} = sfc_{od} D$ 
 $W = W - f_{uel}$ 
 $h = RC * 3600 + h$ 
 $\rho = \rho_{sl} * 0.297076 \exp \left( \frac{-(h - 11000)}{6341.6184} \right)$ 
 $C_l = \frac{2W}{\rho V^2 S}$ 
end do

```

After computation the results are presented at table B.8.1.

| Hours | Weight, lb | Altitude, ft | Lift drag ratio | Rate of climb, fps | Distance. NM |
|-------|------------|--------------|-----------------|--------------------|--------------|
| 1     | 22374.36   | 51573.68     | 31.92           | 0.44               | 330.67       |
| 2     | 21967.49   | 53163.44     | 31.69           | 0.44               | 664.78       |
| 3     | 21562.72   | 54785.91     | 31.42           | 0.45               | 1002.33      |
| 4     | 21159.20   | 56414.03     | 31.12           | 0.45               | 1343.32      |
| 5     | 20756.85   | 58007.39     | 30.78           | 0.44               | 1687.75      |
| 6     | 20354.79   | 59662.54     | 30.41           | 0.46               | 2035.62      |
| 7     | 19959.20   | 61292.55     | 30.15           | 0.45               | 2386.93      |

Table B.8.1 Constant rate of climb, 0.46 fps

From table B.8.1, after flying 7 hours, the altitude reaches to 61,292 ft, but at this time the engine net thrust is less than thrust required, the aircraft even can't maintain level flight at this altitude.

Therefore, we tried to reduce the rate of climb to 0.37 fps. By using the same program to calculate, after climb 9 hours, aircraft reaches to 61,539 ft.



The result is listed in table B.8.2.

| Hours | Weight, lb | Altitude, ft | Lift drag ratio | Rate of climb, fps | Distance, NM |
|-------|------------|--------------|-----------------|--------------------|--------------|
| 1     | 22376.16   | 51323.14     | 31.92           | 0.37               | 330.67       |
| 2     | 21971.99   | 52570.25     | 31.73           | 0.35               | 664.78       |
| 3     | 21571.19   | 53862.93     | 31.56           | 0.36               | 1002.33      |
| 4     | 21172.92   | 55186.53     | 31.36           | 0.37               | 1343.32      |
| 5     | 20777.4    | 56479.13     | 31.14           | 0.36               | 1687.75      |
| 6     | 20384.67   | 57754.85     | 30.92           | 0.35               | 2035.62      |
| 7     | 19994.73   | 58997.41     | 30.7            | 0.35               | 2386.93      |
| 8     | 19612.37   | 60237.95     | 30.62           | 0.34               | 2741.68      |
| 9     | 19236.63   | 61539.11     | 30.41           | 0.36               | 3099.86      |

Table B.8.2 Constant rate of climb, 0.37 fps

## B.8.2 Cruise segment

Due to deficiency of engine thrust, by the end of climb, the altitude is not reach to 65,000 ft. According to reference 38, in cruise segment the aircraft should loiter at constant altitude, for 24 hours. Therefore, first the aircraft needs cruise-climb from 61,539 ft to 65,000 ft, with the speed 0.6 M, then stay with constant speed and constant altitude loiter at 65,000 ft.

### B.8.2.1 Cruise-climb method

As in appendix A.12.1, by using the same method, the weight is 19,236.63 lbs, starts from 61,539 ft, cruise-climb to 65,000 ft.

After calculation, cruise-climb to 65,000 ft, the aircraft needs 9 hours, the result is listed in table B.8.3.

| Hours | Weight, lbs | Altitude, ft |
|-------|-------------|--------------|
| 0     | 19236.63    | 61539.11     |
| 1     | 18873.14    | 61936.01     |
| 2     | 18517.12    | 62332.24     |
| 3     | 18168.26    | 62727.95     |
| 4     | 17826.39    | 63123.18     |
| 5     | 17491.37    | 63517.93     |
| 6     | 17163.04    | 63912.18     |

|   |          |          |
|---|----------|----------|
| 7 | 16841.27 | 64305.94 |
| 8 | 16526.01 | 64699.11 |
| 9 | 16217.14 | 65091.65 |

Table B.8.3 Cruise-climb to 65000 ft

### B.8.2.2 Constant speed and constant altitude loiter

From reference 38, the loiter is 24 hours, when subtract by 9 hours from cruise-climb, there is 15 hours left for loiter.

During loiter, the altitude is 65,000 ft, and cruise speed, 0.6 M,

do  $i = 1, 15$

$$C_l = \frac{2W}{\rho V^2 S}$$

call drag subroutine to calculate drag coefficient ( $C_d$ )

$$L/D = \frac{C_l}{C_d}$$

$$T_{req} = D = \frac{W}{L/D}$$

call subroutine to calculate net thrust ( $T_{net}$ )

call subroutine to calculate specific fuel consumption

if ( $T_{net} > T_{req}$ ) then

continue

else

end of calculation

end if

$$sfc_{od} = sfc \left( 1 + 0.1 \left( \frac{T_{req}}{T_{net}} - 1 \right) \right)$$

$$f_{uel} = sfc_{od} D$$

$$W = W - f_{uel}$$

end do

After calculation, the aircraft weight is 12094.48 lbs. The result are in table B.8.4.

| Hours | Weight, lbs | Altitude, ft | Hours | Weight, lbs | Altitude, ft |
|-------|-------------|--------------|-------|-------------|--------------|
| 1     | 15912.62    | 65000        | 9     | 13706.38    | 65000        |
| 2     | 15615.21    | 65000        | 10    | 13455.22    | 65000        |
| 3     | 15324.57    | 65000        | 11    | 13208.75    | 65000        |
| 4     | 15040.39    | 65000        | 12    | 12966.78    | 65000        |
| 5     | 14762.35    | 65000        | 13    | 12729.13    | 65000        |
| 6     | 14490.19    | 65000        | 14    | 12495.6     | 65000        |
| 7     | 14223.64    | 65000        | 15    | 12266.04    | 65000        |
| 8     | 13962.45    | 65000        |       |             |              |

Table B.8.4 Constant speed cruise

### B.8.3 Egress

According to reference 38, the following segment is egress, 9 hours.

As we did at ingress segment, a variable speed, from 0.6 to 0.5 M, and constant rate of descending, 0.45 fps, is being used to descend. Following pseudo program is illustrated how to do calculation.

$$C_l = \frac{2W}{\rho V^2 S}$$

do I = 1, 9

call drag subroutine to calculate drag coefficient ( $C_d$ )

$$L/D = \frac{C_l}{C_d}$$

$$D = \frac{W}{L/D}$$

call subroutine to calculate net thrust ( $T_{net}$ )

call subroutine to calculate specific fuel consumption

if ( $T_{net} > D$ ) then ! compare  $T_{net}$  and  $D$

continue

else

end of calculation

end if

do j = 50, 600

$$RC = \frac{V(T - J - D)}{W}$$

if ( $RC < -0.45$ ) then ! in 9 hours descend 15000 ft, so

$T = T - j$  the rate of descend is -0.46 fps.

out of do loop

```

else
  continue
end if
end do
 $T_{req} = T - j$ 
 $sfc_{od} = sfc \left( 1 + 0.1 \left( \frac{T_{req}}{T} - 1 \right) \right)$ 
 $f_{uel} = sfc_{od} D$ 
 $W = W - f_{uel}$ 
 $h = h - RC * 3600$ 
 $\rho = \rho_{sl} 0.297076 \exp \left( \frac{-(h - 11000)}{6341.6184} \right)$ 
 $C_i = \frac{2W}{\rho V^2 S}$ 
end do

```

After computation, the result is presented at table B.8.5.

| Hours | Weight, lbs | Altitude, ft |
|-------|-------------|--------------|
| 0     | 12266       | 65000        |
| 1     | 12046.15    | 63348.88     |
| 2     | 11830.14    | 61688.59     |
| 3     | 11617.42    | 59928.89     |
| 4     | 11405.74    | 58298.76     |
| 5     | 11196.7     | 56553.13     |
| 6     | 10987.31    | 54862.08     |
| 7     | 10779.45    | 53211.08     |
| 8     | 10573.3     | 51445.95     |
| 9     | 10367.02    | 49682.13     |

Table B.8.5 Variable speed, constant rate of descending egress

## B.9 Take-off performance

As in appendix A.14.

### B.9.1 Take-off field length

As in appendix A.14.1, eq. A.14.1 can be used to calculate the take-off field length.

By using eq. A.14.2, the  $T$  is 5978.333 lbf ( $\lambda$  is engine by-pass ratio, 5, and  $T_{TO}$  is sea level static net thrust, 7174 lbf).

The take-off weight  $W_{TO}$  is 23,388.2 lb. If take-off velocity is 1.2 multiplied by the stall velocity (from paragraph B.5.3, the airplane stall velocity is 148.3 fps, the take off speed is 177.9 fps.

As in appendix A.9.2, by using eq. A.14.3, A.14.4, the value of  $\mu'$  and  $\gamma_{LOF}$  are 0.0312 and 0.171 respectively.

The wing loading is 44.64 and  $\left(\frac{V_3}{V_{STO}}\right)$  is 1.3, and according to reference 7, table 5.1,  $f_{TO}$  is 1.15, obstacle height  $h_{TO}$  is 35 ft.

Substituting these variables into eq. A.14.1, after computation, the take-off field length is 3,374.9 ft.

### **B.9.2 Take-off ground distance**

As in appendix A.14.2, eq. A.14.5 can be used to calculate the take-off ground distance, after calculation, take-off ground distance is 2,192.7 ft.

## **B.10 Landing performance**

As stated in appendix A.15.

### **B.10.1 The distance of $S_{air}$**

As in appendix A.15.1, eq. A.15.1 can be used to calculate the distance of  $S_{air}$ .

As in appendix A.15.1, an average value of  $\bar{\gamma}$  is 0.1.

For landing, assuming the aircraft weight is 11,000 lbs, the stall speed is 101.7 fps

Assuming the approach velocity is 1.3 multiplied by the stall speed, after calculation,  $V_{app}$  is 132.2 fps.

By using eq. A.15.3, touch down speed is 125.4 fps.

Substituting these variables into eq. A.15.1, then the distance of  $S_{air}$  is 771.6 ft.

### **B.10.2 Landing ground run $S_{LG}$**

As in appendix A.15.2, eq. A.15.4 can be used to calculate landing ground run  $S_{LG}$ .

The variables in eq. A.15.4, is the same as appendix A.15.2, substituting these variables into eq. A.15.4, the distance of  $S_{LG}$  is 444.46 ft.

Thus, the total landing distance  $S_{air}$  plus  $S_{LG}$  is 1,216.07 ft.

## B.11 Stability analysis

As in appendix A.18, the following paragraphs were taken to estimate aircraft stability.

### B.11.1 Trim in cruise

As stated in appendix A.18.1, eq. A.18.1 can be used to calculate trim in cruise, and eq. A.18.2 can be used to calculate the value of angle of incidence of horizontal surface,  $\alpha_T$ .

Before calculation, there are 13 variables need to be given.

a. Distance of thrust line below centre of gravity,  $Z_T$ .

The thrust line of Tier-II Plus is above centre of gravity, is negative (-3.94 ft).

b. Mean aerodynamic chord,  $c$ , is 4.99 ft.

c. Wing area,  $S$ , is 523.8 ft<sup>2</sup>.

d. Wing body pitch moment coefficient at zero lift,  $C_{m0WB}$

As in appendix A.18.1, d. the wing body pitch moment coefficient at zero lift is;

d.1 Estimation of wing alone zero lift angle of attack,  $\alpha_{orW}$

As in eq. A.18.3, the wing alone zero lift angle of attack  $\alpha_{orW}$  is comprised of two parts,  $\alpha_{or1}$ , and  $\alpha_{or2}$ .

d.1.1 Calculate the value of  $\alpha_{or1}$

As in appendix A.18.1, d.1.1, the value of wing airfoil zero lift incidence is -2.328 degrees.

d.1.2 Calculate the value of  $\alpha_{or2}$

The taper ratio,  $\lambda$ , is 0.35, by using eq. A.18.6, the value of  $\delta$  is -3 degrees.

The aspect ratio is 25.78, and quarter wing chord sweep angle ( $\tan\Lambda_{1/4}$ ) is 5 degrees, thus the value of  $AR \tan \Lambda_{1/4}$  is 2.255. By using eq. A.18.7, the value of  $\alpha_{or2}$  is 1.1781.

Adding together, the value of  $\alpha_{orW}$  is -1.15 degrees.

d.2 Estimate of body effect on zero lift angle of attack,  $\Delta\alpha_{or}$

The width of fuselage is 4.72 ft and the wing span is 116.2 ft. By using eq. A.18.8, the value of  $K_2/K_1$  is 0.92.

The wing body incidence,  $i_w$  is 6 degrees, and the value of wing airfoil zero lift incidence,  $\alpha_{or1}$  is -2.328 degrees. By using eq. A.18.9, the value of body effect on zero lift angle of attack,  $\Delta\alpha_{or}$ , is 0.663 degrees.

### **d.3 Estimating of wing body zero lift angle of attack, $\alpha_{0rWB}$**

By using eq. A.18.10, the value of wing body zero lift angle of attack is -0.487 degrees.

### **d.4 Estimating of wing alone pitching moment coefficient at zero lift, $C_{m0W}$**

As in appendix A.18.1, d.4, the zero lift pitching moment coefficient can be divided into  $C_{m01}$ ,  $C_{m02}$ , two parts.

#### **d.4.1 Evaluation of $C_{m01}$ , planform effect**

As in appendix A.18.1, d.4.1, the value of incompressible pitching moment coefficient at zero total wing lift,  $C_{m0i}$  is -0.0685, the value of the empirical correction factor, F is 0.8585.

The wing is straight taper and camber line is unchanged across the span,  $\cos \Lambda_{\frac{1}{2}}$  is 5 degrees, by using eq. A.18.14,  $C_{m0i1}$  is -0.05745.

By using eq. A.18.15, the value of  $C_{m0i1}$  is -0.07147.

#### **d.4.2 Evaluation of $C_{m0i2}$ , linear twist**

As in appendix A.18.1, d.4.2, the value of K is 0.019. By using eq. A.18.17, the value of  $C_{m02}$  is -0.00697.

Substituting the values of  $C_{m01}$ ,  $C_{m02}$  into eq. A.18.11, the zero lift pitching moment coefficient is -0.07844.

### **d.5 Estimation of body effect pitching moment coefficient at zero lift, $\Delta C_{m0}$**

As in appendix A.18.1, d.5,  $w$  is maximum body width, 4.72 ft, and  $S_B$  is platform area of body, 192.5 ft<sup>2</sup>.  $S_{Bn}$  is planform area of body forward of lateral line through quarter chord point of MAC, 143.3 ft<sup>2</sup>,  $l_{Bn}$  is the distance from body nose to quarter chord point of MAC, 23.2 ft and  $l_B$  is the length of body, 44.4 ft.  $S_W$  is planform of gross area, 491.34 ft<sup>2</sup>,  $\psi$  is the angle between zero lift plane of wing and airflow direction for zero body alone pitching moment, 7.15 degrees.

By using eq. A.18.19, the value of  $C_{m0b}$  is -0.0386.

$Z$  is vertical displacement of quarter chord point of  $C_r$  above mid body position, -1.4 ft, and  $h$  is maximum height of body, 4.72 ft. By using eq. A.18.20, the contribution due to wing height is -0.002966.

By using eq. A.18.21, the contribution due to wing sweep is -0.01355.

Substituting these three variables into eq. A.18.18, the value of body effect pitching moment coefficient at zero lift,  $\Delta C_{m0}$  is -0.055.

By using eq. A.18.22, the value of wing body pitching moment coefficient at zero lift,  $C_{m0WB}$  is -0.1336.

**e. Location of horizontal surface lift after of centre of gravity,  $l_T$**

From reference 45,  $l_T$  is the distance from the location of aircraft most after C.G. to the location of tailplane A.C. For Tier-II Plus is 17.82 ft.

**f. Wing body incidence,  $\alpha_w$  is -6 degrees.**

**g. Position of centre of gravity of leading edge of MAC, H is 0.385.**

From above calculation, 7 variables of eq. A.18.2 has been solved, there are still 6 variables remaining unknown, and these variables varies with different cruise condition.

**A. At the beginning of cruise,  $C'$ .**

At the beginning of cruise, the aircraft altitude is around 62,500 ft , weight is around 18,980 lbs and velocity is 0.6 M (580.8 fps). The rest of variables in eq. A.18.2 are;

a. Thrust: Thrust required at this moment is 622.7 lbf.

b. Dynamic pressure is 33.24 lb/ft<sup>2</sup>.

c. Wing body lift curve slope,  $C_{L\alpha_w}$  is 6.66.

d. Tailplane lift curve slope,  $C_{L\alpha_T}$  is 4.589.

e. Down wash gradient at the horizontal tail,  $\frac{d\varepsilon}{d\alpha}$  is 0.208.

f. Wing body aerodynamic centre,  $H_0$ .

As in appendix A.18.1, f, wing body aerodynamic centre is 0.1999, as the same condition, to calculate the trim in cruise,  $\alpha_T$ .

Substituting these variables into eq. A.18.2, the following results are obtained.

| $H$   | $s_r/s$ | $\alpha_T$ |
|-------|---------|------------|
| 0.346 | 0.15    | -1.453     |
| 0.346 | 0.2     | -1.435     |
| 0.346 | 0.25    | -1.425     |
| 0.346 | 0.3     | -1.418     |
| 0.386 | 0.15    | -0.442     |
| 0.386 | 0.2     | -0.677     |
| 0.386 | 0.25    | -0.818     |
| 0.386 | 0.3     | -0.912     |
| 0.426 | 0.15    | 0.568      |
| 0.426 | 0.2     | 0.08       |



|       |      |        |
|-------|------|--------|
| 0.426 | 0.25 | -0.212 |
| 0.426 | 0.3  | -0.407 |

Table B.11.1 Trim in beginning cruise,  $\alpha_T$ .

As in appendix A.18.1, since the range of aircraft C.G. movement is small, therefore we take an average value of aircraft C.G., the elevator angle,  $\alpha_T$  is -0.71 degrees.

**B. At the end of cruise,  $e$ .**

At the end of cruise, the aircraft altitude is 65,000 ft, weight is around 12180 lbs and velocity is 0.6 M (580.8 fps). The rest of variables in eq. A.18.2 are

- Thrust: Thrust required at this moment is 392.38 lbf.
- Dynamic pressure is 29.68 lb/ft<sup>2</sup>
- Wing body lift curve slope,  $C_{L_{\alpha wf}}$  is 6.623.
- Tailplane lift curve slope,  $C_{L_{\alpha T}}$  is 4.573.
- Down wash gradient at the horizontal tail,  $\frac{d\varepsilon}{d\alpha}$  is 0.2082.
- Wing body aerodynamic centre,  $H_0$  is 0.1996.

| $H$   | $s_x/s$ | $\alpha_T$ |
|-------|---------|------------|
| 0.346 | 0.15    | -0.278     |
| 0.346 | 0.2     | -0.05      |
| 0.346 | 0.25    | 0.087      |
| 0.346 | 0.3     | 0.178      |
| 0.386 | 0.15    | 0.455      |
| 0.386 | 0.2     | 0.5        |
| 0.386 | 0.25    | 0.526      |
| 0.386 | 0.3     | 0.544      |
| 0.426 | 0.15    | 1.188      |
| 0.426 | 0.2     | 1.049      |
| 0.426 | 0.25    | 0.966      |
| 0.426 | 0.3     | 0.911      |

Table B.11.2 Trim in final cruise,  $\alpha_T$ .

As with section A, we take an average value of  $\alpha_T$  at aircraft C.G. as 0.386, then trim in cruise,  $\alpha_T$  at the end of cruise is 0.51 degrees.

### B.11.2 Trim on approach to land

As in appendix A.18.2, eq. A.18.25 can be used to calculate trim on approach to landing, but the value of  $\alpha_T$  is obtained from the previous paragraph with  $\eta$  being the variable.

Assuming approach thrust is 360 lbf. The value of lift coefficient due to control surface deflection,  $C_{\pi}$  can be calculated by using eq. A.18.26.

First the proportion of control surface chord to wing chord,  $c/c$ , is assumed to be 0.3. the Prandtl-Glauert compressibility factor,  $\beta$  is 0.8 and tailplane projected aspect ratio is 4.736. Substituting these variables into eq. A.18.26, the value of lift coefficient due to control surface deflection,  $\frac{C_{\pi}}{C_{L\alpha T}}$  is 0.6946.

The value of elevator deflection,  $\eta$  is calculated as follows :

#### B.11.2.1 $\alpha_T$ is -0.71 and tail lift curve slope, $C_{L\alpha T}$ is 5.9285.

As in appendix A.18.2.1, the value of tailplane sweep angle,  $\Lambda$  is 7 degrees. By using eq. A.18.27, the value of  $K_1, K_2$  is 0.2163 and 1 respectively.

By using eq. A.18.28, the value of  $C_{\pi}$  is 3.227.

| $H$   | $s_r/s$ | $\eta$ |
|-------|---------|--------|
| 0.346 | 0.15    | 0.7    |
| 0.346 | 0.2     | 1.134  |
| 0.346 | 0.25    | 1.395  |
| 0.346 | 0.3     | 1.568  |
| 0.386 | 0.15    | 1.673  |
| 0.386 | 0.2     | 1.864  |
| 0.386 | 0.25    | 1.978  |
| 0.386 | 0.3     | 2.055  |
| 0.426 | 0.15    | 2.645  |
| 0.426 | 0.2     | 2.593  |
| 0.426 | 0.25    | 2.562  |
| 0.426 | 0.3     | 2.541  |

Table B.11.3 Trim on approach,  $\alpha_T$  is -0.71 degrees.

The results obtained in table B.11.3 are substituting these variables into eq. A.18.25, and assuming the aircraft C.G. to be within the range  $\pm 0.04$ ,  $s_r/s$  as 0.15, 0.2, 0.25 and 0.3. Finally, the value of elevator deflection,  $\eta$ , trim on approach to land is obtained.

#### B.11.2.2 $\alpha_T$ is 0.51 and tail lift curve slope, $C_{l_{\alpha T}}$ is 5.8923

If the value of tail lift curve slope,  $C_{l_{\alpha T}}$  is 5.8923, by using eq. A.18.28, the value of lift coefficient due to control surface deflection,  $C_{l_T}$  is 3.207.

Assume  $\alpha_T$  to be 0.51, and assuming the aircraft C.G. to be within the range  $\pm 0.04$ ,  $s_r/s$  as 0.15, 0.2, 0.25 and 0.3, the value of elevator deflection,  $\eta$ , is;

| $H$   | $s_r/s$ | $\eta$ |
|-------|---------|--------|
| 0.346 | 0.15    | -1.075 |
| 0.346 | 0.2     | -0.64  |
| 0.346 | 0.25    | -0.379 |
| 0.346 | 0.3     | -0.205 |
| 0.386 | 0.15    | -0.096 |
| 0.386 | 0.2     | 0.094  |
| 0.386 | 0.25    | 0.208  |
| 0.386 | 0.3     | 0.285  |
| 0.426 | 0.15    | 0.882  |
| 0.426 | 0.2     | 0.828  |
| 0.426 | 0.25    | 0.796  |
| 0.426 | 0.3     | 0.774  |

Table B.11.4 Trim on approach,  $\alpha_T$  is 0.51 degrees.

### B.11.3 Cruise static stability

As in appendix A.18.3, eq. A.18.29 can be used for calculating cruise static stability, at a stick fixed static margin,  $K_n$  is 0.05 and 0.1, and assume the value of  $G$  is 1.

#### B.11.3.1 Initial cruise

After computation, the value of  $s_r/s$  is

| $H$    | $K_n$ | $s_r/s$ |
|--------|-------|---------|
| 0.3461 | 0.05  | 0.1007  |
| 0.3861 | 0.05  | 0.1212  |
| 0.4261 | 0.05  | 0.1418  |
| 0.3461 | 0.1   | 0.1264  |
| 0.3861 | 0.1   | 0.1469  |
| 0.4261 | 0.1   | 0.1674  |

Table B.11.5 Initial cruise static stability

### B.11.3.2 Final cruise

After computation, the value of  $s_r/s$  is

| $H$    | $K_n$ | $s_r/s$ |
|--------|-------|---------|
| 0.3461 | 0.05  | 0.1006  |
| 0.3861 | 0.05  | 0.1211  |
| 0.4261 | 0.05  | 0.1416  |
| 0.3461 | 0.1   | 0.1262  |
| 0.3861 | 0.1   | 0.1467  |
| 0.4261 | 0.1   | 0.1672  |

Table B.11.6 Final cruise static stability

### B.11.4 Rotation at take off

As in appendix A.18.4, assuming the elevator up deflection,  $\eta$  at take off to be -25 degrees, by using eq. A.18.30, the value of  $s_r/s$  at take off rotation can be calculated.

In eq. A.18.30, the take off thrust is 7170 lbf, the distance from thrust line above ground,  $h_T$  is 9.6 ft, the location of leading edge of MAC of the most after wheel position,  $H_G$  is 0.737, the wing body lift curve slope,  $C_{L_{\alpha wf}}$  is 5.7425, the take off

weight is 23195.9 lbs, the down wash gradient at the horizontal tail,  $\frac{d\epsilon}{d\alpha}$  is 0.1726, the

distance from wing leading edge to tailplane aerodynamic centre,  $l_T^1$  is 19.75 ft, the wing body aerodynamic centre,  $H_0$ , is 0.1314, the pitch radius of gyration,  $K_B$  is 6.66, and the position of centre of gravity after of leading edge of MAC,  $H$  is 0.386.

The only variable varied is the angle of incidence of horizontal surface,  $\alpha_T$ . From section B.11.1, we obtained two values of  $\alpha_T$ .

For Tier-II Plus, the value of  $s_r/s$  is 0.153, location of horizontal surface lift aft. of centre of gravity,  $l_r$ , is 17.82 ft. By using eq. A.18.31, the value of tail volume coefficient is 0.548.

#### B.11.4.1 $\alpha_T$ is -0.71

The proportion of control surface chord to wing chord,  $c_j/c$ , is 0.3 and 0.4 respectively.

| T.O. velocity, fps              | $s_r/s$ | $\bar{V}$ | Tier-II Plus, $\bar{V}$ |
|---------------------------------|---------|-----------|-------------------------|
| 177, (1.2V <sub>stall</sub> )   | 0.1967  | 0.703     | 0.548                   |
| 192, (1.3V <sub>stall</sub> )   | 0.1627  | 0.581     |                         |
| 206.7, (1.4V <sub>stall</sub> ) | 0.1349  | 0.482     |                         |

(a)  $c_j/c$  as 0.3

| take off velocity, fps           | $s_r/s$ | $\bar{V}$ |
|----------------------------------|---------|-----------|
| 177, (1.2V <sub>stall</sub> )    | 0.1617  | 0.578     |
| 184.6, (1.25V <sub>stall</sub> ) | 0.1468  | 0.524     |
| 192, (1.3V <sub>stall</sub> )    | 0.1334  | 0.476     |

(b)  $c_j/c$  as 0.4

Table B.11.7 Tail volume coefficient for rotation as  $\alpha_T$  is -0.71

#### B.11.4.2 $\alpha_T$ is 0.51

The proportion of control surface chord to wing chord,  $c_j/c$ , is 0.3 and 0.4 respectively.

| T.O. velocity, fps              | $s_r/s$ | $\bar{V}$ | Tier-II Plus, $\bar{V}$ |
|---------------------------------|---------|-----------|-------------------------|
| 177, (1.2V <sub>stall</sub> )   | 0.21    | 0.749     | 0.548                   |
| 192, (1.3V <sub>stall</sub> )   | 0.1736  | 0.62      |                         |
| 206.7, (1.4V <sub>stall</sub> ) | 0.144   | 0.5147    |                         |

(a)  $c_j/c$  as 0.3

| take off velocity, fps           | $s_T/s$ | $\bar{V}$ |
|----------------------------------|---------|-----------|
| 177, (1.2V <sub>stall</sub> )    | 0.171   | 0.61      |
| 184.6, (1.25V <sub>stall</sub> ) | 0.155   | 0.553     |
| 192, (1.3V <sub>stall</sub> )    | 0.141   | 0.5       |

(b)  $c/c$  as 0.4

Table B.11.8 Tail volume coefficient for rotation as  $\alpha_T$  is 0.51

### B.11.5 Dynamic stability

As in appendix A.18.5, the following paragraphs are estimating dynamic stability.

#### B.11.5.1 Short period oscillation

By using eq. A.18.32, A.18.33, the short period damping ratio,  $\zeta_s$  and short period frequency,  $\omega_s$  can be computed.

As in the previous section, during cruise, we take two extreme conditions, initial and final cruise to facilitate calculations.

##### a. Initial cruise, $\alpha_T$ is -0.71

After calculation, by using eq. A.18.32, and A.18.33, the short period damping ratio,  $\zeta_s$  and short period frequency,  $\omega_s$  is

| $H$   | $s_T/s$ | $\zeta_s$ |
|-------|---------|-----------|
| 0.346 | 0.15    | 0.198     |
| 0.346 | 0.2     | 0.174     |
| 0.346 | 0.25    | 0.166     |
| 0.346 | 0.3     | 0.163     |
| 0.386 | 0.15    | 0.242     |
| 0.386 | 0.2     | 0.193     |
| 0.386 | 0.25    | 0.178     |
| 0.386 | 0.3     | 0.172     |
| 0.426 | 0.15    | 0.341     |
| 0.426 | 0.2     | 0.221     |
| 0.426 | 0.25    | 0.194     |
| 0.426 | 0.3     | 0.183     |

(a) Damping ratio

| $H$   | $\omega_s$ | $\zeta_s$ |
|-------|------------|-----------|
| 0.346 | 0.263      | 0.198     |
| 0.346 | 0.346      | 0.174     |
| 0.346 | 0.413      | 0.166     |
| 0.346 | 0.471      | 0.163     |
| 0.386 | 0.215      | 0.242     |
| 0.386 | 0.311      | 0.193     |
| 0.386 | 0.384      | 0.178     |
| 0.386 | 0.446      | 0.172     |
| 0.426 | 0.153      | 0.341     |
| 0.426 | 0.272      | 0.221     |
| 0.426 | 0.354      | 0.194     |
| 0.426 | 0.419      | 0.183     |

(b) Frequency

Table B.11.9 Short period oscillation,  $\alpha_T$  is -0.71

**b. Final cruise,  $\alpha_T$  is 0.51**

| $H$   | $s_T/s$ | $\zeta_s$ |
|-------|---------|-----------|
| 0.346 | 0.15    | 0.232     |
| 0.346 | 0.2     | 0.204     |
| 0.346 | 0.25    | 0.195     |
| 0.346 | 0.3     | 0.192     |
| 0.386 | 0.15    | 0.283     |
| 0.386 | 0.2     | 0.227     |
| 0.386 | 0.25    | 0.209     |
| 0.386 | 0.3     | 0.202     |
| 0.426 | 0.15    | 0.394     |
| 0.426 | 0.2     | 0.259     |
| 0.426 | 0.25    | 0.227     |
| 0.426 | 0.3     | 0.215     |

(a) Damping ratio

| $H$   | $\omega_s$ | $\zeta_s$ |
|-------|------------|-----------|
| 0.346 | 0.313      | 0.232     |
| 0.346 | 0.411      | 0.204     |
| 0.346 | 0.491      | 0.195     |
| 0.346 | 0.559      | 0.192     |
| 0.386 | 0.257      | 0.283     |
| 0.386 | 0.371      | 0.227     |
| 0.386 | 0.457      | 0.209     |
| 0.386 | 0.529      | 0.202     |
| 0.426 | 0.184      | 0.394     |
| 0.426 | 0.325      | 0.259     |
| 0.426 | 0.421      | 0.227     |
| 0.426 | 0.498      | 0.215     |

(b) Frequency

Table B.11.10 Short period oscillation at ,  $\alpha_T$  is 0.51

### B.11.6 Phugoid in cruise

As in appendix A.18.6, eq. A.18.35, A.18.36, A.18.37, A.18.38 and A.18.39 can be used to calculate the value of ,  $F_1, F_2, \omega, \nu$  and the static fixed margin,  $k_n$ .

The damping ratio in phugoid motion can be calculated by using eq. A.18.34.

As in appendix A.18.6, two extreme points, during cruise, were taken to calculate the phugoid motion.

#### B.11.6.1 Initial cruise,

The value of wing body lift curve slope,  $C_{L_{\alpha wf}}$  is 6.66, tailplane lift curve slope,  $C_{L_{\alpha t}}$  is 4.589, aircraft mass is 587 lb-sec<sup>2</sup>/ft, density,  $\rho$  is 0.0002 slugs/ft<sup>3</sup>, down wash gradient at the horizontal tail,  $\frac{d\varepsilon}{d\alpha}$  is 0.208, wing body aerodynamic centre,  $H_0$  is 0.1999, the drag coefficient,  $C_D$  is 0.0346, the zero drag coefficient,  $C_{D_0}$  is 0.0142, and lift coefficient,  $C_L$  is 1.069.

If we take two locations of aircraft C.G to calculate short period frequency phugoid, one is -0.04 from aircraft C.G., the other is at aircraft C.G.

a. -0.04 from aircraft C.G. (  $H$  is 0.346 )



From eq. A.18.39, the static fixed margin,  $k_n$ , is 0.1528. Then from eq. A.18.37 and A.18.38, we can obtain the value of  $\omega$  and  $\nu$  is 320.7 and 2.521 respectively.

Substituting those values into eq. A.18.35 and A.18.36, then the value of  $F_1$  and  $F_2$  is 0.7436 and 0.0104.

Finally, from eq. A.18.34, the value of damping ratio in phugoid motion,  $\zeta_L$  is 0.022.

**b. At aircraft C.G. (H is 0.386)**

As in the previous section, by using different equations, the static fixed margin,  $k_n$ , is 0.1128, the values of  $\omega$  and  $\nu$  are 236.7 and 2.521, and the value of  $F_1$  and  $F_2$  is 0.7414 and 0.0094 respectively.

Finally, the value of damping ratio in phugoid motion,  $\zeta_L$  is 0.021.

**B.11.6.2 Final cruise**

The value of wing body lift curve slope,  $C_{L_{awf}}$  is 6.623, tailplane lift curve slope,  $C_{L_{at}}$  is 4.573, aircraft mass is 376.6 lb-sec<sup>2</sup>/ft, density,  $\rho$  is 0.00018 slugs/ft<sup>3</sup>, downwash gradient at the horizontal tail,  $\frac{d\varepsilon}{d\alpha}$  is 0.208, wing body aerodynamic centre,  $H_0$  is 0.1996, the drag coefficient,  $C_D$  is 0.0252, the zero drag coefficient,  $C_{D_0}$  is 0.0032, and lift coefficient,  $C_L$  is 0.7623.

As with section C.12.6.1, two position of aircraft C.G. are taken into consideration.

**a. -0.04 from aircraft C.G. (H is 0.346)**

As in section B.11.6.1 a., by using different equations, the static fixed margin,  $k_n$ , is 0.153, the values of  $\omega$  and  $\nu$  are 227.9 and 2.513, and the value of  $F_1$  and  $F_2$  is 0.741 and 0.02346 respectively.

Finally, the value of damping ratio in phugoid motion,  $\zeta_L$  is 0.016.

**b. At aircraft C.G. (H is 0.386)**

As in section B.11.6.1 a., by using different equations, the static fixed margin,  $k_n$ , is 0.1128, the values of  $\omega$  and  $\nu$  are 168.4 and 2.513, and the value of  $F_1$  and  $F_2$  is 0.7382 and 0.0219 respectively.

Finally, the value of damping ratio in phugoid motion,  $\zeta_L$  is 0.0151.

## Appendix C

### Cost analysis

As stated in chap 4.1, instead of estimating the whole life cycle cost, we only dealing with the most important parts of life cycle cost, operation cost and acquisition cost.

#### C.1 Operation cost

According to reference 52, chap 6, military airplane operation cost is based on some assumptions, airplane quantity and service year. In this example, regarding reference 26, there are eight Tier-II Plus to be employed by United States, and we assumed the service of Tier-II Plus in the military to be twenty-five years.

Reference 52, eq. 6.1, can be used to acquire the operation cost of military airplanes,

$$C_{OPS} = C_{POL} + C_{PDIR} + C_{PIND} + C_{CMAT} + C_{SP} + C_{DEP} + C_{MIS} \quad (C.1.1)$$

In eq. C.1.1,  $C_{POL}$  is the program fuel, oil and lubricants cost,  $C_{PDIR}$  is the program cost of direct personnel,  $C_{PIND}$  is the program cost of indirect personnel,  $C_{CMAT}$  is the program cost of consumable materials used in conjunction with maintenance,  $C_{SP}$  is the program cost of spares,  $C_{DEP}$  is the program cost associated with depots, and  $C_{MIS}$  is the program miscellaneous cost.

##### C.1.1 The program fuel, oil and lubricants cost, $C_{POL}$

Reference 52, eq. 6.2 can be used to give estimation of program cost of fuel, oil, and

lubricants.

$$C_{POL} = F_{OL} W_{Fused} \frac{FP}{FD} N_{mis} N_{serv} N_{yr} \quad (C.1.2)$$

In eq. C.1.2,  $F_{OL}$  is a factor which accounts for the cost of oil and lubricants. From reference 52, at this stage, the value of 1.005 is recommended.  $W_{Fused}$  is the mission fuel used in pounds per mission. For Tier-II plus, the mission fuel is 13,300 lbs. FP is the price of fuel in USD per gallon. In this example we assume JP-4 as the fuel to be used, and from reference 52, fig 5.3, in 1996 the price is 0.9 USD/gallon. FD is the fuel density in pounds per gallon. From reference 52, p: 88, it is 6.55 lbs/gallon.  $N_{mis}$  is the number of mission flown per year, and can be calculated by:

$$N_{mis} = \frac{U_{anu}}{T_{mis}} \quad (C.1.3)$$

In eq. C.1.3,  $U_{anu}$  is the annual utilisation in flight hours. According to reference 26, the Pentagon will purchase eight Tier-II Plus. If two out of eight is under repair or overhaul, then six of them should stay in standby condition. Assume each sortie can last around two days, then each aircraft should fly in twelve days interval. Therefore, in one year, each aircraft should carry out thirty sorties, and in each sortie, the mission time,  $T_{mis}$ , takes forty-two hours. Thus, in one year each aircraft should fly 1260 hours. Then from eq. C.1.3,  $N_{mis}$  is 30.

$N_{serv}$  is the number of airplanes of the type in actual service. From reference 52, eq. 6.4, this number can be calculated as;

$$N_{serv} = N_{acq} - N_{res} - 0.5N_{loss} \quad (C.1.4)$$

In eq. C.1.4,  $N_{acq}$  is the number of airplanes which are acquired, the number is 8.  $N_{res}$  is the number of airplanes held in active reserve. From reference 52, 10 percent of acquired airplanes is held in active reserve, then  $N_{res}$  is 0.8.  $N_{loss}$  is the number of airplanes lost through accidents over the service life,  $N_{yr}$ .

$$N_{loss} = L_R N_{serv} U_{anu} N_{yr} \quad (C.1.5)$$

Assuming the loss rate is two times per 100,000 flight hours, and  $N_{yr}$  is 25 years. Thus,

$$N_{loss} = \frac{2 \cdot 1260 \cdot 25}{100000} N_{serv} = 0.63 N_{serv}$$

Substituting this value into eq. C.1.4, then the value of  $N_{serv}$  is 5.475 units.

Finally, substituting above variables into eq. C.1.2, then the value of the program cost of fuel, oil, and lubricants,  $C_{pol}$  is USD 7,541,614.

### C.1.2 The program cost of direct personnel, $C_{PDIR}$

From reference 52, the program cost of direct personnel can be split into two elements, the program cost for aircrews,  $C_{crewpr}$ , and the program cost for direct maintenance personnel,  $C_{mpdir}$ .

$$C_{PDIR} = C_{crewpr} + C_{mpdir} \quad (C.1.6)$$

The Tier-II Plus is unmanned aircraft, so the program cost for aircrews,  $C_{crewpr}$ , is zero. From reference 52, the program cost for direct maintenance personnel,  $C_{mpdir}$ , can be calculated by eq. 6.11.

$$C_{mpdir} = N_{serv} N_{yr} U_{anu} MHR_{flt} R_{mml} \quad (C.1.7)$$

In eq. C.1.7, there are five variables need to be solved. From previous section, three of them are given. Therefore, we only need to calculate two variables.

$MHR_{flt}$  is the number of maintenance manhours required per flight hour. Due to lacking real data, from reference 52, table 6.5, we categorise Tier-II Plus as an bomber aircraft, so the value of  $MHR_{flt}$  is between 25 to 50.

$R_{mml}$  is the military maintenance labour rate in USD/hr, from reference 52, p: 157, in 1989, is assumed to be 45 USD/hr. By using reference 52, eq. 6.12, then year  $R_{mml}$  can be calculated.

$$R_{mml_{thenyr}} = \frac{CEF_{thenyr}}{CEF_{1989}} R_{mml_{1989}} \quad (C.1.8)$$

From reference 52, fig 2.7, the value of  $CEF_{1989}$  is 3.02, and  $CEF_{thenyr}$ , 1996, is 3.25. Thus, in 1996 the value of  $R_{mml}$  is 48.43 USD/hr.

Substituting these variables into eq. C.1.7, the value of the program cost for direct maintenance personnel,  $C_{mpdir}$ , is between USD 208,808,972 and USD 417,617,944.

Since the program cost for aircrews,  $C_{crewpr}$ , is zero, so the value of  $C_{PDIR}$  is equal to the value of  $C_{mpdir}$ .

### C.1.3 The program cost of indirect personnel, $C_{PIND}$

Reference 52 stated that this cost component covers the cost of all squadron level personnel which are not directly involved in flight operations or airplane maintenance. Its value can be calculated by using eq. 6.13.

$$C_{PIND} = f_{pind} C_{OPS} \quad (C.1.9)$$

$f_{pind}$  is the operating cost fraction which are not directly involved in flight operation or airplane maintenance. From reference 52, table 6.6, we chose the figure of electronic

warfare airplane F-111, 0.14 as its value. Therefore, the program cost of indirect personnel,  $C_{PIND}$ , is  $0.14C_{OPS}$ .

#### C.1.4 The program cost of consumable materials, $C_{CMAT}$

From reference 52, chap 6.4, this cost component covers the cost of consumable materials used in conjunction with military airplane maintenance functions. Its value can be calculated by using eq. 6.14.

$$C_{CMAT} = N_{serv} N_{yr} U_{amu} MHR_{flt} R_{cmat} \quad (C.1.10)$$

In eq. C.1.10, there are five variables should be given. From previous calculation, four of them are obtained. Therefore, only one variable needs to be solved.

$R_{cmat}$  is the average cost in USD/hr (per maintenance hour) for consumable materials. From reference 52, eq. 6.15 is used to calculate its value.

$$R_{cmat} = 6.5 \frac{CEF_{thenyr}}{CEF_{1989}} \quad (C.1.11)$$

From eq. C.1.8, the value of  $CEF_{thenyr}$  and  $CEF_{1989}$  were given, so  $R_{cmat}$  is 6.995.

Thus, substituting these variables into eq. C.1.10, the value of the program cost of consumable materials,  $C_{CMAT}$ , is from USD 30,159,522 to USD 60,319,045.

#### C.1.5 The program cost of spares, $C_{SP}$

From reference 52, the program cost of spares may be estimated by using eq. 6.16.

$$C_{SP} = f_{sp} C_{OPS} \quad (C.1.12)$$

$f_{sp}$  is the operating cost fraction about program spares. From reference 52, table 6.6, as in C.1.4, we take F-111 as reference, but with technology improvement, we take 0.25 as its value. Thus, the program cost of spares,  $C_{SP}$ , is  $0.25C_{OPS}$ .

#### C.1.6 The program cost associated with depots, $C_{DEP}$

From reference 52, the program cost associated with depots may be estimated by using eq. 6.17.

$$C_{DEP} = f_{dep} C_{OPS} \quad (C.1.13)$$

$f_{dep}$  is the operating cost fraction about depot overhaul and maintenance. From reference 52, table 6.6, we take 0.2 as its value. Thus, the program cost associated with depots,  $C_{DEP}$ , is  $0.2C_{OPS}$ .

#### C.1.7 The program miscellaneous cost, $C_{MIS}$

The program miscellaneous cost is comprised of many factors, such as requirement for

technical data and maintenance functions, requirement for support equipment etc.

If the real value needs to be taken, the detailed cost estimating relations of above item should be provided in advance.

In preliminary design, according to reference 52, eq. 6.19 can be used to estimate the program miscellaneous cost,  $C_{MIS}$ .

$$C_{MIS} = 4C_{CMAT} \quad (C.1.14)$$

### C.1.8 The operation cost of military airplanes, $C_{OPS}$

In order to calculate the value of the operation cost of military airplanes, according to reference 52, chap 6.8, eq. C.1.1 can be rewritten as:

$$C_{OPS} = \frac{C_{POL} + C_{PDIR} + 5C_{CMAT}}{1 - f_{pind} - f_{sp} - f_{dep}} \quad (C.1.15)$$

Substituting these variables, from the previous results, into eq. C.1.15, the value of the operation cost of military airplanes,  $C_{OPS}$ , is from USD 917,870,490 to USD 1,816,886,958.

Above difference is due to the number of maintenance manhours,  $MHR_{fl}$ , required per flight hour. In this example, we assume its value is from 25 to 50, so the difference is large.

## C.2 Acquisition cost

Reference 52, Chap. 4 stated that the acquisition cost is the manufacturing cost plus the profit made by the manufacturer.

$$C_{ACQ} = C_{MAN} + C_{PRO} \quad (C.2.1)$$

The price paid by the customer to the airplane manufacturers depends on several factors. One of the most important factors is the total number of airplanes built by the manufacturer. The more airplanes they built, the less the price will be paid by the customer. From reference 52, eq. 4.4, the total airplane program manufacturing cost can be broken down into the following cost categories.

$$C_{MAN} = C_{aedm} + C_{apcm} + C_{fom} + C_{finm} \quad (C.2.2)$$

In eq. C.2.2,  $C_{aedm}$  is airframe engineering and design cost,  $C_{apcm}$  is airplane production cost,  $C_{fom}$  is production flight test operations cost, and  $C_{finm}$  is cost of financing the manufacturing program. Following sections are the calculation.

### C.2.1 The airframe engineering and design cost, $C_{aedm}$

The airframe engineering and design cost can be calculated by reference 52 eq. 4.5a.

$$C_{aedm} = MHR_{aedp}R_{em} - MHR_{aedr}R_{er} \quad (C.2.3)$$

$R_{em}$  and  $R_{er}$  is engineering manhour rate (USD per hour) at manufacturing phase and RDT&E phase respectively and reference 52 suggests these values may be assumed equal to each other.  $MHR_{aedp}$  is the total amount of engineering manhours required for the entire airplane program, for total number of airplanes produced,  $N_{pro}$ . Reference 52, eq. 4.6 can be used to calculate its value.

$$MHR_{aedp} = 0.0396W_{ampr}^{0.791}V_{max}^{1.526}N_{pro}^{0.183}F_{diff}F_{cad} \quad (C.2.4)$$

$W_{ampr}$  is the aeronautical manufacturers planning report weight. From reference 52, eq. 3.4, it is equal to the empty weight subtract the weight of engine, trapped fuel and oil, and weight of APU etc. From reference 8, eq. 5.1, empty weight,  $W_{emp}$ , is:

$$W_{emp} = W_{TO} - W_{fuel} - W_{pay} \quad (C.2.5)$$

Tier-II Plus take-off weight is 23,195.8 lbs, mission fuel is 13,312 lbs and payload weight is 2000 lbs. Substituting these variables into eq. C.2.5, then we obtain the weight of empty is 7883.8 lbs.

If we assume the weight of wheels, brakes tires and tyres to be half the weight of landing gear systems, 208.22 lbs, the engine weight is 1581 lbs, and weight of instruments, electronics equipment, and electrical power supply systems etc, is assumed equal to the weight of instrumentation and avionics plus electronics, 538.55 lbs and subtracted by the empty weight, finally we obtain the value of  $W_{ampr}$  is 5556 lbs. The maximum speed is assumed to be 1.25 times the maximum climb speed at sea level. For Tier-II Plus, the maximum climb speed at sea level is assumed to be 200 KEAS. Therefore, the maximum speed,  $V_{max}$ , is 250 KEAS.

From reference 52, eq. 4.1,  $N_{pro}$  the number of airplanes produced during an airplane program.

$$N_{pro} = N_m + N_{rdte} \quad (C.2.6)$$

$N_m$  is the number of airplanes produced to production standard during an airplane program, from reference 26, the number is 8.  $N_{rdte}$  is the number of airplanes produced during the RDTE phases, from reference 26, the number is 2. Therefore, the number of airplanes produced during an airplane program is 10.

$F_{diff}$  is a judgement factor which accounts for the difficulty of a new airplane program. From reference 52, Chap 3.1, the value of  $F_{diff}$  for Tier-II Plus is between 1.5 to 2.0, it involves moderately to very aggressive use of advanced technology. In order to simplify its calculation, we take 1.8 as  $F_{diff}$ .

$F_{cad}$  is a judgement factor which accounts for the effect of computer aided design capability on airframe engineering and design cost. From reference 52, Chap. 3.1, the value of  $F_{cad}$  is 0.8; for manufacturers which are experienced in the use of CAD. According to reference 52, the value of  $R_{em}$  and  $R_{er}$  are equal, and may be calculated by eq. 3.6.

$$R_{emthernyr} = R_{em1989} \frac{CEF_{thernyr}}{CEF_{1989}} \quad (C.2.7)$$

From section C.1.2, the value of  $CEF_{1989}$  and  $CEF_{1996}$  were given, and from reference 52, fig. 3.3, the value of engineering manhour rate (USD per hour),  $R_{em1989}$  is 61 USD/hr. Substituting these variables into eq. C.2.7, the value of  $R_{em1996}$  is 65.65 USD/hr.

Substituting these variables into eq. C.2.4, then the value of  $MHR_{aedp}$  is 363,430 hours. Now the only unknown variable in eq. C.2.3 is  $MHR_{aedr}$ . From reference 52, eq. 3.2 can be used.

$$MHR_{aedr} = 0.0396 W_{ampr}^{0.791} V_{max}^{1.526} N_{rdte}^{0.183} F_{diff} F_{cad} \quad (C.2.8)$$

In eq. C.2.8, only one variable is unknown,  $N_{rdte}$ . From reference 26, the number of airplane were build in RDTE phase is 2. Substituting these variables into eq. C.2.8, the value of  $MHR_{aedr}$  is 270,713 hours.

Finally, substituting these variables into eq. C.2.3, then the airframe engineering and design cost,  $C_{aedm}$  is USD 41,631,485.

## C.2.2 The airplane program production cost, $C_{apcm}$

Reference 52, eq.4.7 can be used to calculate airplane program production cost.

$$C_{apcm} = C_{(e+a)m} + C_{intm} + C_{manm} + C_{matm} + C_{toolm} + C_{qcm} \quad (C.2.9)$$

$C_{(a+e)m}$  is the cost of engine and avionics as acquired from vendors,  $C_{intm}$  is the cost of interior,  $C_{manm}$  is the manufacturing labour cost,  $C_{matm}$  is manufacturing material cost,  $C_{toolm}$  is the tooling cost, and  $C_{qcm}$  is quality control cost.

### C.2.2.1 The cost of engine and avionics, $C_{(a+e)m}$

From reference 52, the cost of engine and avionics can be estimated by eq. 4.7.

$$C_{(a+e)m} = (C_{em} N_e + C_{pm} N_p + C_{aviom}) N_m \quad (C.2.10)$$

In eq. C.2.10,  $C_{em}$  is the cost per engine during the manufacturing phase. Due to lacking further information, from reference 52, eq. B.10, may be used to estimate its value.

$$EP_{1989} = inv \log(2.3044 + 0.8858 \log T_{Toe}) \quad (C.2.11)$$



For Tier-II Plus, the engine net thrust at sea-level is 7150 lbs, substituting into eq. C.2.11, the value of per engine, in 1989, is USD 522,998.

By using eq. C. 2.7, the value of  $EP_{1989}$  can be modified with the 1996 to 1989 CEF-ratio, Thus,  $EP_{1996}$  is USD 562,829. Due to lacking of real data, we assume the price of engine during production is the same as the price in RDTE. Therefore, per engine cost,  $C_{em}$ , in 1996, is USD 562,829.

$N_e$  is the number of engine per airplane, for Tier-II Plus is 1.

$C_{pm}$  is the cost per propeller, for Tier-II Plus the propulsion system is turbofan engine. Therefore,  $C_{pm}$  is zero.

$C_{aviom}$  is the cost of avionics equipment per airplane. Reference 53, Chap. 1 stated that the avionics equipment on a modern military aircraft, such as maritime patrol aircraft can account for 40 percent of the total cost of the aircraft. For an airborne early warning aircraft like AWACS, can be over 75 percent of the total cost of the aircraft.

The Tier-II Plus is much like an electronic warfare airplane, for lacking real information, we assume the avionics equipment is 40 percent of the total aircraft price. Reference 26 stated that the aircraft price, in 1994, is USD 10 million each, therefore the avionics price, in 1994, is 4 million. By using eq. C.2.7, and from reference 52, fig 2.7, the value of  $CEF_{1994}$  is 3.18, so the avionics equipment price, in 1996, is USD 4,088,050.

$N_m$  is the number of airplane built during production. according to reference, the number is 8.

Substituting these variables into eq. C.2.10, then the value of  $C_{(a+c)m}$  is USD 37,207,035.

### C.2.2.2 The cost of interior, $C_{intm}$

$C_{intm}$  is the cost of the airplane interior. For unmanned aircraft this item is zero.

### C.2.2.3 The manufacturing labour cost, $C_{manm}$

$C_{manm}$  is the labour cost incurred in manufacturing  $N_m$  airplanes to production standard. From reference 52, eq. 4.10a can be used to calculate its value.

$$C_{manm} = MHR_{manp} R_{mm} - MHR_{manr} R_{mr} \quad (C.2.12)$$

In eq. C.2.12,  $MHR_{manp}$  represents the total number of manhours required for the manufacturing of  $N_{pro}$  airplanes, and can be found from reference 52, eq. 4.11.

$$MHR_{manp} = 28.984 W_{ampr}^{0.74} V_{max}^{0.543} N_{pro}^{0.524} F_{diff} \quad (C.2.13)$$

The four variables in eq. C.2.13 had been solved in section C.2.1. Substituting these variables into eq. C.2.13, the value of  $MHR_{manp}$  is 2,063,651 hours.

$R_{mm}$  is the manufacturing labour rate per hour for the entire program. From reference 52,  $R_{mm}$  is assumed to be equal to the manufacturing labour rate per hour during RDTE,  $R_{mr}$ . By using reference 52, fig 3.4. in 1989,  $R_{mm}$  is 35.35 USD/mhr, by using eq. C.2.7, we obtain the value of  $R_{mm}$  in 1996 is 38 USD/mhr.

$MHR_{manr}$  is the number of manufacturing manhours required in RDTE, and can be estimated by using reference 52, eq. 3.11.

$$MHR_{manr} = 28.984 W_{ampr}^{0.74} V_{max}^{0.543} N_{rde}^{0.524} F_{diff} \quad (C.2.14)$$

From reference 26, in RDTE phase, the number of airplanes,  $N_{rde}$  is 2. Therefore, substituting these variables into eq. C.2.14, the value of  $MHR_{manr}$  is 887,924 hours.

Finally, substituting these variables into eq. C.2.12, the value of the manufacturing labour cost,  $C_{manm}$  is USD 44,677,626.

#### C.2.2.4 The manufacturing material cost, $C_{matm}$

$C_{matm}$  is the materials cost incurred while manufacturing  $N_m$  airplanes, and can be estimated by using reference 52, eq. 4.12.

$$C_{matm} = C_{matp} - C_{matr} \quad (C.2.15)$$

In eq. C.2.15,  $C_{matp}$  is the total materials cost associated with building  $N_{pro}$  airplanes, and can be estimated by using reference 52, eq. 4.13.

$$C_{matp} = 37.632 F_{mat} W_{ampr}^{0.689} V_{max}^{0.624} N_{pro}^{0.792} CEF \quad (C.2.16)$$

$C_{matr}$  is the cost of materials to manufacture the flight test airplanes, and can be estimated by using reference 52, eq. 3.12.

$$C_{matr} = 37.632 F_{mat} W_{ampr}^{0.689} V_{max}^{0.624} N_{rde}^{0.792} CEF \quad (C.2.17)$$

$F_{mat}$  is a correction factor which depends on the type of materials used in the construction of the airplane. From reference 52, p 31, the value of  $F_{mat}$  is in between 2.5 to 3. For Tier-II Plus, most of the materials is advanced composite. Thus, we take 2.8 as the value of  $F_{mat}$ . From section C.1.2, the cost escalation factor, CEF, for 1996 is 3.25.

Substituting these variables into eq. C.2.16, and eq. C.2.17, the value of  $C_{matp}$  and  $C_{matr}$  are USD 25,296,151, and USD 7,070,827 respectively.

Finally, from eq. C.2.15, we obtain the value of the materials cost incurred while manufacturing  $N_m$  airplanes,  $C_{matm}$ , is USD 18,225,324.

#### C.2.2.5 The tooling cost, $C_{toolm}$

$C_{toolm}$  is the tooling cost to produce  $N_m$  airplanes, and can be estimated by using reference 52, eq. 4.14a.

$$C_{toolm} = MHR_{toolp} R_{tm} - MHR_{toolr} R_{tr} \quad (C.2.18)$$

In eq. C.2.18,  $MHR_{toolp}$  is the total number of tooling manhours required to build  $N_{pro}$  airplanes, and can be estimated by using reference 52, eq. 4.15.

$$MHR_{toolp} = 4.0127W_{ampr}^{0.764}V_{max}^{0.899}N_{pro}^{0.178}N_{rm}^{0.066}F_{diff} \quad (C.2.19)$$

In eq. C.2.19,  $N_{rm}$  is the airplane manufacturing rate in unit per month. From reference 26, there are 8 airplanes will be build in 2 years. Thus, the production rate 0.3333 unit per month. Substituting these variables into eq. C.2.19, the value of  $MHR_{toolp}$  is 1,051,872 hours.

$R_{tm}$  is the tooling labour rate in USD per manhour. From reference 52, fig 3.5, in 1989,  $R_{tm}$  is 43.06. Thus, by using eq. C.2.7, by 1996, the value of  $R_{tm}$  is 46.34. From reference 52, p: 54, the value of  $R_{tm}$  may different from  $R_{tr}$ , but in preliminary design phase, we assume that they are equal.

$MHR_{toolr}$  is the tooling manhours required in RDTE phase, and can be estimated by using reference 52, eq. 3.14.

$$MHR_{toolr} = 4.0127W_{ampr}^{0.764}V_{max}^{0.899}N_{rdte}^{0.178}N_{rr}^{0.066}F_{diff} \quad (C.2.20)$$

$N_{rr}$  is the airplane manufacturing rate in the RDTE phase per month. From reference 26, there are 2 airplanes will be build in 31 months. Thus, the production rate 0.0645 unit per month. Substituting these variables into eq. C.2.20, the value of  $MHR_{toolr}$  is 708,712 hours.

Substituting these variables into eq. C.2.18, the value of the tooling cost to produce  $N_m$  airplanes,  $C_{toolm}$ , is USD 15,902,034.

#### C.2.2.6 The quality control cost. $C_{qcm}$

$C_{qcm}$  is the quality control cost associated with building  $N_m$  airplanes, and can be estimated by using reference 52, eq. 4.16.

$$C_{qcm} = 0.13C_{manm} \quad (C.2.21)$$

From section C.2.2.3, the value of the manufacturing labour cost,  $C_{manm}$  is USD 44,677,626. Substituting this variable into eq. C.2.21, then the value of the quality control cost associated with building  $N_m$  airplanes,  $C_{qcm}$ , is USD 5,808,091.

Finally, substituting these variables, from C.2.2.1 to C.2.2.6, into eq. C.2.9, then the airplane program production cost,  $C_{apcm}$ , is USD 123,007,003.

#### C.2.3 The production flight test operations cost, $C_{ftom}$

From reference 52, the production flight test operations cost can be calculated by using eq. 4.17.

$$C_{ftom} = N_m C_{ophr} t_{pft} F_{ftoh} \quad (C.2.22)$$

In eq. C.2.22, only one variable,  $N_m$ , is given. There are three variables need to be solved.

$C_{ophr}$  is the airplane operating cost per hour. From section C.1.8, the value of the operation cost of military airplanes,  $C_{OPS}$ , is USD 917,870,490 to USD 1,816,886,958. From reference 52, eq. 6.23, the airplane operating cost per hour,  $C_{ophr}$ , is

$$C_{ophr} = \frac{C_{OPS}}{N_{serv} N_{yr} U_{anu}} \quad (C.2.23)$$

Substituting these variables, from the previous calculation, into eq. C.2.23, then the value of  $C_{ophr}$  is from 5322 to 10535 USD/hr.

$t_{pft}$  is the number of flight test hours flown by the manufacturer before airplane delivery to the customer. From reference 52, p: 55, it suggests 20 hours for military airplanes.

$F_{floh}$  is the overhead factor associated with the production flight test activities. Due to lacking real information, from reference 52, it suggests choosing 4 as its value.

Substituting these variables into eq. C.2.22, then the production flight test operations cost,  $C_{ftom}$ , is from USD 3,406,080 to 6,742,380.

#### C.2.4 The cost to finance the manufacturing phase, $C_{finm}$

From reference 52, in preliminary design, the cost to finance the manufacturing phase can be calculated by eq. 4.18.

$$C_{finm} = F_{finm} C_{MAN} \quad (C.2.24)$$

In eq. C.2.24 the value of financing cost,  $F_{finm}$ , is depends on the interest rates. Due to lacking of real data, from reference 52, p: 56, the value of 0.12 has been taken.

#### C.2.5 The total airplane program manufacturing cost, $C_{MAN}$

In order to obtain the value of the total airplane program manufacturing cost,  $C_{MAN}$ , we rewrite eq. C.2.2 as;

$$C_{MAN} = \frac{C_{aedm} + C_{apcm} + C_{ftom}}{1 - F_{finm}} \quad (C.2.25)$$

Substituting theses variables, from section C.2.1 to C.2.4, into eq. C.2.25, we obtain the value of the total airplane program manufacturing cost,  $C_{MAN}$ , is from USD 190,959,736 to 194,750,986.

#### C.2.6 Profit, $C_{PRO}$

Privately held aeronautical enterprises will want to make a profit on airplane

manufacturing activities, and this profit should include in the cost of airplane acquisition. From reference 52, eq. 4.19, can be used to estimate its value.

$$C_{PRO} = F_{prom} C_{MAN} \quad (C.2.26)$$

Though profit can vary significantly with many factors, such as politics or market conditions. For lacking of real information, from reference 52, p: 56, we take profit factor,  $F_{prom}$ , as 10 percent. And the value of  $C_{MAN}$  was given by the previous section. Substituting these variables into eq. C.2.26, the value of Profit,  $C_{PRO}$ , is from USD 19,095,974 to 19,475,099.

From eq. C.2.1, the acquisition cost is the manufacturing cost plus the profit made by the manufacturer. Thus, the value of  $C_{ACQ}$  is from USD 210,055,710 to 214,226,085.