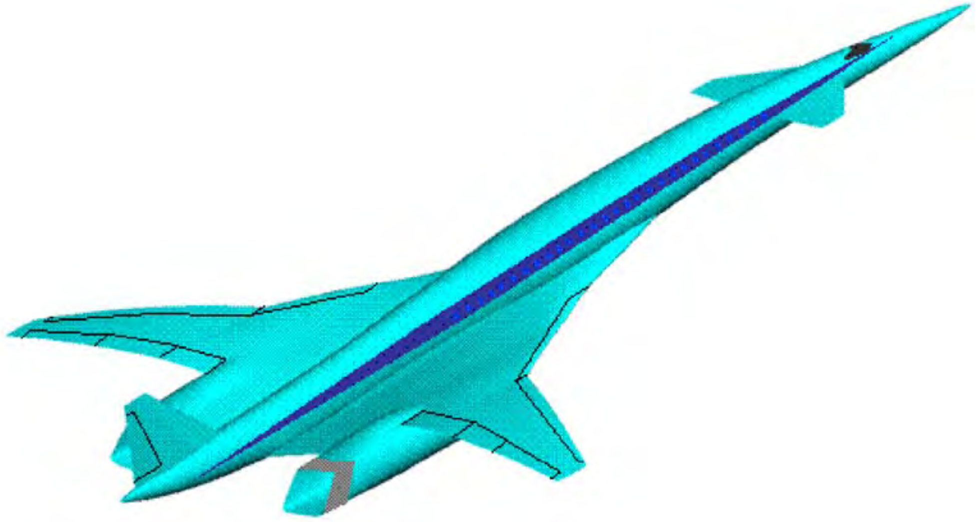
AIAA Foundation Student Design Competition 2017/18
Undergraduate Team – Engine

Candidate Engines for a Next Generation Supersonic Transport



- Request for Proposal –

Abstract

New engine designs are solicited for the next generation supersonic transport. Entry into service is expected to be 2010 – 2025. This current solicitation is motivated by NASA's National Research Announcement (NRA) back in 2006 (NNH06ZEA001N, Amendment 6, Task 4.7) for a supersonic transport vehicle. The NRA is calling for an aircraft that is a generation beyond the supersonic business aircraft that is currently being considered and smaller than the supersonic airliners of past NASA programs (i.e. High Speed Civil Transport). The baseline propulsion system is based on the engine modeled in NASA/CR-2010-216842. The candidate engines must demonstrate at least 5% improvement in TSFC (at specified thrust levels) and substantiate weight savings. Furthermore, both cruise emissions goal and noise constraint (represented by exit jet velocity) are imposed.

Data for a generic baseline model of the baseline power plant is supplied. Responders should use the provided typical, multi-segment, mission to address the design improvements, especially for design point and off-design engine operations. The performance and total fuel consumption of the candidate engine should be estimated for critical mission points and stated clearly in the proposal. Special attention should be paid to engine mass, dimensions and integration with the aircraft.

1.0 Introduction

The supersonic transport under consideration (ref. Welge et al. 2010) is intended to carry 100 passengers at Mach 1.6 over a range of 4000 nmi. Some relevant aircraft characteristics are given in Table 1.

Table 1: General Aircraft Characteristics (Welge, et al, 2010)

General characteristics	
Max. take-off weight	317,499 lb
Payload weight	21,000 lb
Operating empty weight	146,420 lb
Wing loading (takeoff)	77.5 psf
Power plant	2 × mixed-flow turbofans; 61,000 lbf each @ SLS
Performance	
Maximum speed	Mach 1.8 at 55,000 feet
Cruise speed	Mach 1.6 at 50,000-55,000 feet
Range	4000 nmi
Cruise L/D	9.2

For reference, a supersonic non-stop mission profile is presented in Figure 1 and Figure 2.

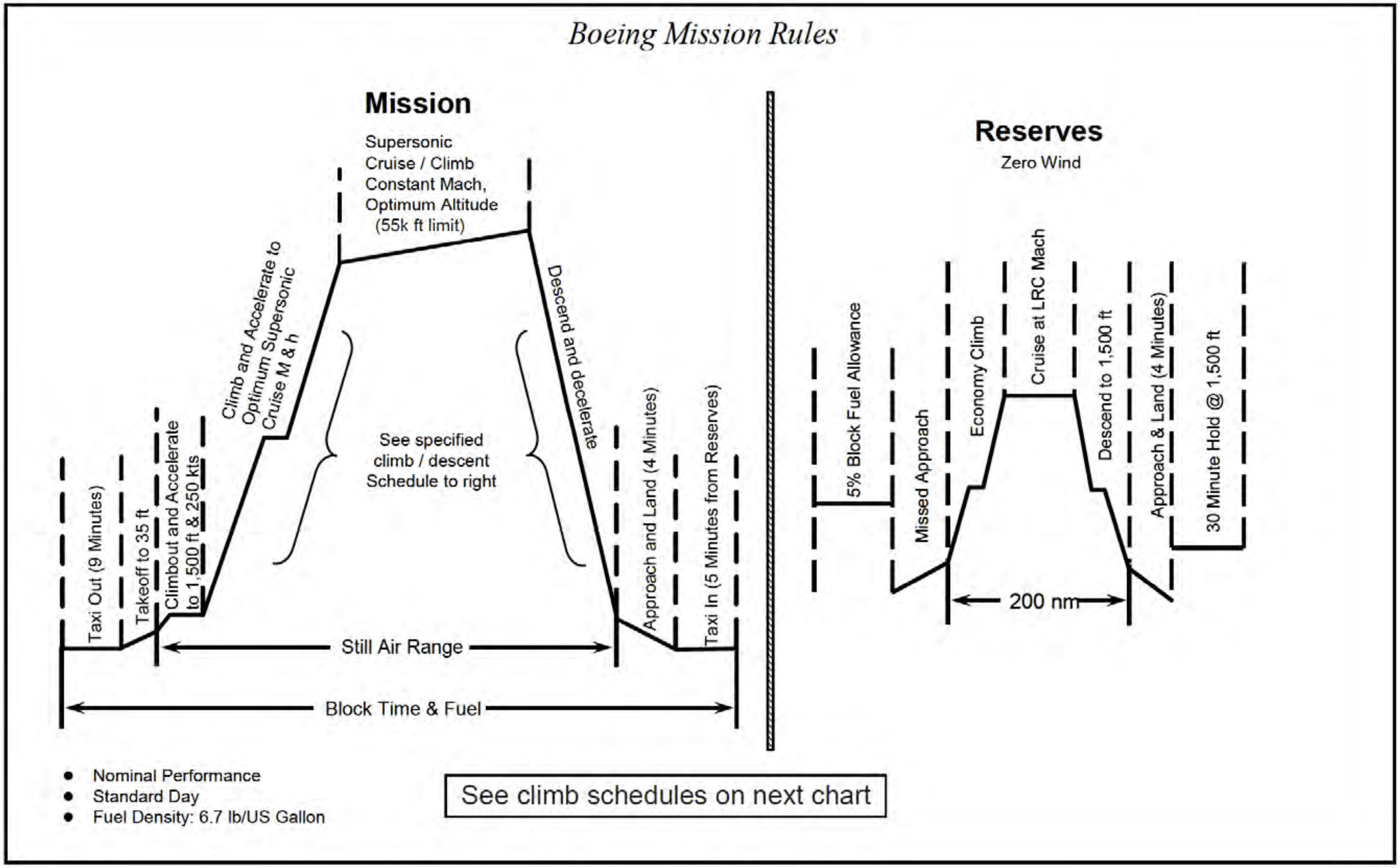
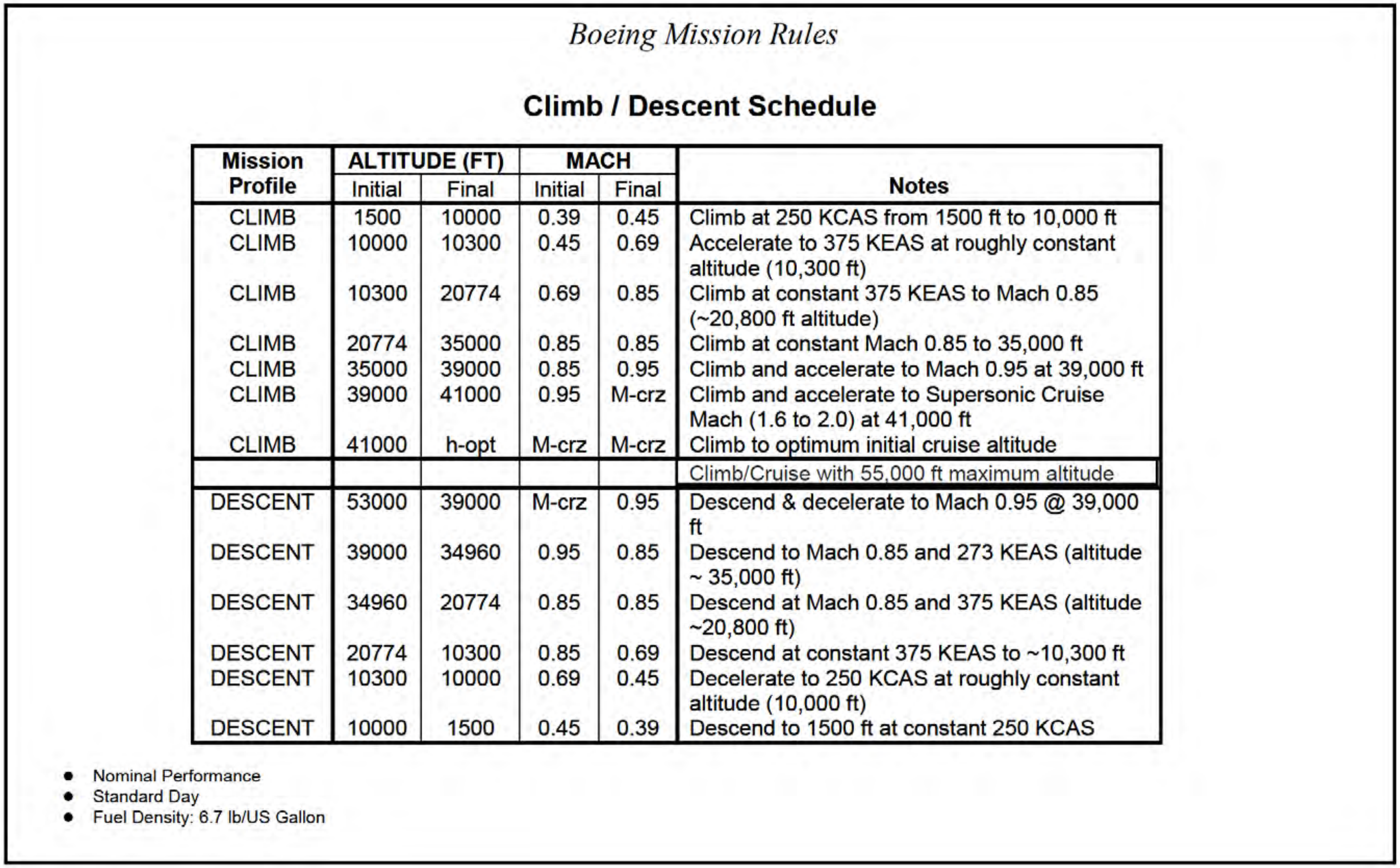


Figure 1. Supersonic Non-Stop Mission Profile (Welge, et al, 2010)



Boeing Mission Rules

Climb / Descent Schedule

Mission Profile	ALTITUDE (FT)		MACH		Notes
	Initial	Final	Initial	Final	
CLIMB	1500	10000	0.39	0.45	Climb at 250 KCAS from 1500 ft to 10,000 ft
CLIMB	10000	10300	0.45	0.69	Accelerate to 375 KEAS at roughly constant altitude (10,300 ft)
CLIMB	10300	20774	0.69	0.85	Climb at constant 375 KEAS to Mach 0.85 (~20,800 ft altitude)
CLIMB	20774	35000	0.85	0.85	Climb at constant Mach 0.85 to 35,000 ft
CLIMB	35000	39000	0.85	0.95	Climb and accelerate to Mach 0.95 at 39,000 ft
CLIMB	39000	41000	0.95	M-crz	Climb and accelerate to Supersonic Cruise Mach (1.6 to 2.0) at 41,000 ft
CLIMB	41000	h-opt	M-crz	M-crz	Climb to optimum initial cruise altitude
					Climb/Cruise with 55,000 ft maximum altitude
DESCENT	53000	39000	M-crz	0.95	Descend & decelerate to Mach 0.95 @ 39,000 ft
DESCENT	39000	34960	0.95	0.85	Descend to Mach 0.85 and 273 KEAS (altitude ~ 35,000 ft)
DESCENT	34960	20774	0.85	0.85	Descend at Mach 0.85 and 375 KEAS (altitude ~20,800 ft)
DESCENT	20774	10300	0.85	0.69	Descend at constant 375 KEAS to ~10,300 ft
DESCENT	10300	10000	0.69	0.45	Decelerate to 250 KCAS at roughly constant altitude (10,000 ft)
DESCENT	10000	1500	0.45	0.39	Descend to 1500 ft at constant 250 KCAS

- Nominal Performance
- Standard Day
- Fuel Density: 6.7 lb/US Gallon

Figure 2. Supersonic Non-Stop Mission Profile (cont'd) (Welge, et al, 2010)

2.0 Engine Design Objectives & Requirements

- A new engine design is required for a future version of a 100-passenger supersonic transport, with an entry-into-service date of 2025.

- The future flight envelope ranges from take-off at static sea-level conditions to supersonic cruise at up to 55,000 feet/Mach 1.8. It is hoped that the range might be extended by reducing the fuel consumption and minimizing engine mass.
- The generic baseline engine model given in section 4.0 should be used as a starting point, and the new design should be optimized for minimum engine mass and fuel burn, based on trade studies to determine the best combination of fan pressure ratio, bypass ratio, overall pressure ratio and turbine entry temperature. Students should attempt to maximize vehicle flight range. Values of these four major design parameters should be compatible with those expected to be available in 2025 and the selected design limits should be justified in the proposal. Teams should use the provided aircraft trade factors to justify tradeoffs between engine weight and fuel consumption on vehicle performance.
- Based on the entry into service date, the development of new materials and an increase in design limits may be assumed. The development and potential application of carbon matrix composites is of particular interest. Based on research of available literature, justify carefully your choices of any new materials, their location within the engine and the appropriate advances in design limits that they provide.
- Engine Physical Characteristics
  Different engine architecture is permitted, but accommodation within the existing nacelle envelope is preferred. The baseline engine fan diameter is 87.5 inches, and the bare engine weight, excluding the inlet, is estimated to be 13,000 pounds. Engine architectures incorporating variable cycle technology are permitted; however, if applied, additional justification must be provided showing off-design performance calculations and supporting details. Accuracy of this additional information will be included in the judging process.
- Engine Thrust and TSFC Requirements
  The engine shall meet or exceed the installed thrust levels shown in Table 2 at the specified flight conditions. In addition, the thrust specific fuel consumption (TSFC) at each condition shall be at least 5% lower than the values shown in the table when evaluated at the required thrust as described below.

Table 2. Installed Engine Thrust and TSFC Requirements (see notes 1-3)

	Altitude (ft)	Mach	dTamb (°F)	FN (lbf)	TSFC (lbm/hr/lbf)
SLS	0	0	0	64625	0.520
Hot Day Take-off	0	0.25	27	56570	0.652
Transonic Pinch	40550	1.129	0	14278	0.950
Supersonic Cruise	52500	1.6	0	14685	1.091

Notes:

(1) International Standard Atmosphere (ISA) day. The fuel lower heating value (LHV) is 18400 BTU/lbm.

(2) Installed performance includes inlet recovery, inlet drag and nozzle drag per installation curves provided in section 5.0, 1% HP compressor interstage customer bleed air, and 100 HP customer power extraction from the HP spool.

(3) Inlet recovery at SLS is 0.95. Inlet recovery at Hot Day Take-off is 0.9588. Inlet recoveries at supersonic conditions are provided in section 5.0.

Uninstalled engine performance characteristics are also provided in Table 3 for reference.

Table 3. Uninstalled Engine Thrust and TSFC Requirements (see note 1)

	Altitude (ft)	Mach	dTamb (°F)	FN (lbf)	TSFC (lbm/hr/lbf)
SLS	0	0	0	70551	0.494
Hot Day Take-off	0	0.25	27	61190	0.620
Transonic Pinch	40550	1.129	0	17197	0.804
Supersonic Cruise	52500	1.6	0	16471	0.993

Notes:

(1) International Standard Atmosphere (ISA) day, uninstalled (i.e., zero customer bleed and zero customer horsepower extraction, and MIL-E-5007D inlet pressure recovery). The fuel lower heating value (LHV) is 18400 BTU/lbm.

- Procedure for Evaluation of Thrust and TSFC Margins
  To demonstrate that the design meets the thrust and TSFC requirements, the margins shall be evaluated as shown in Figure 3. The engine must have a positive thrust margin. The TSFC margin shall be at least 5% and must be computed at the specified thrust level, not at the maximum thrust level of the engine.

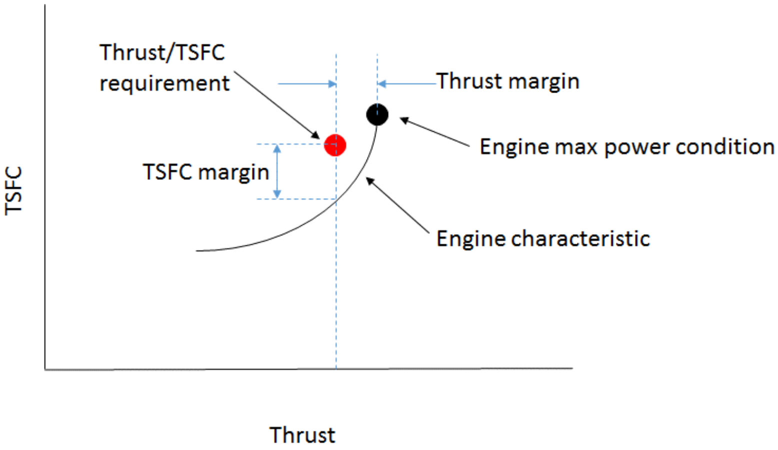


Figure 3. Thrust and TSFC Margin Definitions

$$Thrust\ Margin = \left(\frac{Engine\ Maximum\ Thrust - Required\ Thrust}{Required\ Thrust}\right) * 100$$

$$TSFC\ Margin = \left(\frac{Required\ TSFC - Engine\ TSFC@Required\ Thrust}{Required\ TSFC}\right) * 100$$

- Inlets and Nozzles

  An appropriate inlet must be designed. The inlet should be designed to optimize internal performance (e.g. inlet pressure recovery) and to minimize inlet propulsion system drags. A 2-ramp inlet, either axisymmetric or 2-dimensional configuration is suggested but is not mandatory.

  To enable efficient supersonic cruise, and to meet current noise restrictions at take-off, an appropriate convergent-divergent noise-attenuating nozzle must also be designed. The nozzle should be designed to optimize internal performance (e.g. gross thrust coefficient) and to minimize nozzle propulsion system drags.

- Noise

  Analysis has indicated that airport noise requirements may be met without a noise suppressor on the nozzle if the exhaust jet velocity can be kept at or below 1100 ft/sec. However low jet velocity generally corresponds to higher fan diameter, which is detrimental to supersonic aircraft performance. The engine shall be evaluated to determine if it may be operated at reduced power at takeoff, to meet an exhaust jet velocity requirement of 1375 ft/sec or lower while providing adequate takeoff thrust.

- NOx Emissions

  Extensive efforts are underway to develop combustors with lower nitrogen oxides (NOx) emissions for use in both subsonic and supersonic civil aircraft engines. Current standards are expressed in terms of a parameter which consists of the total mass of the emission produced during a prescribed landing-takeoff (LTO) operational cycle per kilonewton (kN) of rated takeoff thrust at sea level, standard day operating conditions. These cycles are intended to be representative of operations that occur at or near airports and at altitudes up to 900 meters above the airport. The definitions of the LTO cycles for both subsonic and supersonic engines are shown in Table 4.

Table 4. LTO Cycle Definitions

Operating Mode	Subsonic Engines		Supersonic Engines	
	Power (%)	Time in mode (minutes)	Power (%)	Time in mode (minutes)
Takeoff	100	0.7	100	1.2
Climbout	85	2.2	65	2.0
Descent	N/A	N/A	15	1.2
Approach	30	4.0	34	2.3
Taxi/Idle	7	26.0	5.8	26.0

The total emissions per unit of thrust for the LTO cycle is computed as follows:

$$Emission\ Mass\ per\ unit\ of\ Thrust\ \left(\frac{g}{kN}\right) = \sum \left[Emission\ Index\ \left(\frac{g}{kg\ fuel}\right) * TSFC\left(\frac{kg\ fuel}{hr\ kN}\right) * Time\ in\ Mode\ (hr)\right]$$

Current standards for LTO NOx for subsonic and supersonic aircraft may be found in (ICAO Annex 16). For supersonic aircraft, the maximum permissible NOx for the LTO cycle is a function of the engine sea level static maximum overall pressure ratio (OPR) as follows:

$$Allowable\ NOx\ Emission\ Mass\ per\ unit\ of\ Thrust\ \left(\frac{g}{kN}\right) = 36.0 + 2.42(OPR)$$

In the case of advanced supersonic transport aircraft engines, which cruise at higher altitudes where NOx emissions may contribute to depletion of the stratospheric ozone layer, standards have also been proposed at high altitude cruise conditions. While previous studies suggested that the impact of supersonic aircraft on the atmosphere was primarily through the role of water vapor emissions, recent work (Dessens et al, 2007) has re-emphasized the significance of NOx. A NOx emissions index of 5 g/kg fuel at cruise conditions has been established by NASA as a goal for future supersonic transport engines (Wesocky and Prather, 1991).

NOx emissions for the LTO cycle shall be estimated and shown to be in compliance with current standards for supersonic aircraft. NOx emissions for supersonic cruise shall be estimated and compared to the goal of 5 g/kg fuel. The NOx emissions at supersonic cruise conditions may be estimated from the sea level emissions using the “P3-T3” method or similar method as described in (Norman et al, 2003)

3.0 Proposal Requirements

- The design must be shown to meet the performance requirements presented in Section 2.0.

- Design proposals must include engine mass, engine dimensions, net thrust values, specific fuel consumption, thermal and propulsive efficiencies at take-off (standard sea-level conditions) and supersonic cruise. Details of the major flow path components must be given. These include inlet, fan, HP compressor, primary combustor, HP turbine, LP turbine, exhaust nozzle, bypass duct, and any inter-connecting ducts. A complete report compliance matrix is provided in Table 6.

- Required Justification for Design Assumptions
  The design report must include adequate justification for the component stage loadings and efficiencies assumed, the maximum turbine rotor inlet temperature assumed, the material properties assumed, and the levels of turbine cooling effectiveness assumed. The design report shall also justify the assumed nozzle gross thrust coefficient and, if a mixed exhaust is used, the exhaust mixing effectiveness.

- In addition to providing details of the design, teams must provide justification of design choices through appropriate trade studies and presentation of publicly available information regarding chosen technology levels and assumptions. To help guide teams a list of required trade studies, tables, and plots to be carried out and presented are listed below:

    - Aircraft constraint diagram showing chosen thrust to weight and wing loading that satisfy the aircraft requirements given in Table 1.
    - The following engine trade studies are encouraged to justify choice of the optimum cycle that meets the vehicle requirements. Bonus points will be awarded for teams that perform an in-depth investigation of the drivers and explain the resulting trends.

        - FPR vs. BPR vs. Mission Fuel Burn
        - OPR vs. T4.1 max vs. Mission Fuel Burn
        - FPR vs. OPR vs. Mission Fuel Burn
        - BPR vs. T4.1 vs. Mission Fuel Burn
        - FPR vs. BPR vs. cruise TSFC
        - OPR vs. T4.1 max vs. cruise TSFC
        - FPR vs. OPR vs. cruise TSFC
        - BPR vs. T4.1 vs. cruise TSFC
        - FPR vs. BPR vs. engine weight
        - OPR vs. T4.1 max vs. engine weight
        - FPR vs. OPR vs. engine weight
        - BPR vs. T4.1 vs. engine weight

    - An in-depth cycle summary showing information from Table 7
    - Perform a design point design of the engine and show:

        - Velocity triangles for each stage of the compressor and turbine at the hub, mid-section, and tip
        - Provide a cross-section of the engine flowpath, showing 2D geometry for the inlet, all compressors, the combustor, turbines, nozzle(s), and any transition ducts
        - Provide one set of hand calculations showing velocity triangle calculations for the first stage of each component

    - Bonus points will be awarded for 3D drawings of the engine components.
    - Table 8 shows some of the required detailed stage information for all compressors and turbines, other stage and component performance may be required to complete this information and should be shown as appropriate.

Table 5. Performance Requirements Matrix

Parameter	Required Value	Design Value	Margin Relative to Requirement
Takeoff Thrust			
Max Thrust at Transonic Pinch Point			
TSFC at Transonic Pinch Point			
Max Thrust at Supersonic Cruise			
TSFC at Supersonic Cruise			
Fan Diameter			
Bare Engine Weight (excl. inlet)			
Takeoff Exhaust Jet Velocity			
LTO NOx			
Supersonic Cruise NOx			

Table 6: Compliance Matrix

General characteristics	
Wing area	
Max. take-off weight	
Takeoff-Thrust	
Design Afterburning Thrust	
Performance	
Maximum speed	
Cruise speed	
Mission Fuel Burn	
Cruise TSFC	
Takeoff TSFC	
Engine Weight	
Fan Diameter	
Required Trade Studies	
Aircraft Constraint Diagram Page #	
Engine Cycle Design Space Carpet Plots Page #	
In-Depth Cycle Summary Page #	
Final engine flowpath (Page #)	
Final cycle study using chosen cycle program (Page #)	
Detailed stage-by-stage turbomachinery design information (page # for each component)	
Detailed design of velocity triangles for first stage of each component (list page #'s and component)	
Detailed inlet and nozzle performance characteristics (Page #)	

Table 7: Engine Summary Table

Summary Data	
Design MN	
Design Altitude	
Design Fan Mass Flow	
Design Gross Thrust	
Design Bypass Ratio	
Design Net Thrust	
Design TSFC	
Design Overall Pressure Ratio	
Design T4.1	
Design Engine Pressure Ratio	
Design Fan / LPC Pressure Ratio	
Design Chargeable Cooling Flow (%@25)	
Design Non-Chargeable Cooling Flow (%@25)	
Design Adiabatic Efficiency for Each Turbine	
Design Polytropic Efficiency for Each Compressor	
Design HP/IP/LP Shaft RPM	
Flow Station Data (List for Each Engine Component at Design Condition)	
Inflow	
Corrected Inflow	
Inflow Total Pressure	
Inflow Total Temperature	
Inflow Fuel-air-Ratio	
Inflow Mach #	
Inflow Area	
Pressure Loss/Rise Across Component	
Additional Information	
Design HP/LP Shaft Off-take Power	
Design Customer Bleed Flow	

Table 8: Required Detailed Stage and Component Information

Compressor		**Turbine**	
Lieblein Diffusion Factor		Zweifel Coefficient	
De Haller Number		AN^2	
Stage Pressure Ratio		Stage Pressure Ratio	
Work Coefficient		Work Coefficient	
Flow Coefficient		Flow Coefficient	
Hub-to-Tip Ratio		Hub-to-Tip Ratio	
Mean radius		Mean radius	
Number of Blades (Rotor & Stator)		Number of Blades (Rotor & Stator)	
Aspect Ratio		Aspect Ratio	
Taper Ratio		Taper Ratio	
Tip Speed		Tip Speed	
Stagger Angle		Stagger Angle	
Velocity Triangles (hub, mean, & tip)		Velocity Triangles (hub, mean, & tip)	
Blade chord		Blade chord	
Degree of Reaction		Degree of Reaction	
Mach Numbers (absolute & relative)		Mach Numbers (absolute & relative)	
		Turbine Rotor Inlet Temperature	
		Cooling Flow Details	

4.0 Baseline Engine Model

The baseline engine is a dual spool mixed-flow turbofan. A generic model has been generated from publicly-available information using *NPSS*. Certain details of this model are given below to assist with construction of a baseline case and to provide some indication of typical values of design parameters.

4.1 Overall Characteristics

Table 9 contains a summary of basic engine characteristics.

Table 9: Baseline Engine: Basic Data, Overall Geometry and Performance

Design Features of the Baseline Engine	
Engine Type	Mixed-flow turbofan
Fan pressure ratio	2.25
Overall pressure ratio at max. power	35.0
Bypass ratio at max. power	1.71
Maximum net thrust at sea level	69,600 lbf
Specific fuel consumption at max. power	0.51 lbm/hr/lbf
Fan diameter	89 inches
Number of fan stages	2
Number of compressor stages	11
Number of HP turbine stages	2
Number of LP turbine stages	4

4.2 Cycle Performance Summary

For modeling purposes, the component schematic diagram for the baseline engine is presented in Figure 4. Sample model output is provided in **Figure 5**. This model output was generated using NPSS. Nomenclature definitions for this output are provided in Table 10.

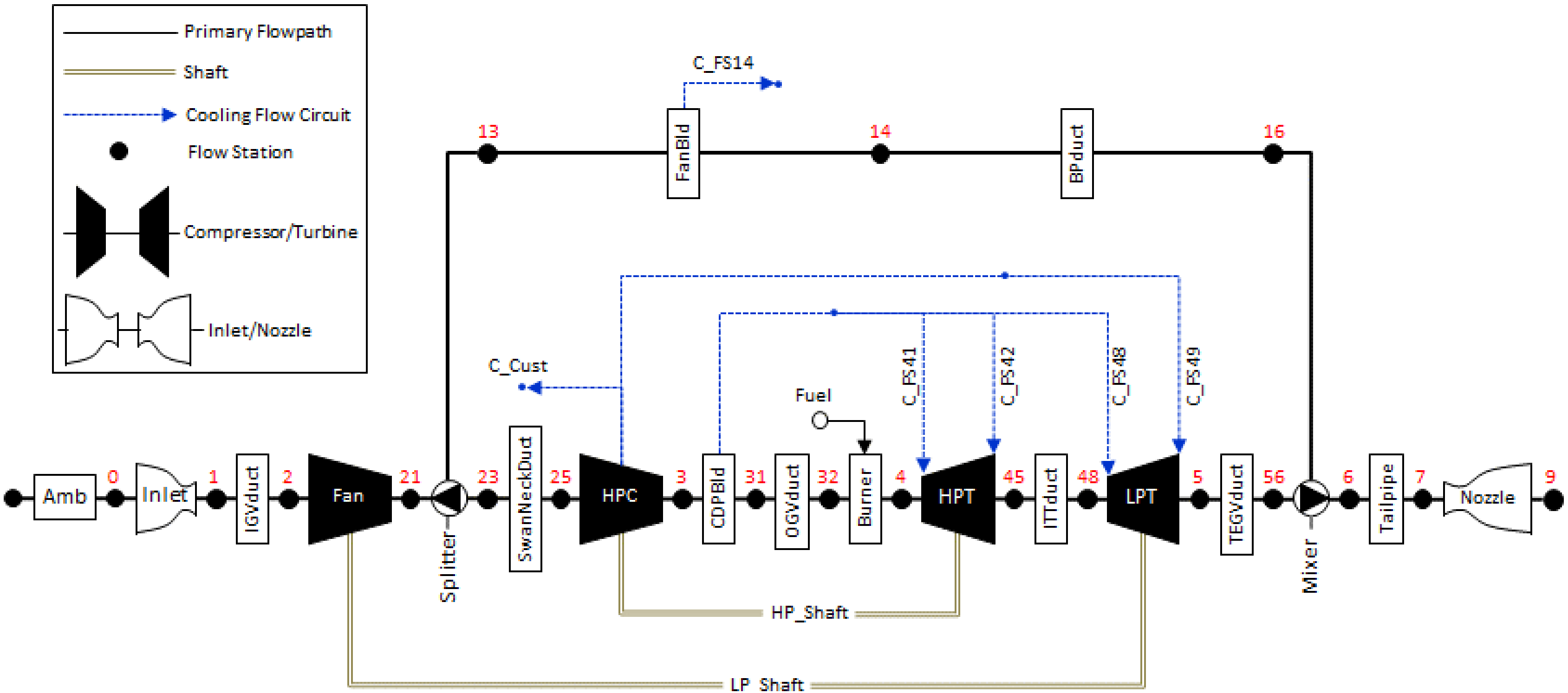
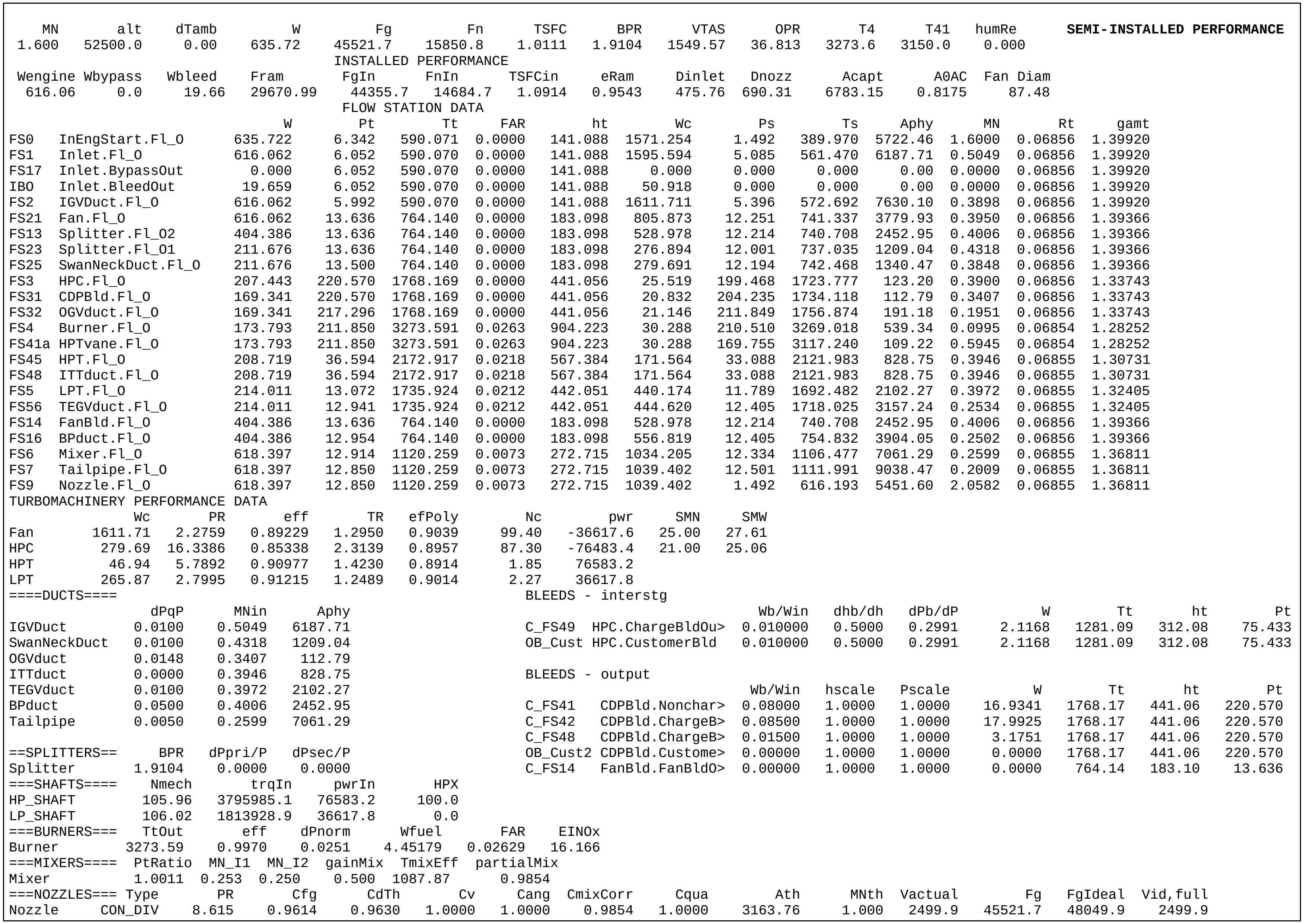


Figure 4. Baseline Engine Model Schematic



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    MN       alt    dTamb         W         Fg         Fn      TSFC      BPR      VTAS      OPR       T4      T41   humRe      SEMI-INSTALLED PERFORMANCE
 1.600   52500.0     0.00    635.72    45521.7    15850.8    1.0111   1.9104   1549.57   36.813   3273.6   3150.0    0.000
                                       INSTALLED PERFORMANCE
 Wengine Wbypass   Wbleed    Fram       FgIn      FnIn      TSFCin     eRam     Dinlet   Dnozz      Acapt      A0AC  Fan Diam
  616.06     0.0     19.66   29670.99    44355.7   14684.7   1.0914   0.9543    475.76    690.31  6783.15    0.8175     87.48
                                        FLOW STATION DATA
                                 W        Pt        Tt     FAR        ht        Wc        Ps        Ts     Aphy      MN       Rt     gamt
FS0   InEngStart.Fl_O      635.722     6.342   590.071  0.0000   141.088  1571.254     1.492   389.970  5722.46  1.6000  0.06856  1.39920
FS1   Inlet.Fl_O           616.062     6.052   590.070  0.0000   141.088  1595.594     5.085   561.470  6187.71  0.5049  0.06856  1.39920
FS17  Inlet.BypassOut        0.000     6.052   590.070  0.0000   141.088     0.000     0.000     0.000     0.00  0.0000  0.06856  1.39920
IBO   Inlet.BleedOut        19.659     6.052   590.070  0.0000   141.088    50.918     0.000     0.000     0.00  0.0000  0.06856  1.39920
FS2   IGVDuct.Fl_O         616.062     5.992   590.070  0.0000   141.088  1611.711     5.396   572.692  7630.10  0.3898  0.06856  1.39920
FS21  Fan.Fl_O             616.062    13.636   764.140  0.0000   183.098   805.873    12.251   741.337  3779.93  0.3950  0.06856  1.39366
FS13  Splitter.Fl_O2       404.386    13.636   764.140  0.0000   183.098   528.978    12.214   740.708  2452.95  0.4006  0.06856  1.39366
FS23  Splitter.Fl_O1       211.676    13.636   764.140  0.0000   183.098   276.894    12.001   737.035  1209.04  0.4318  0.06856  1.39366
FS25  SwanNeckDuct.Fl_O    211.676    13.500   764.140  0.0000   183.098   279.691    12.194   742.468  1340.47  0.3848  0.06856  1.39366
FS3   HPC.Fl_O             207.443   220.570  1768.169  0.0000   441.056    25.519   199.468  1723.777   123.20  0.3900  0.06856  1.33743
FS31  CDPBld.Fl_O          169.341   220.570  1768.169  0.0000   441.056    20.832   204.235  1734.118   112.79  0.3407  0.06856  1.33743
FS32  OGVduct.Fl_O         169.341   217.296  1768.169  0.0000   441.056    21.146   211.849  1756.874   191.18  0.1951  0.06856  1.33743
FS4   Burner.Fl_O          173.793   211.850  3273.591  0.0263   904.223    30.288   210.510  3269.018   539.34  0.0995  0.06854  1.28252
FS41a HPTvane.Fl_O         173.793   211.850  3273.591  0.0263   904.223    30.288   169.755  3117.240   109.22  0.5945  0.06854  1.28252
FS45  HPT.Fl_O             208.719    36.594  2172.917  0.0218   567.384   171.564    33.088  2121.983   828.75  0.3946  0.06855  1.30731
FS48  ITTduct.Fl_O         208.719    36.594  2172.917  0.0218   567.384   171.564    33.088  2121.983   828.75  0.3946  0.06855  1.30731
FS5   LPT.Fl_O             214.011    13.072  1735.924  0.0212   442.051   440.174    11.789  1692.482  2102.27  0.3972  0.06855  1.32405
FS56  TEGVduct.Fl_O        214.011    12.941  1735.924  0.0212   442.051   444.620    12.405  1718.025  3157.24  0.2534  0.06855  1.32405
FS14  FanBld.Fl_O          404.386    13.636   764.140  0.0000   183.098   528.978    12.214   740.708  2452.95  0.4006  0.06856  1.39366
FS16  BPduct.Fl_O          404.386    12.954   764.140  0.0000   183.098   556.819    12.405   754.832  3904.05  0.2502  0.06856  1.39366
FS6   Mixer.Fl_O           618.397    12.914  1120.259  0.0073   272.715  1034.205    12.334  1106.477  7061.29  0.2599  0.06855  1.36811
FS7   Tailpipe.Fl_O        618.397    12.850  1120.259  0.0073   272.715  1039.402    12.501  1111.991  9038.47  0.2009  0.06855  1.36811
FS9   Nozzle.Fl_O          618.397    12.850  1120.259  0.0073   272.715  1039.402     1.492   616.193  5451.60  2.0582  0.06855  1.36811
TURBOMACHINERY PERFORMANCE DATA
               Wc       PR       eff       TR   efPoly        Nc        pwr     SMN     SMW
Fan       1611.71   2.2759   0.89229   1.2950   0.9039     99.40   -36617.6   25.00   27.61
HPC        279.69  16.3386   0.85338   2.3139   0.8957     87.30   -76483.4   21.00   25.06
HPT         46.94   5.7892   0.90977   1.4230   0.8914      1.85    76583.2
LPT        265.87   2.7995   0.91215   1.2489   0.9014      2.27    36617.8
====DUCTS====                                                 BLEEDS - interstg
                 dPqP      MNin      Aphy                                                 Wb/Win   dhb/dh   dPb/dP          W        Tt       ht        Pt
IGVDuct        0.0100    0.5049   6187.71                     C_FS49  HPC.ChargeBldOu>  0.010000   0.5000   0.2991     2.1168   1281.09   312.08    75.433
SwanNeckDuct   0.0100    0.4318   1209.04                     OB_Cust HPC.CustomerBld   0.010000   0.5000   0.2991     2.1168   1281.09   312.08    75.433
OGVduct        0.0148    0.3407    112.79
ITTduct        0.0000    0.3946    828.75                     BLEEDS - output
TEGVduct       0.0100    0.3972   2102.27                                                Wb/Win   hscale   Pscale          W        Tt       ht        Pt
BPduct         0.0500    0.4006   2452.95                     C_FS41   CDPBld.Nonchar>  0.08000   1.0000   1.0000    16.9341   1768.17   441.06   220.570
Tailpipe       0.0050    0.2599   7061.29                     C_FS42   CDPBld.ChargeB>  0.08500   1.0000   1.0000    17.9925   1768.17   441.06   220.570
                                                              C_FS48   CDPBld.ChargeB>  0.01500   1.0000   1.0000     3.1751   1768.17   441.06   220.570
==SPLITTERS==     BPR   dPpri/P   dPsec/P                     OB_Cust2 CDPBld.Custome>  0.00000   1.0000   1.0000     0.0000   1768.17   441.06   220.570
Splitter       1.9104    0.0000    0.0000                     C_FS14   FanBld.FanBldO>  0.00000   1.0000   1.0000     0.0000    764.14   183.10    13.636
===SHAFTS====    Nmech       trqIn     pwrIn       HPX
HP_SHAFT        105.96   3795985.1   76583.2     100.0
LP_SHAFT        106.02   1813928.9   36617.8       0.0
===BURNERS===   TtOut       eff    dPnorm      Wfuel       FAR    EINOx
Burner        3273.59    0.9970    0.0251    4.45179   0.02629   16.166
===MIXERS====  PtRatio  MN_I1  MN_I2  gainMix  TmixEff  partialMix
Mixer          1.0011  0.253  0.250    0.500  1087.87      0.9854
===NOZZLES=== Type       PR       Cfg      CdTh       Cv     Cang  CmixCorr     Cqua        Ath      MNth  Vactual         Fg   FgIdeal  Vid,full
Nozzle     CON_DIV    8.615    0.9614    0.9630   1.0000   1.0000    0.9854   1.0000    3163.76     1.000   2499.9    45521.7   48049.9    2499.9
```

Figure 5. Baseline Engine Cycle Data

Table 10. Engine Model Output Nomenclature

Summary Output Data			**Inlets**		
MN	Flight Mach number		eRam	Ram recovery, Pt1/Pt0	
alt	Altitude	Feet	Afs	Free stream area	square inches
dTs	Delta temperature from standard day	degrees R	Fram	Ram drag	lbf
W	Mass Flow	lbm/sec			
Fg	Gross Thrust	lbf	**Ducts**		
Fn	Net Thrust	lbf	dPqP	Stagnation pressure loss fraction	
TSFC	Thrust Specific Fuel Consumption	lbm/hr/lbf	Mnin	Entrance Mach number	
Wfuel	Fuel Flow	lbm/sec	Aphy	Cross-section area	square inches
OPR	Overall Pressure Ratio, Pt3/Pt2				
T41	Turbine Rotor Inlet Temperature	degrees R	**Splitters**		
			BPR	Bypass ratio, secondary flow / primary flow	
Flow Station Data			dPpri/P	Total pressure loss fraction in pri stream	
W	Mass Flow	lbm/sec	dPsec/P	Total pressure loss fraction in sec stream	
Pt	Stagnation pressure	psi			
Tt	Stagnation temperature	degrees R	**Mixers**		
ht	Stagnation enthalpy	BTU/lbm	Aout	Exit area	square inches
FAR	Fuel/Air ratio		PtRatio	Ratio of total pressures, sec stream / pri stream	
Wc	Corrected flow	lbm/sec	MN_I1	Mach number at primary stream entrance	
Ps	Static pressure	psi	MN_I2	Mach number at secondary stream entrance	
Ts	Static temperature	degrees R			
Aphy	Cross-section area	square inches	**Shafts**		
MN	Mach number	BTU/lbm/deg R	Nmech	Shaft mechanical speed	RPM
Rt	Gas constant		pwrIn	Shaft input power	Horsepower
gamt	Ratio of specific heats		HPX	Customer horsepower extraction	Horsepower
Compressors and Turbines			**Burners**		
Wc	Entrance corrected flow	lbm/sec	TtOut	Exit temperature	degrees R
PR	Pressure Ratio		eff	Efficiency	
efPoly	Polytropic effciency		dPqP	Stagnation pressure loss fraction	
eff	Adiabatic efficiency		LHV	Fuel lower heating value	BTU/lbm
Nc	Corrected speed		Wfuel	Fuel flow	lbm/sec
pwr	Power	Horsepower	FAR	Fuel/Air ratio	
Bleeds			**Nozzles**		
Wb/Win	Bleed flow ratio to entrance flow		PR	Pressure Ratio, Pt entrance / Ps exit	
W	Mass Flow	lbm/sec	Cfg	Gross Thrust coefficient	
Tt	Stagnation temperature	degrees R	CdTh	Discharge or Flow coefficient	
ht	Stagnation enthalpy	BTU/lbm	Cv	Velocity coefficient	
Pt	Stagnation pressure	psi	Cang	Angularity coefficient	
			CmixCorr	Mixing effectiveness	
			Ath	Throat area	square inches
			Mnth	Throat Mach number	
			PsExit	Exit static pressure	psi
			Aexit	Exit area	square inches

5.0 Hints & Suggestions

You should first model the baseline engine with the same software that you will use for your new engine design. Your results may not match the generic baseline model exactly but will provide a valid comparison of weights and performance for the new concept.

In general, engines with supersonic capabilities tend to be sized at “top-of-climb” (the beginning of cruise) conditions, rather than at take-off.

The efficiencies of the turbomachinery components may be improved relative to the baseline engine, sufficient justification should be provided.

Installed engine performance accounts for the effects of customer bleed air, customer horsepower extraction as well as inlet pressure recovery, inlet drags, and nozzle or afterbody drags.

Customer bleed air is typically extracted from the high pressure compressor, at a stage providing the desired air pressure, and is used for such purposes as cabin pressurization and air conditioning, wing de-icing, and pressurization of hydraulic reservoirs used for aircraft systems. Customer horsepower extraction is typically extracted from the HP shaft and is used for mechanically-driven aircraft accessories such as generators or alternators, hydraulic pumps, fuel pumps, and oil pumps.

Customer bleed air, customer horsepower extraction and inlet pressure recovery affect the engine match point and therefore the engine airflow at a given flight condition and engine power setting. Inlet and nozzle drags do not affect the engine match point, but do vary with the engine airflow and so are often referred to collectively as throttle-dependent drags. Net thrust computed accounting for customer bleed, horsepower, and inlet recovery but neglecting the throttle-dependent drags is sometimes referred to as “semi-installed” net thrust or sometimes “uninstalled” net thrust. Installed thrust is the semi-installed net thrust minus the throttle-dependent drags:

$$F_{net,installed} = F_{net,semi-installed} - D_{inlet} - D_{nozzle}$$

Computation of inlet recovery and inlet drags requires book-keeping the mass flows and their associated losses throughout the inlet from the freestream to the engine face. Book-keeping nomenclature for a typical mixed-compression inlet is shown in Figure 6.

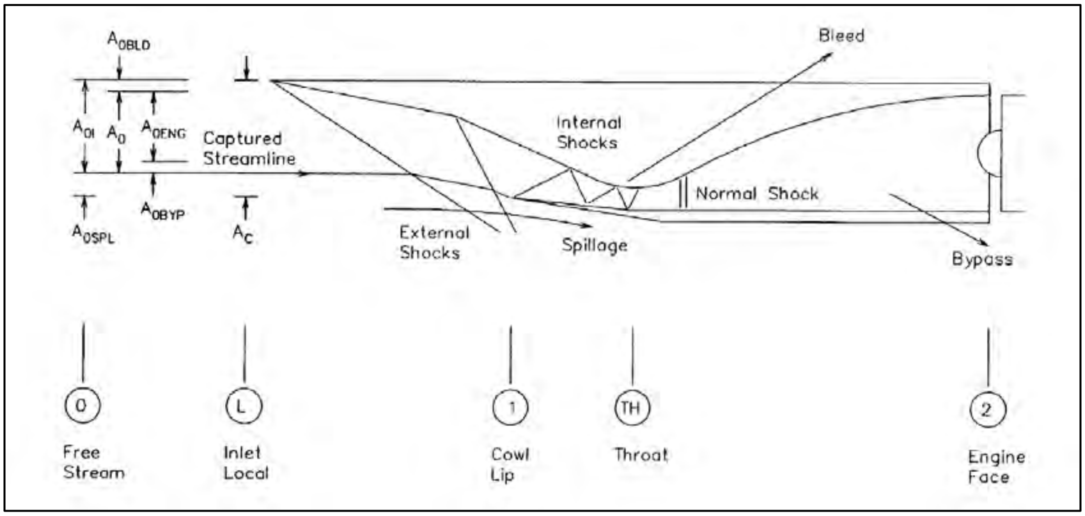

Figure 6. Inlet Airflow Accounting (Barnhart, 1997)

Terminology is defined in Table 11.

Table 11. Inlet Airflow Accounting Terminology

$\boldsymbol{A_C}$	Capture area (projected cowl lip area)
$\boldsymbol{A_0}$	Inlet air demand passing through the inlet throat
$\boldsymbol{A_{0BLD}}$ or $\boldsymbol{A_{0BLC}}$	Inlet boundary layer control bleed air
$\boldsymbol{A_{0BYP}}$	Inlet-engine bypass air
$\boldsymbol{A_{0ENG}}$	Engine demand airflow
$\boldsymbol{A_{0I}}$	Inlet air demand in the freestream
$\boldsymbol{A_{0SPL}}$	Inlet spillage

It is clear from the figure that:

$$A_C = A_{0I} + A_{0SPL}$$
$$A_{0I} = A_0 + A_{0BLD}$$
$$A_0 = A_{0ENG} + A_{0BYP}$$

Inlet drags include spillage, bleed, and bypass drags. The inlet will “capture” an amount of freestream air based on its projected cowl lip area, A_C , also called the inlet capture area. Air not needed by the engine will be “spilled” or diverted around the inlet lip. Spillage drag comprises the effects of the momentum change in this air, the additive drag, and cowl lip suction. Bleed air may be extracted upstream of the inlet throat and dumped overboard for purposes of inlet stability or boundary layer control. Bypass air may be extracted downstream of the inlet throat and dumped overboard for purposes of engine/inlet airflow matching. Inlet bleed and bypass air which is dumped overboard also create drag penalties due to the momentum change of the air.

A systematic procedure for inlet performance characterization has been developed by Ball and Hickox (1978) and extended by Kowalski (1979) and others. In this procedure, the relative amounts of engine, spillage, bleed, and bypass air flows are described in terms of their equivalent freestream tube areas divided by the inlet capture area. Empirical performance curves or maps of standardized format are developed in terms of these area ratios. Example performance maps are shown in Figure 7. It may be seen that a total of 14 maps are required to completely characterize the inlet performance.

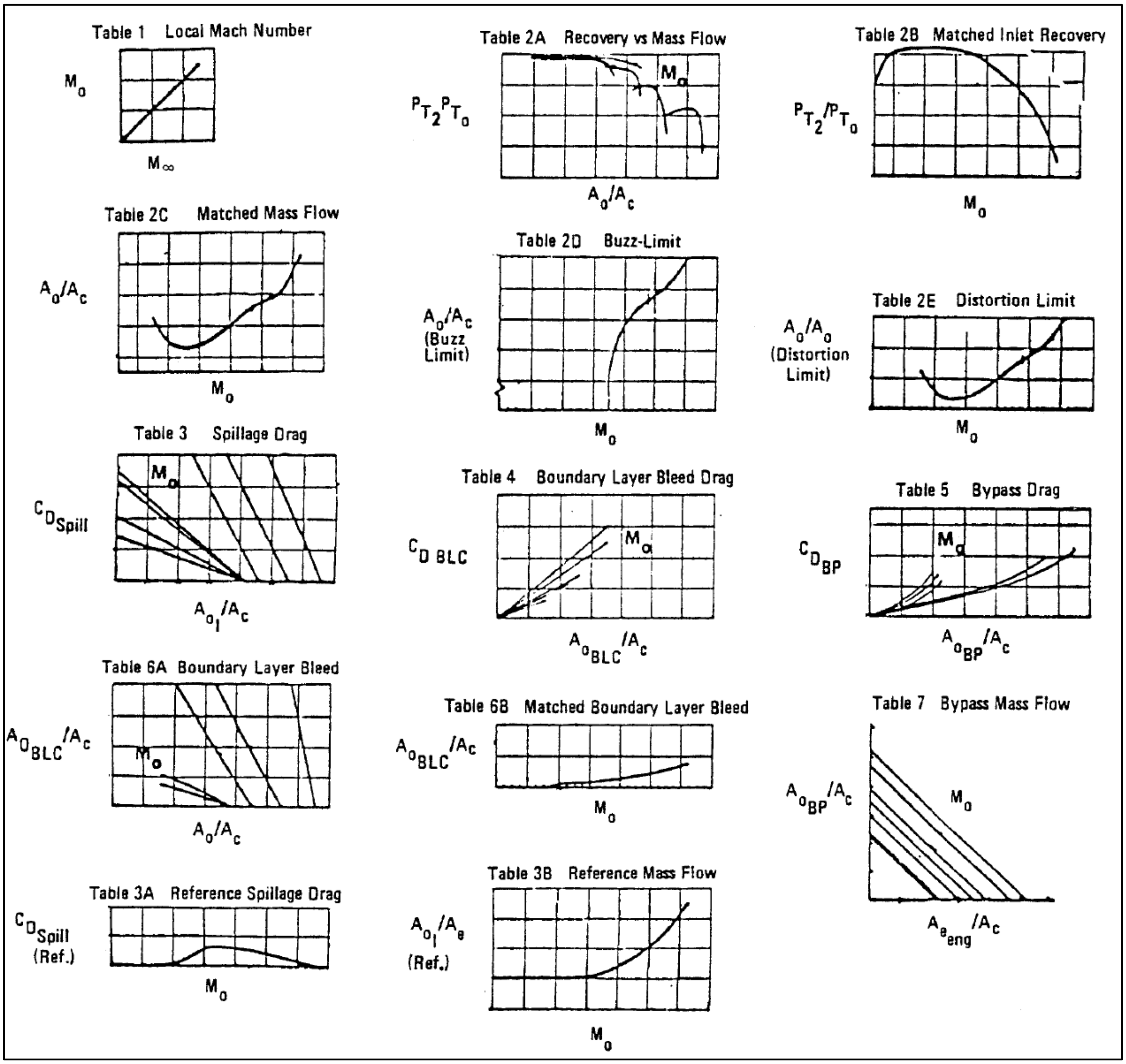

Figure 7. Example Inlet Performance Maps (Tjonneland, 1986)

The mass flow delivered at the engine face depends upon the inlet recovery and the inlet bleed, bypass, and spillage flows. These factors are a function of the capture area, which must be sized appropriately. Because of the interdependence of the flows, an iterative procedure is required. As implemented in NPSS, the engine mass flow W_{ENG} is added as an independent variable to the Newton-Raphson solver. The corresponding solver dependent variable requires that the inlet component entrance flow equals the calculated inlet demand flow W_I. To clarify calculation procedures for both on- and off-design are summarized below.

The inlet capture area may be sized at the design condition using the "matched" performance maps, Tables 2B and 2C. At the matched condition, the inlet is appropriately sized and provides the maximum pressure recovery with no spillage drag and no bypass drag. The design sizing calculation is as follows:

1. Read the matched (design) mass flow ratio $(A_0/A_C)_{des}$ from Table 2C.
2. Compute the design flow area using $A_{0,des} = \frac{W_{ENG}}{\rho V}$. This ensures that there is no bypass air at the design point.
3. Compute the required capture area as $A_C = \frac{A_{0,des}}{(A_0/A_C)_{des}}$
4. Read the matched inlet recovery from Table 2B.
5. Read the matched bleed flow ratio $A_{0BLD}/A_C$from Table 6B.
6. Compute the inlet demand flow using $W_{0I} = \rho V \left(\frac{A_0}{A_C} + \frac{A_{0BLD}}{A_C}\right) A_C$

Once the capture area is known, the inlet off-design performance may be calculated as follows. Note that frequent use is made of the fact that the area ratios and their corresponding mass flow ratios are equal.

1. Calculate the capture mass flow using $W_C = \rho V A_C$
2. Calculate the engine area ratio using $\frac{A_{0ENG}}{A_C} = \frac{W_{ENG}}{W_C}$
3. Read the bypass area ratio from Table 7
4. Calculate the bypass mass flow using $W_{BYP} = \frac{A_{0BYP}}{A_C} W_C$
5. Calculate the mass flow at the inlet throat $W_0 = W_{ENG} + W_{BYP}$
6. Calculate the area ratio using $\frac{A_0}{A_C} = \frac{W_0}{W_C}$
7. Read the inlet recovery from Table 2A
8. Read the bleed area ratio from Table 6
9. Calculate the bleed mass flow using $W_{BLD} = \frac{A_{0BLD}}{A_C} W_C$
10. Calculate the inlet demand mass flow $W_I = W_0 + W_{BLD}$
11. Calculate the inlet demand area ratio using $\frac{A_{0I}}{A_C} = \frac{W_I}{W_C}$
12. Read the coefficients of spillage drag, bleed drag, and bypass drag from Tables 3, 4, and 5, respectively.
13. Compute the inlet drag $D_{inlet} = \left(C_{d,spill} + C_{d,bleed} + C_{d,bypass}\right) \frac{1}{2} \rho V^2 A_C$

The internal performance of the exhaust nozzle is characterized by the gross thrust coefficient, which accounts for the effects of friction, angularity, and under- or over-expansion. An example gross thrust coefficient curve is presented in Figure 8.

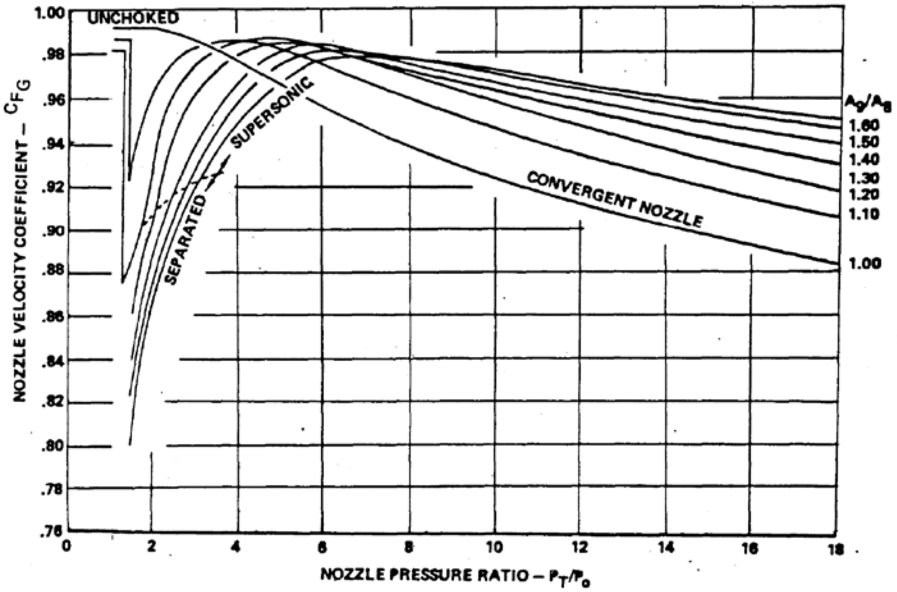

Figure 8. Nozzle Gross Thrust Coefficient (Kowalski, 1979)

Note that the gross thrust coefficient is a function of nozzle pressure ratio as well as the nozzle area ratio, A_9/A_8. In many performance models, including NPSS, the nozzle exit area computed by the model corresponds to the area required for full expansion to the ambient pressure, and does not reflect the true geometric A_9. The user must provide the correct A_9 for determination of the gross thrust coefficient. In many cases the nozzle area ratio A_9/A_8 is variable and may be scheduled by the engine control. An example A_9/A_8 schedule is shown in Figure 9.

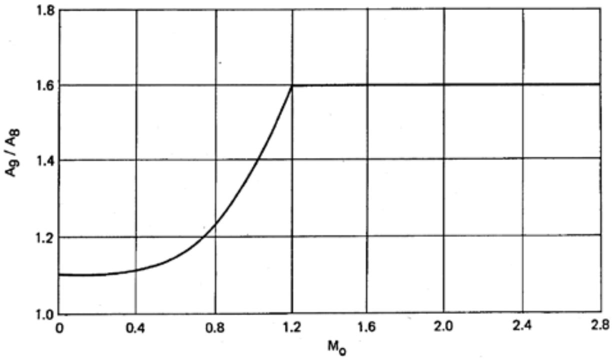

Figure 9. Nozzle Area Ratio Schedule (Kowalski, 1979)

For a mixed-flow exhaust, an additional correction to the gross thrust calculation is required to account for the effects of non-uniform temperature distribution due to incomplete mixing. This phenomenon is discussed by Frost (Frost, 1966).

Nozzle afterbody drag results from shear forces (friction), pressure drag, and shock losses. It is a primarily a function of freestream Mach number and nozzle boat-tail curvature. An example nozzle drag curve is shown in Figure 10.

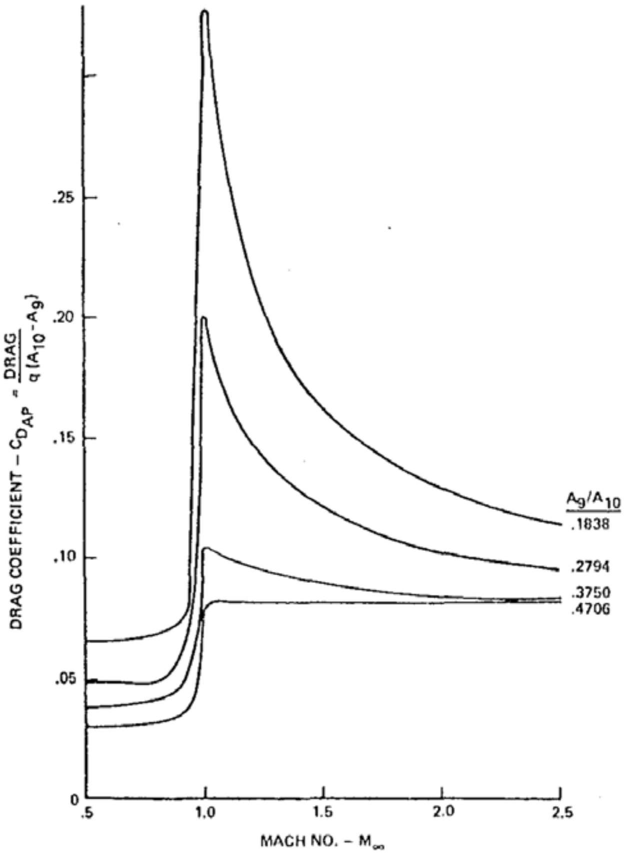

Figure 10. Nozzle Afterbody Drag (Decher, 1978)

In this example, the drag coefficient is presented as a function of nozzle exit area A_9 and the upstream maximum cross-sectional area A_{10}. Other curves may include the nozzle boat-tail angle as a parameter.

The specific installation performance curves used to generate the installed performance requirements for this RFP are provided in the following figures. Please note that for the inlet recovery curve, Table 2A, the x-axis is not A_0/A_C but rather $(A_0/A_C)/eRam$. This modification prevents the vertical lines in the Table (multi-valued solutions) which would create convergence problems in NPSS. However, an additional solver pair is required to iterate on the recovery.

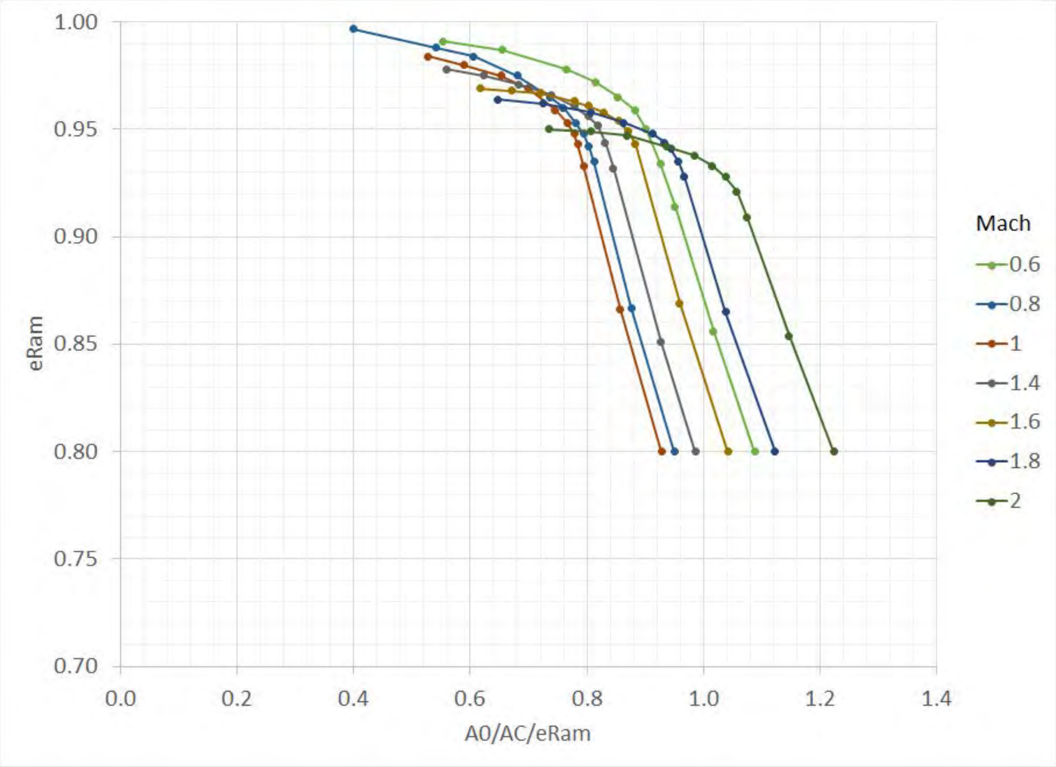

Figure 11. Table 2A Inlet Recovery

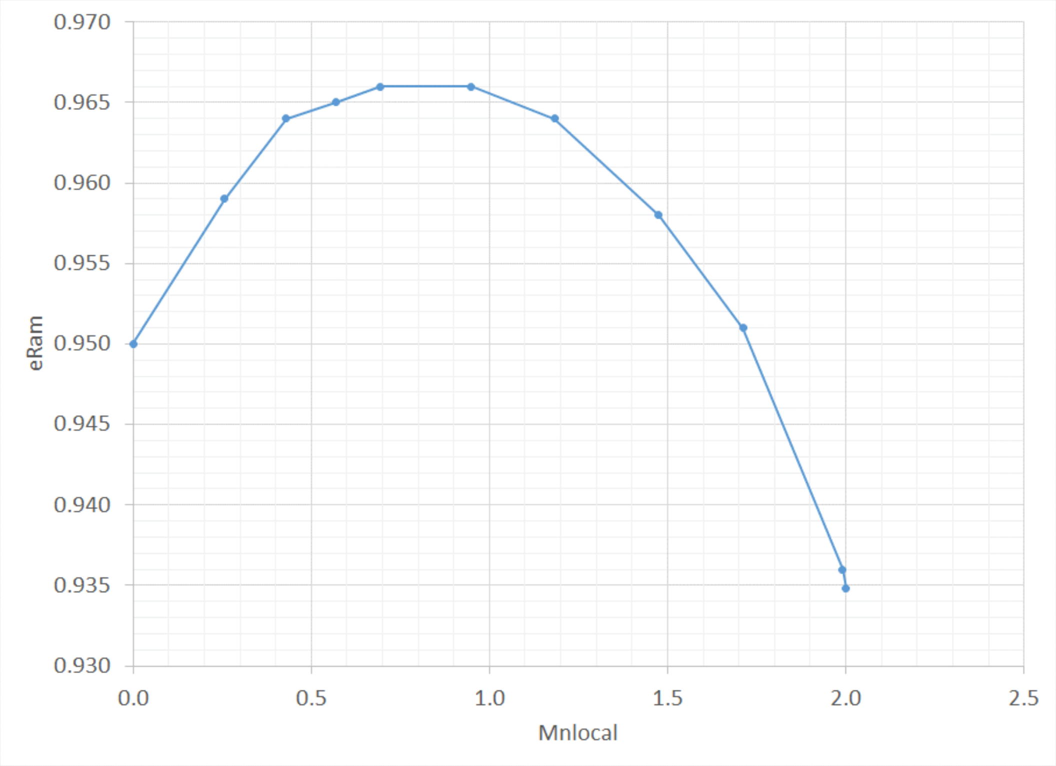

Figure 12. Table 2B Matched Inlet Recovery

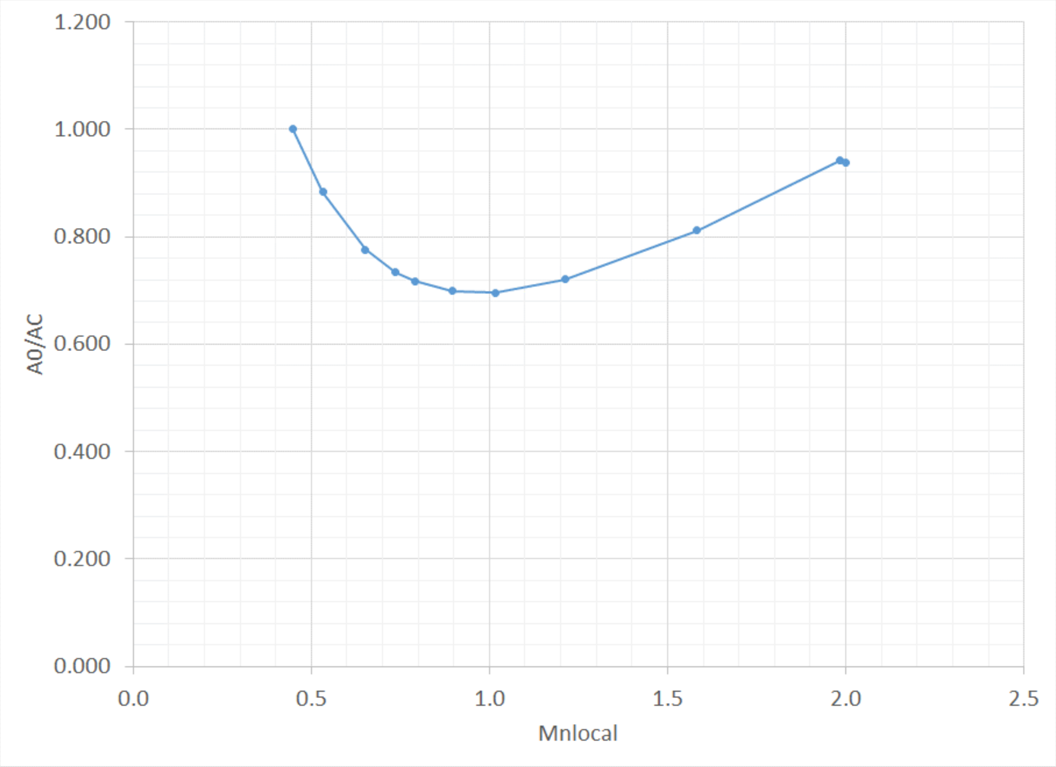

Figure 13. Table 2C Matched Inlet Mass Flow Ratio

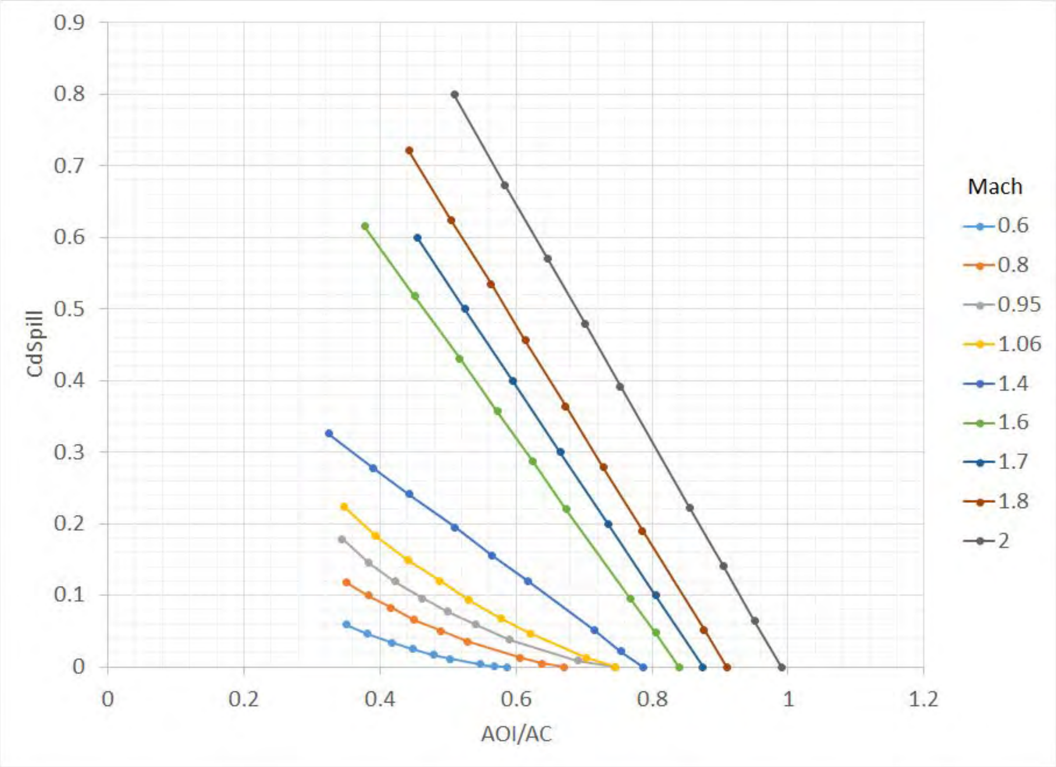

Figure 14. Table 3 Spillage Drag Coefficient

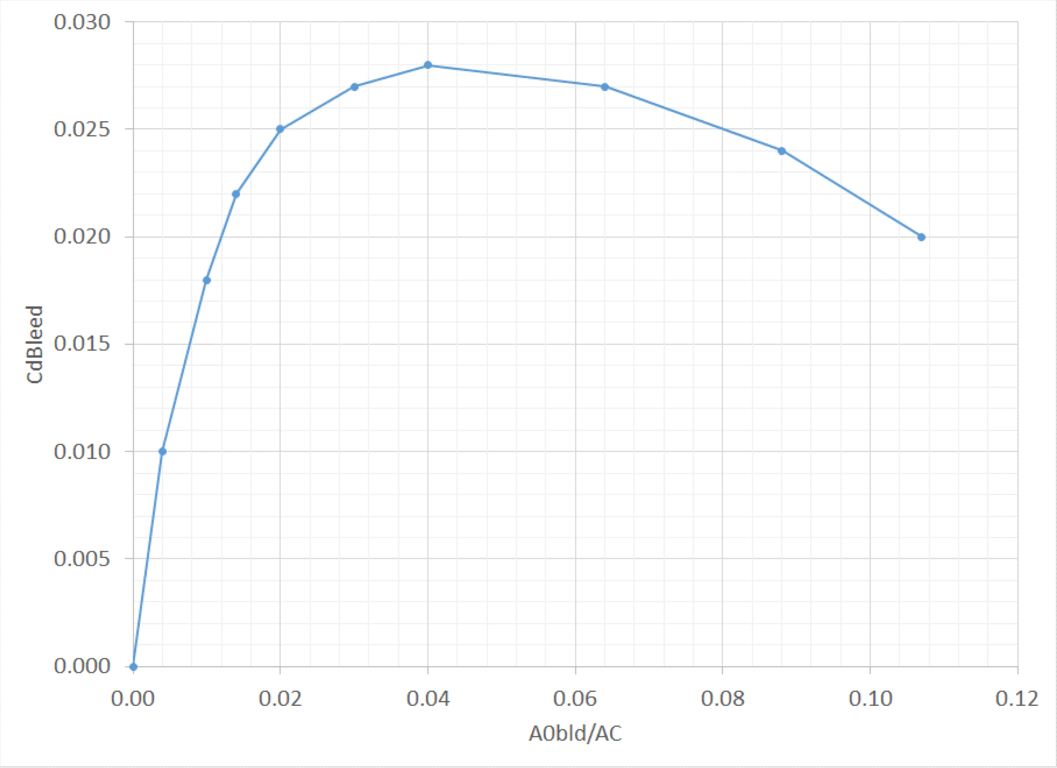

Figure 15. Table 4 Inlet Bleed Drag Coefficient

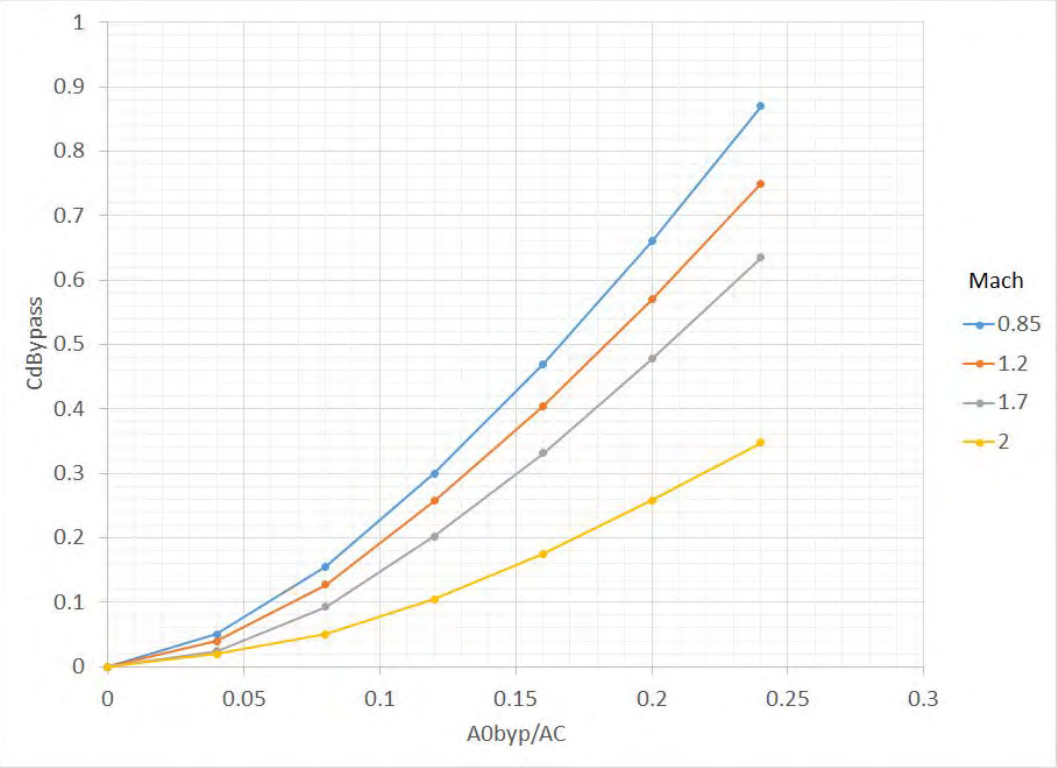

Figure 16. Table 5 Inlet Bypass Drag Coefficient

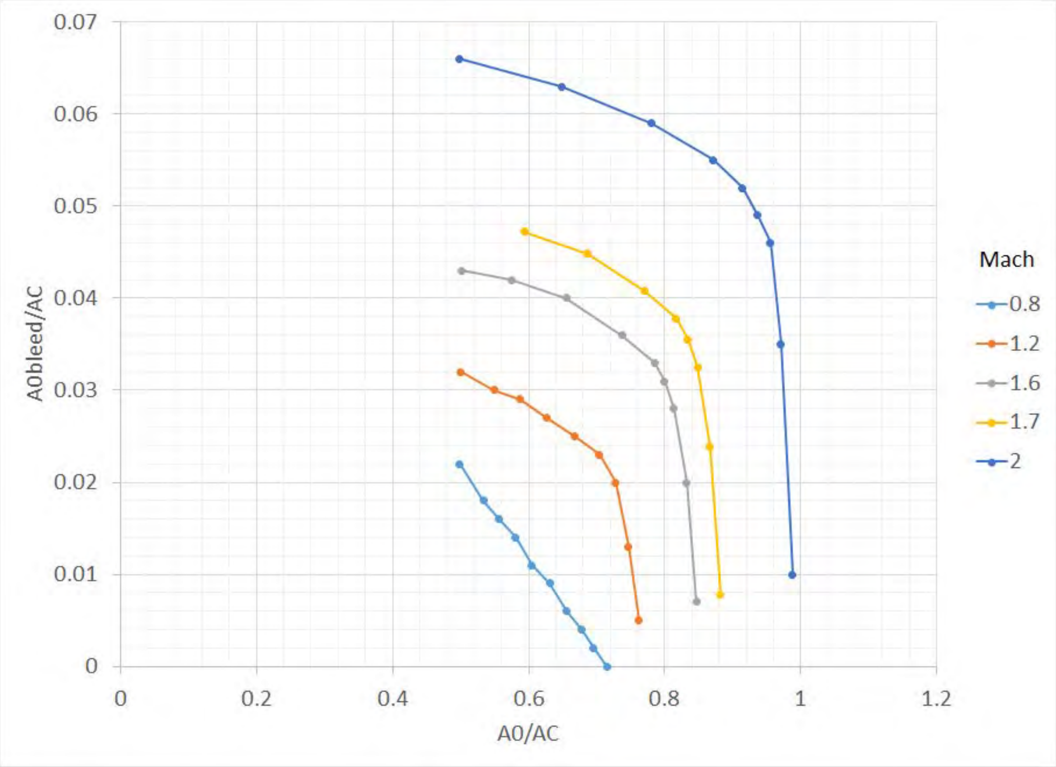

Figure 17. Table 6A Inlet Bleed Mass Flow Ratio

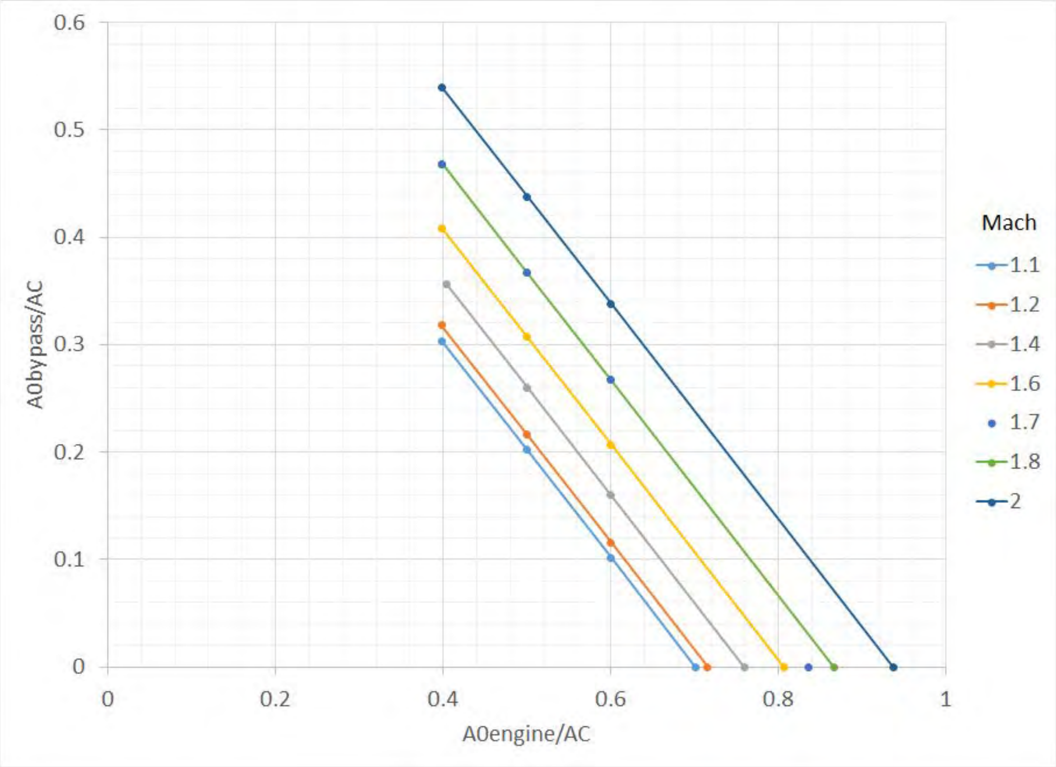

Figure 18. Table 7 Inlet Bypass Mass Flow Ratio

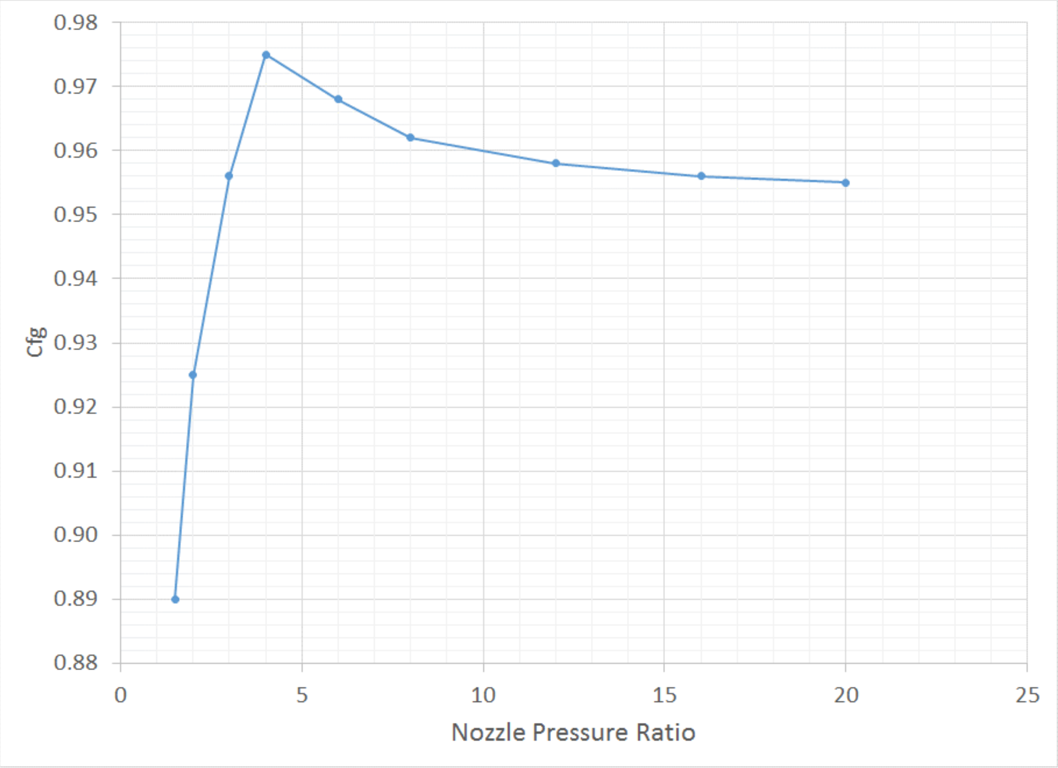

Figure 19. Nozzle Gross Thrust Coefficient

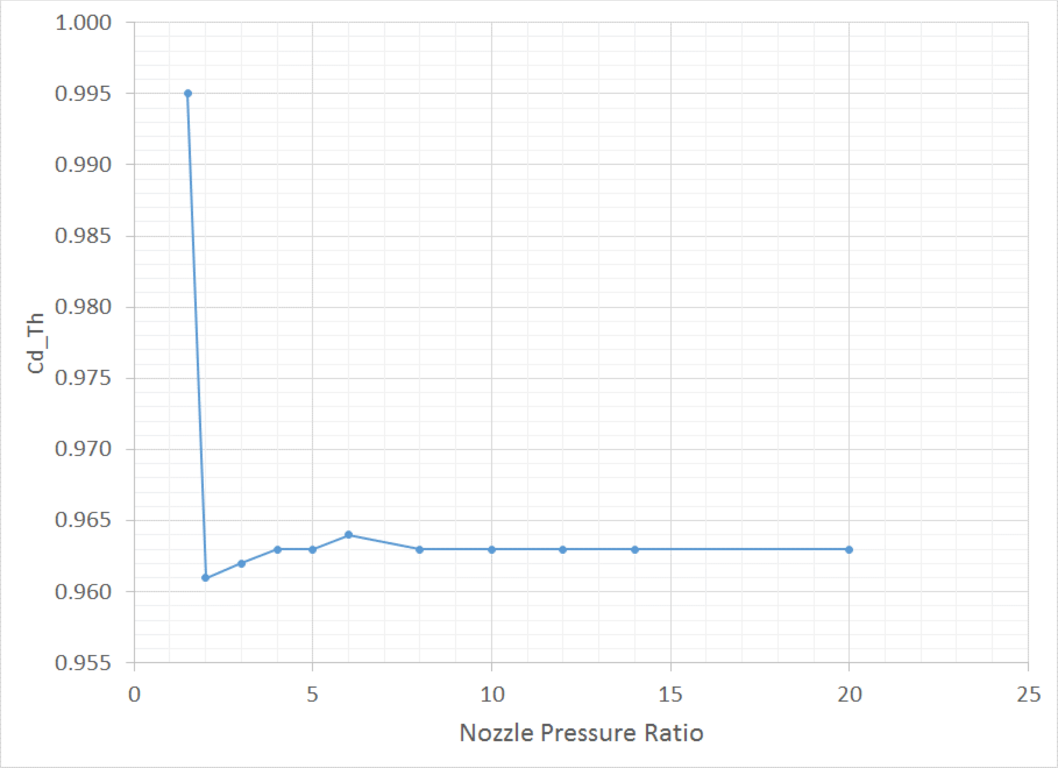

Figure 20. Nozzle Discharge Coefficient

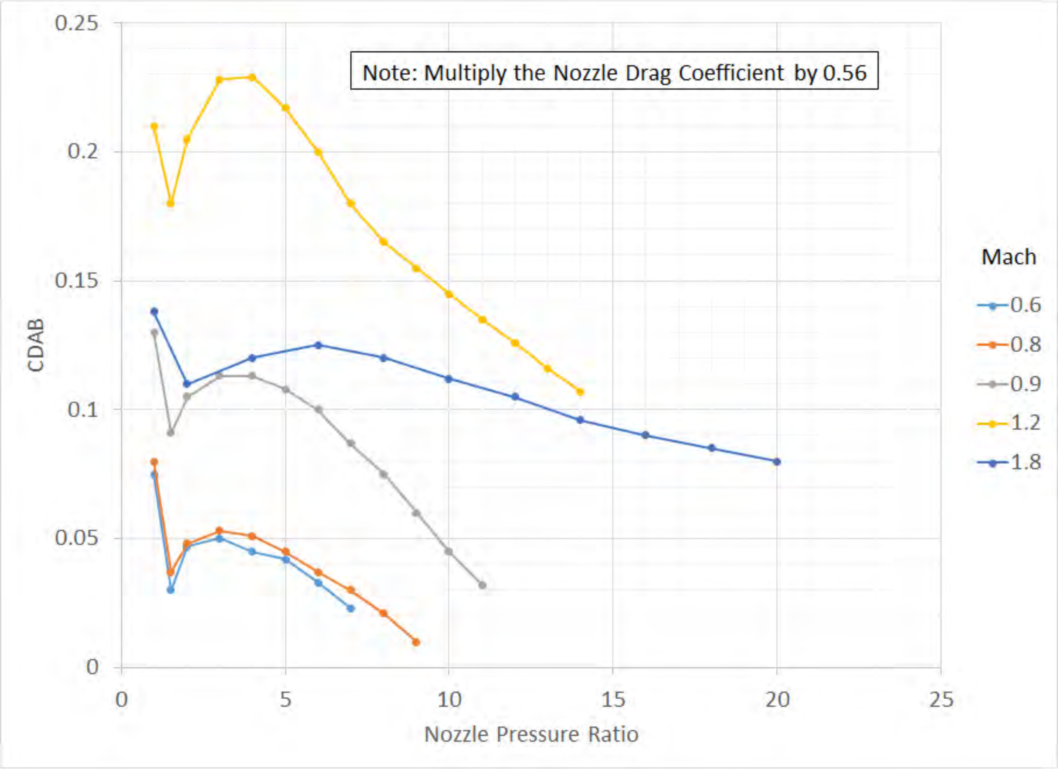

Figure 21. Nozzle Drag Coefficient

6.0 Competition Expectations

The existing rules and guidelines for the *AIAA Foundation Student Design Competition* should be observed and these are provided in *Appendix*. In addition, the following specific suggestions are offered for the event.

This is a preliminary engine design. It is not expected that student teams produce design solutions of industrial quality, however it is hoped that attention will be paid to the practical difficulties encountered in a real-world design situation and that these will be recognized and acknowledged. If such difficulties can be resolved quantitatively, appropriate credit will be given. If suitable design tools and/or knowledge are not available, then a qualitative description of an approach to address the issues is quite acceptable.

In a preliminary engine design the following features must be provided:

- Definition and justification of the mission and the critical mission point(s) that drive the candidate propulsion system design(s).
- Completion of the compliance matrices and required trade studies listed on Table 6, Table 7, and Table 8, including but not limited to:
    - Clear and concise demonstration that the overall engine performance satisfies the mission requirements.
    - Documentation of the trade studies conducted to determine the preferred engine cycle parameters such as fan pressure ratio, bypass ratio, overall pressure ratio, turbine inlet temperature, etc.

- An engine configuration with a plot of the flow path that shows how the major components fit together, with emphasis on operability at different mission points.
- A clear demonstration of design feasibility, with attention having been paid to technology limits. Examples of some, but not all, velocity diagrams are important to demonstrate viability of turbomachinery components.
- Stage count estimates, again, with attention having been paid to technology limits.
- Estimates of component performance and overall engine performance to show that the assumptions made in the cycle have been achieved.

While only the preliminary design of major components in the engine flow path is expected to be addressed quantitatively in the proposals, it is intended that the role of secondary systems such as fuel & lubrication be given serious consideration in terms of modifications and how they would be integrated in to the new engine design. Credit will be given for clear descriptions of how any appropriate upgrades would be incorporated and how they would affect the engine cycle.

This is not an aircraft design competition, so credit will not be given for derivation of aircraft flight characteristics.

The use of design codes from industrial or government contacts, that are not accessible to all competitors, is not allowed.

Each proposal should contain a brief discussion of any computer codes or *Microsoft Excel* spreadsheets used to perform engine design & analysis, with emphasis on any additional special features generated by the team.

Proposals should be limited to fifty pages, which will not include the administrative/ contents or the “signature” pages.

7.0 References

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7. Kowalski, Edward J., A Computer Code for Estimating Installed Performance of Aircraft Gas Turbine Engines, Vols. 1, 2, and 3, NASA Contractor Reports CR-159691, 159692, and 159693, 1979

8. Norman, P.D.; Lister, D.H.; Lecht, M.; Madden, P.; Park, K.; Penanhoat, O.; Plaisance, C.; Renger, K.: NEPAIR Final Technical Report. NEPAIR/WP4/WPR/01, G4RD-CT-2000-00182, 2003
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11. Wesocky and Prather, "Atmospheric Effects Of Stratospheric Aircraft: A Status Report From NASA's High-Speed Research Program," Proceedings Of Tenth International Symposium On Air Breathing Engines, September 1991

8.0 Suggested Reading

1. "*Gas Turbine Theory*"**,** H.I.H Saravanamuttoo, G.F.C Rogers &.H. Cohen, Prentice Hall, 5th Edition 2001.
2. "*Aircraft Engine Design*", J.D. Mattingly, W.H. Heiser, & D.H. Daley, AIAA Education Series, 1987.
3. "*Elements of Propulsion – Gas Turbines and Rockets*", J.D. Mattingly, AIAA Education Series, 2006.
4. "*Jet Propulsion*", N. Cumpsty, Cambridge University Press, 2000.
5. "*Gas Turbine Performance*", P. Walsh & P. Fletcher, Blackwell/ASME Press, 2nd Edition, 2004.
6. "*Fundamentals of Jet Propulsion with Applications*", Ronald D. Flack, Cambridge University Press, 2005.
7. "*The Jet Engine*", Rolls-Royce plc. 2005.
8. "*Mechanics and Thermodynamics of Propulsion*", Hill, Philip G. and Peterson Carl R., Addison-Wesley Publishing Company, Reading, Massachusetts, 1965.

9.0 Allowable and Available Software & Additional Reference Material

Students may use the following approved cycle analysis and design codes:

- Student-developed codes written specifically for this project (i.e., Excel or Matlab)
- GasTurb 12 (http://www.gasturb.de/)

  GasTurb 12 is a comprehensive code for the preliminary design of propulsion and industrial gas turbine engines. It encompasses design point and off-design performance, based on extensive libraries of engine architectures and component performance maps, all coupled to impressive graphics. A materials database and plotting capabilities enable a detailed engine model to be generated, with stressed disks and component weights. A student license for this code is_available at a very low price directly from the author (*Reference 3*) strictly for academic work only.
- NLR's GSP http://gspteam.com/
- NPSS Learning Edition

http://www.wolverine-ventures.com/index.php?page=shop.browse&category_id=3&option=com_virtuemart&Itemid=18&vmcchk=1&Itemid=18

- AxSTREAM by SoftInWay Inc.

  AxSTREAM is a design and analysis code that permits the topic of propulsion and power generation by gas and steam turbine to progress beyond velocity diagrams in the course of university class. A suite of compressor and turbine modules cover the design steps from meanline and streamline solutions to detailed design of airfoils. Use of this code is also supported fully by excellent graphics. SoftInWay Inc. recently announced the availability of AxSTREAM Lite to students that covers the design of turbines. However, an expanded license will be provided to participants in the Joint AIAA–IGTI Undergraduate Team Engine Design Competition that also includes fans and compressors for an appropriate time period prior to submission of proposals.

  How to obtain AxSTREAM (expanded license):

  Once a Letter of Intent has been received by AIAA, the names of team members will be recognized as being eligible to be granted access to the AxSTREAM software. Students must then apply to SoftInWay Inc. SoftInWay will not contact team members.

Design Competition Rules

General Rules

- All AIAA Student members are eligible and encouraged to participate. Membership with AIAA must be current to submit a report and to receive any prizes.
- Students must submit their letter of intent and final report via the online submission system before on the posted deadlines to be eligible to participate. No extensions will be granted.
- More than one design may be submitted from students at any one school.
- If a design group withdraws their final report from the competition, the team leader must notify AIAA Headquarters immediately.
- Design projects that are used as part of an organized classroom requirement are eligible and encouraged for competition.

Categories/Submissions

- Team Submissions
    - Team competitions will be groups of not more than ten AIAA Student Members per entry.
- Individual Submissions
    - Individual competitions will consist of only one AIAA Student member per entry.
- Graduate
    - Graduate students may participate in the graduate categories only.
- Undergraduate
    - Undergraduate students may participate in the undergraduate categories only.
- Letter of Intent (LOI)
    - A Letter of Intent indicating interest in participating in the design competitions is required before submitting a final report.
    - All Letters of Intent must be submitted through the online submission system.
    - Letter of Intent must include student’s names, emails, AIAA membership numbers, faculty advisor(s) names, emails, and project advisor(s) names and emails. Any LOI that is not completed will be ineligible to submit a final report.
- Final Report
    - An electronic copy of the report in Adobe PDF format must be submitted to AIAA using the online submission site. Total size of the file cannot exceed 20 MB.
    - A “Signature” page must be included in the report and indicate all participants, including faculty and project advisors, along with students’ AIAA member numbers and signatures.

    - Each report should be no more than 100 pages, double-spaced (including graphs, drawings, photographs, and appendices) if it were to be printed o 8.5”x11.0” paper, and the font should be no smaller than 10 pt. Times New Roman.

    - Engine Design Competition is limited to 50 pages.

Copyright

All submissions to the competition shall be the original work of the team members.
Any submission that does not contain a copyright notice shall become the property of AIAA. A team desiring to maintain copyright ownership may so indicate on the signature page but nevertheless, by submitting a proposal, grants an irrevocable license to AIAA to copy, display, publish, and distribute the work and to use it for all of AIAA's current and future print and electronic uses (e.g. "Copyright © 20__ by _____. Published by the American Institute of Aeronautics and Astronautics, Inc., with permission.).
Any submission purporting to limit or deny AIAA licensure (or copyright) will not be eligible for prizes.

Conflict of Interest

It should be noted that it shall be considered a conflict of interest for a design professor to write or assist in writing RFPs and/or judging proposals submitted if (s)he will have students participating in, or that can be expected to participate in those competitions. A design professor with such a conflict must refrain from participating in the development of such competition RFPs and/or judging any proposals submitted in such competitions.

Schedule

- Letter of Intent — 10 February 2018 (11:59 pm Eastern Time)
- Proposal delivered to AIAA Headquarters — 10 May 2018 (11:59 pm Eastern Time)
- Announcement of Winners — 31 August 2018 (11:59 pm Eastern Time)
    - Engine Design Competition dates
        - Letter of Intent – 14 February 2018 (11:59 pm Eastern Time)
        - Proposal delivered to AIAA Headquarters – 16 May 2018 (11:59 pm Eastern Time)
        - Round 1 evaluations completed – 30 June 2018 (11:59 pm Eastern Time)
        - Round 2 presentations at AIAA Propulsion and Energy Forum 2018

Awards

The prize money provided for the competitions is funded through the AIAA Foundation. The monetary awards may differ for each competition, with a maximum award of $500. The award amounts are listed below.

The top three design teams will be awarded certificates. One representative from the first place team *may be* invited by the Technical Committee responsible for the RFP to make a presentation of their design at an AIAA forum. A travel stipend *may be* available for some competitions, with a maximum travel stipend of $750 which may be used to help with costs for flight, hotel, or conference registration to attend an AIAA forum.

Aircraft Design Competitions

- Graduate Team Aircraft - Advanced Pilot Training Aircraft
- Undergraduate Team Aircraft – Hybrid-Electric General Aviation Aircraft (HEGAA)

  - 1st Place: \$500; 2nd Place: \$250; 3rd Place \$125
- Undergraduate Individual Aircraft – Close Air Support Aircraft (A-10 Replacement)

Engine Design Competition

- Undergraduate Team Engine –Candidate Engines for a Next Generation Supersonic Transport
  - 1st Place: \$500; 2nd Place: \$250; 3rd Place \$125

Space Transportation Competition

- Undergraduate Team Space Transportation – Pluto Orbiter
  - 1st Place: \$500; 2nd Place: \$250; 3rd Place \$125

Space Design Competition

- Undergraduate Team Space Design – Lunar Prospecting
  - 1st Place: \$500; 2nd Place: \$250; 3rd Place \$125

Structures Design Competition

- Graduate Team Structures – Fuselage Design
- Undergraduate Team Structures – Supersonic Wing
  - 1st Place: \$500; 2nd Place: \$250; 3rd Place \$125

Proposal Requirements

The technical proposal is the most important factor in the award of a contract. It should be specific and complete. While it is realized that all of the technical factors cannot be included in advance, the following should be included and keyed accordingly:

- Demonstrate a thorough understanding of the Request for Proposal (RFP) requirements.
- Describe the proposed technical approaches to comply with each of the requirements specified in the RFP, including phasing of tasks. Legibility, clarity, and completeness of the technical approach are primary factors in evaluation of the proposals.
- Particular emphasis should be directed at identification of critical, technical problem areas. Descriptions, sketches, drawings, systems analysis, method of attack, and discussions of new techniques should be presented in sufficient detail to permit engineering evaluation of the proposal. Exceptions to proposed technical requirements should be identified and explained.
- Include tradeoff studies performed to arrive at the final design.
- Provide a description of automated design tools used to develop the design.

Basis for Judging

1. Technical Content (35 points)

This concerns the correctness of theory, validity of reasoning used, apparent understanding and grasp of the subject, etc. Are all major factors considered and a reasonably accurate evaluation of these factors presented?

2. Organization and Presentation (20 points)

The description of the design as an instrument of communication is a strong factor on judging. Organization of written design, clarity, and inclusion of pertinent information are major factors.

3. Originality (20 points)

The design proposal should avoid standard textbook information, and should show the independence of thinking or a fresh approach to the project. Does the method and treatment of the problem show imagination? Does the method show an adaptation or creation of automated design tools?

4. Practical Application and Feasibility (25 points)

The proposal should present conclusions or recommendations that are feasible and practical, and not merely lead the evaluators into further difficult or insolvable problems.