AIAA Foundation Student Design Competition 2020/21
Undergraduate Team – Engine

Let's Re-Engine the Concorde!

- Request for Proposal -

September 3, 2020

Sponsored by npss

Abstract

In the 1960s, the *British Aerospace/Aerospatiale Concorde* advanced commercial aviation immensely when it made supersonic travel a reality, using four *Rolls-Royce/SNECMA Olympus 593* engines. However, *Concorde* was neither a commercial nor environmental success because of its high fuel consumption, excessive noise at take-off and its high fares. It is easy to wonder what could have been if current tools and technology were applied to that same airframe. So let us address that!

Here we ask for proposals to replace the *Olympus 593* turbojet with modern low bypass ratio turbofans with an entry-into-service date of 2028. Reheat at take-off is to be eliminated, if possible. It is hoped to extend the range by reducing fuel consumption and minimizing engine mass.

A generic model of the baseline *Olympus 593* is supplied and this must be replicated for comparison of your new engine. The primary design point for the proposed engine should be supersonic cruise conditions at 53,000 feet/Mach 2 (ISA +5°C), where the net thrust must be 10,000 lbf. A second "off-design" point should be rolling take-off at sea level/Mach 0.3 (ISA +10°C), where the net thrust must be 33,600 lbf.

The performance and total fuel consumption of the candidate engine should be estimated over a typical mission, stated clearly in the proposal, and compared with those of the *Olympus 593*. Attention should be given to technical feasibility and integration with the *Concorde* airframe.

Dr. Ian Halliwell
AIAA Air Breathing Propulsion Group
Principal – NORTHWIND PROPULSION INC
E-mail: ianhalliwell@earthlink.net

CONTENTS

1. Introduction

1.1 The Aircraft

Figure 1.1: Concorde at Supersonic Cruise

In the 1960s, the *British Aerospace/Aerospatiale Concorde* represented a major milestone in commercial aviation by halving travel times between Singapore and Melbourne and setting world records as it crossed the Atlantic between London and Gander four times in a day (*Reference 1*). However, it is recognized that *Concorde* was neither a commercial nor environmental success. Between Paris and New York, the 25,000 lbm payload was only 6% of the all-up weight, with 31% being structure and airframe, and 63% being powerplant and fuel. (*Reference 2*). Owing to the high fuel burn and noise at take-off, the impact of environmental pollution was clearly intolerable. These factors, combined with high fares, meant that significant growth of SST fleets was never going to happen. Nevertheless, at the time, the degree of technical achievement was immense, especially with the tools available – area rule, slender body theory, and wind tunnels, supported only by very rudimentary design tools in the form of slide rules and thermionic valve computers (the IBM 7040). All of us, from students to seasoned professionals, currently have tools with vastly more speed and capability at our disposal and we also have the benefit of the lessons learned by the engineers who designed and built the *Concorde* and its engines. With that in mind, I wondered what we could accomplish today, if we left the aircraft as it is - even though we know we could improve its aerodynamics (L/D = 7.4 at cruise, L/D = 4.0 at take-off.) - and redesigned the engines.

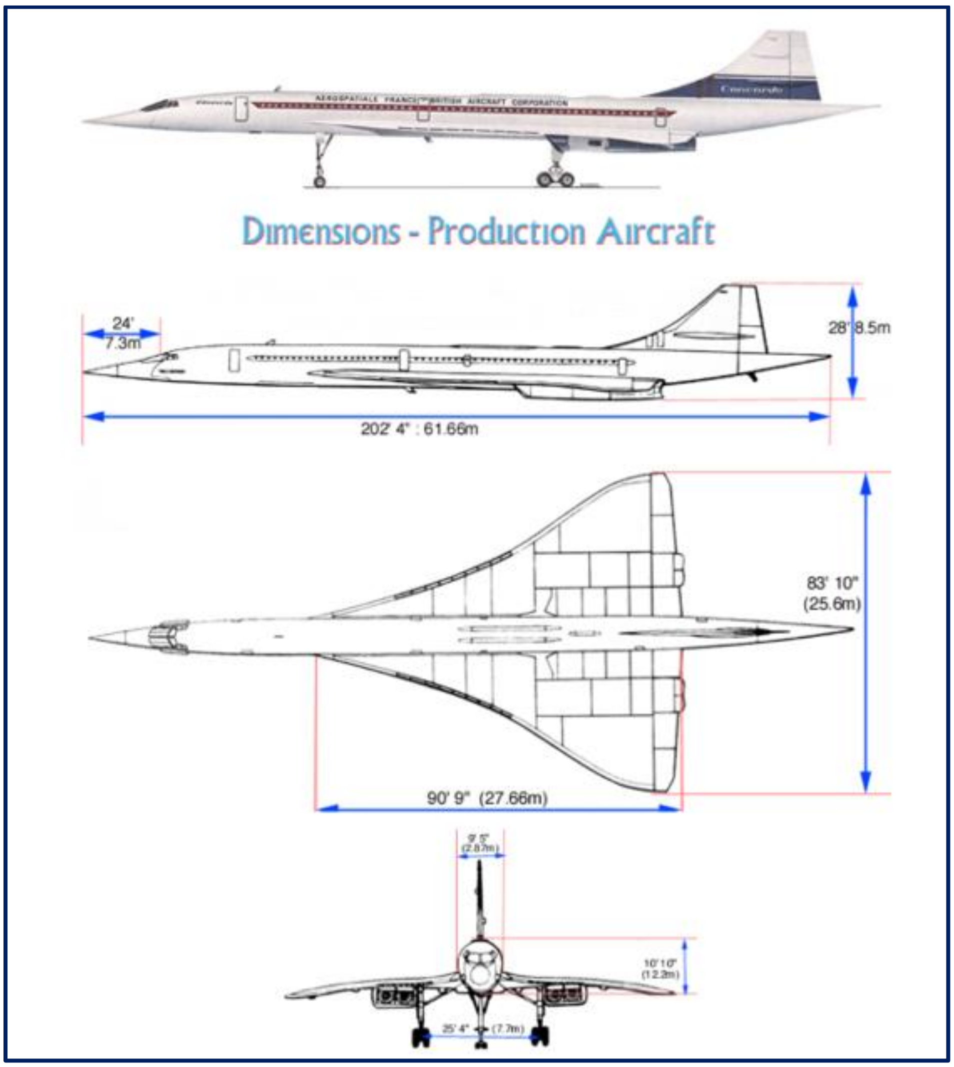

Figure 1.2: Concorde Dimensions

1.2 The Engines

An abbreviated history of the development of the engine that eventually powered *Concorde* is shown in *Figure 1.3*. This culminated in the *Rolls-Royce/SNECMA Olympus 593 Mk 610*.

THE CONCORDE 593 ENGINE VARIANTS

- 593 – Original version designed for ConcordeThrust : 20,000 lbf (89 kN) dry / 30,610 lbf (136 kN) reheat
- 593-22R – Powerplant fitted to prototypes. Higher performance than original engine due to changes in aircraft specification.Thrust : 34,650 lbf (154 kN) dry / 37,180 lbf (165 kN) reheat
- 593-610-14-28 – Final version fitted to production ConcordeThrust : 32,000 lbf (142 kN) dry / 38,050 lbf (169 kN) reheat

Specifications (Olympus 593 Mk 610)

General characteristics

- Type: Turbojet
- Length: 4039 mm (159 in)
- Diameter: 1212 mm (47.75 in)
- Dry weight: 3175 kg (7,000 lb)

Components

- Compressor: Axial flow, 7-stage low pressure, 7-stage high pressure
- Combustors: Nickel alloy construction annular chamber, 16 vaporising burners, each with twin outlets
- Turbine: High pressure single stage, low pressure single stage
- Fuel type: Jet A1

Performance

- Maximum Thrust: wet: 169.2 kN (38,050 lbf) dry: 139.4 kN (31,350 lbf)
- Overall pressure ratio: 15.5:1
- Specific fuel consumption: 1.195 (cruise), 1.39 (SL) lb/(h·lbf)
- Thrust-to-weight ratio: 5.4

Control system

- World's first FADEC control system

Jetpipe

- Straight pipe with pneumatically operated convergent nozzle
- Single ring afterburner
- 'Eyelids' which act as variable divergent nozzles/thrust reversers

Figure 1.3: Variations of the Olympus 593

Some details of specific interest to us are reformatted in *Table 1*.1.

Engine Model	Olympus 593 Mk 610 turbojet
Manufacturer	Rolls-Royce/SNECMA
Number of Engines	4
Max thrust per engine at take off	33,620 lbf (171,78kN) with afterburner
Max thrust per engine at supersonic cruise	10,030 lbf (44.61 kN) without afterburner
Reheat contribution to performance	20% at full thrust during take off
Fuel type	A1 jet fuel
Fuel capacity	43,392 lbm (95,680 kg)
Fuel consumption at full power	23,152 lbm/hr (10,500 kg/hr)
Fuel consumption at full reheated power	49,612 lbm/hr (22,500 kg/hr)
Typical miles/gallon per passenger	17

Table 1.1: Powerplant Specifications

Certain flight conditions soon became important in the preliminary design phase of the *Olympus 593* development program and remain relevant for equivalent modern engine ventures. A "supersonic engine" is never just that, since it also must perform well over a wide range of subsonic speeds before and after cruise conditions. Multiple design points must be considered. Each design point has its own demands but severe compromises must always be made to ensure operational compatibility. Unfortunately for engineers, the compromises are also driven heavily by money!

- Engine performance at cruise conditions is critical because that is where a high percentage of the fuel is consumed; unlike a subsonic aircraft, the engines cannot be throttled back in this region of the mission because it takes a lot of thrust to maintain sujpersonic flight speeds in any aircraft. Often, performance at top-of-climb sizes the propulsion system.

- Take-off must be addressed because the maximumm level of absolute thrust is needed to accelerate the aircraft from brake-release and allow it to take off within a specified distance. However, the engine can be throttled back once the undecarriage is retracted and the drag is reduced. This is fortunate because, as stated earlier, take-off noise is huge issue and that is driven by jet speed. Here we seek to maximize airflow so that required momentum of the exhaust jet can be maintained at a lower value of velocity.

- For some aircraft, regardless of their cruise speed, a “pinch point” occurs between the net thrust an engine can deliver and that which the airplane needs at transonic situations – pushing thorugh the sound barrier, as it used to be called. So this mission segment may turn out to size the engine.

Some values of relevant parameters are shown in *Table 1.2*.

	Cruise	**End of Runway T. O.**	**Max Climb**
Mach Number	2.0	0.302	1.2
Altitude (ft)	53,000	0	40,000
Conditions	ISA + 5°C	ISA + 10°C	ISA + 5°C
Inlet Pressure Recovery	0.937	0.986	0.986
After burner	Off	On	On
Net Thrust (lbf)	10,030	33,620	13,610
Specific Fuel Consumption (lbm/lbf/hr	1.19	1.39	1.41

Table 1.2: Performance Data per Engine at Critical Flight Conditions

The *Concorde* program demonstrated quite dramatically that relatively small increases in the weight of engines, airframe or fuel load result in dramatic reductions in either range or payload (*Reference 2*.), so improving fuel consumption to save, say, 2% of aircraft gross weight is of no value if it is offset by corresponding increase in engine weight.

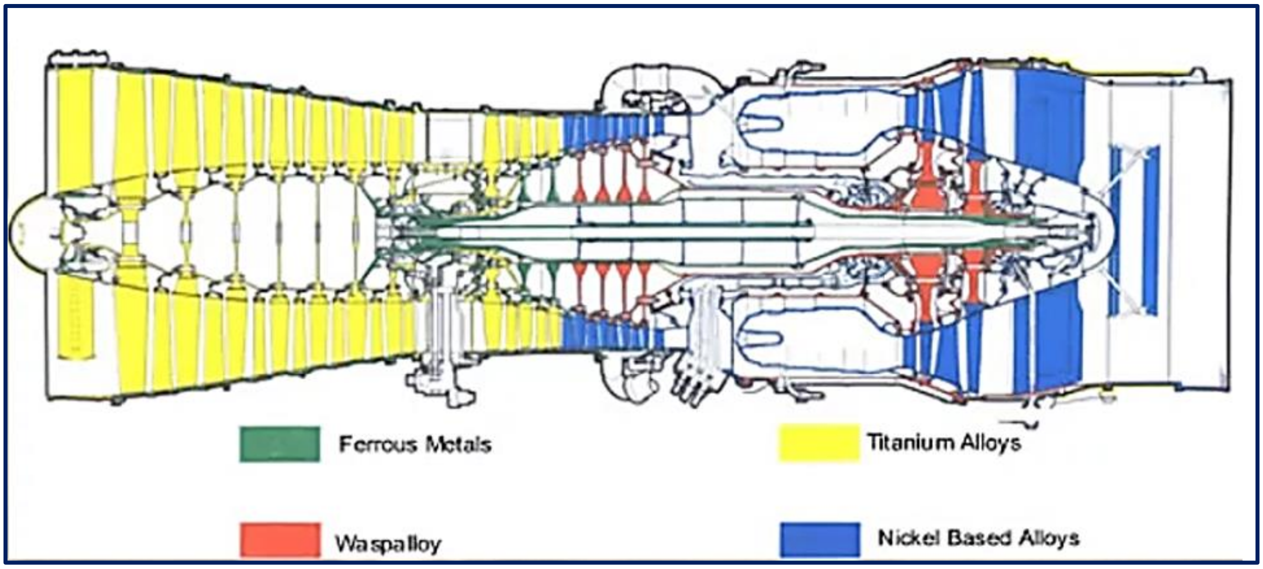

Figure 1.4: Olympus 593 Mk 610 Cross-Section

Figure 1.4 is a cross-section of the *Olympus 593 Mk 610* engine, which illustrates the flowpath geometry and the general categories of materials used. Of course, the latter correspond to prevailing temperatures. The figure omits the inlet and nozzle. The overall length (159 inches) in *Figure 1.3* corresponds to the distance between the leading edge of the inlet centerbody and the trailing edge of the large turbine exit strut. The diameter (47.75 inches) corresponds to the fan tip value. The dry weight (7000 lbm) in the data of *Figure 1.3* excludes the inlet, the tailpipe and the nozzle and covers what is shown in *Figure 1.4.*

1.3 Future Supersonic Transport Engines

Since $sfc = {}^{V_a}/_{\eta_{TH}\eta_P Q}$, to obtain low specific fuel consumption we require an engine that combines high thermal efficiency with high propulsive efficiency. A simple turbojet has high η_{TH} only at high T41 and high η_P only at low T41, but a turbofan engine allows a high $\eta_{TH}\eta_P$ product to be achieved by employing a high T_{41} but transferring energy from its core stream to a bypass stream, from which the jet velocity is much lower. The early quest for fuel economy have led directly to lower emissions at cruise and, somewhat indirectly, to low noise at take-off. Both of these have benefitted us immensely, in light of the tremendous growth of aviation over the past seventy years. In recent years, subsonic commercial aviation has been dominated by higher and higher bypass ratio propulsion systems, enabled by higher turbine entry temperatures based on improved turbine materials and cooling technology. For supersonic missions, the use of turbofans

– although of limited bypass ratio – is extremely attractive to optimize fuel burn at cruise and reduce noise at take-off by maximizing engine airflow. *Reference 2* discusses this extensively.

2. Design Objectives and Requirements

- A new low bypass ratio mixed turbofan engine design is required for the existing *Concorde* airframe, with an entry-into-service date of 2028. Four engines will be used.

- The existing inlet will be retained. Assume inlet pressure recovery values from *Table 2*.

- The primary design point for the new engine should be supersonic cruise conditions at 53,000 feet/Mach 2 (ISA +5^{o}C). The net thrust must be 10,000 lbf.

- The second "off-design" point should be rolling take-off at sea level/Mach 0.3 (ISA +10^{o}C). The net thrust must be 33,600 lbf.

- Reheat at take-off is to be eliminated, if possible.

- It is hoped to extend the range by reducing fuel consumption and minimizing engine mass.

- To accommodate a turbofan configuration, the diameter of the new engines may be increased but should be kept to a minimum.

- Based on the entry-into-service date, development of new materials and an increase in design limits may be assumed. Set a new limit of 3150 R for T4. Consider the use of carbon matrix composites in the HP turbine. Carefully justify your choices of any new materials, their location and the appropriate advances in design limits that they provide.

- T3 should be limited to 1620 R. If reheat cannot be avoided, T7 should be below 2100 R.

- Generate your own version of the *Olympus 593* baseline engine model as a reference and include it in your proposal.

- Your new engine design should be optimized for minimum engine mass and fuel burn. Use trade studies to determine the best combination of design variables.

- A variable-geometry convergent-divergent nozzle is necessary to enable efficient supersonic cruise and meet noise restrictions at take-off. To satisfy the noise requirement, do no more than ensure that the jet velocity at take-off for a fully-expanded nozzle does not exceed 1150 ft/s. Bear in mind that this limit is for "end of runway" measurement purposes.

- Design proposals must include engine mass, engine dimensions, net thrust values, specific fuel consumption, thermal and propulsive efficiencies at supersonic cruise and rolling take-off. Details of the major flow path components must be given. These include inlet, fan, booster, HP compressor, primary combustor, HP turbine, LP turbine, exhaust nozzle, bypass duct, mixer, afterburner and any inter-connecting ducts. Examples of velocity diagrams should be included to demonstrate viability of some of the turbomachinery.

3. Baseline Engine Model

3.1 Cruise Conditions: The Design Point

A generic model of the *Rolls-Royce/SNECMA Olympus 593 Mk 610* has been generated from publicly available information *(Reference 2)* using *GasTurb13*. 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. It should be remembered that we can exceed many of the baseline performance parameters with today's technology, materials and design tools.

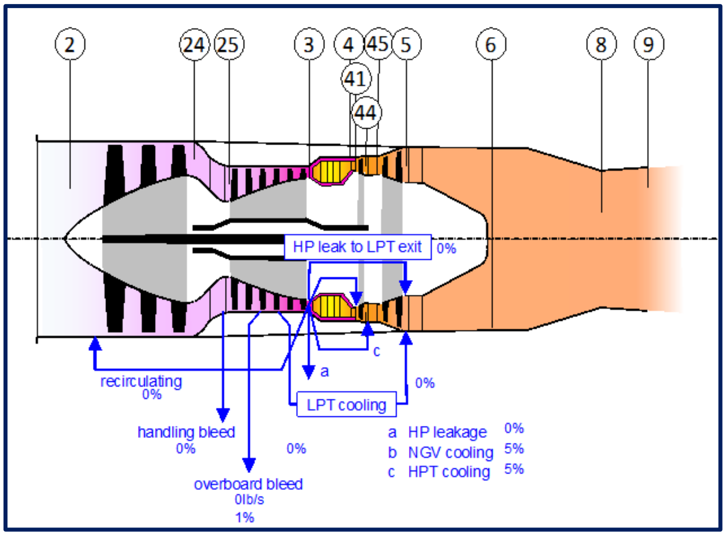

Figure 3.1: Turbojet Engine Schematic with Calculation Stations & Secondary Flows

Figure3.1 contains a general schematic with relevant station numbers and secondary flow data for a non-augmented turbojet engine. *Figure 3.2* shows an after-burning system.

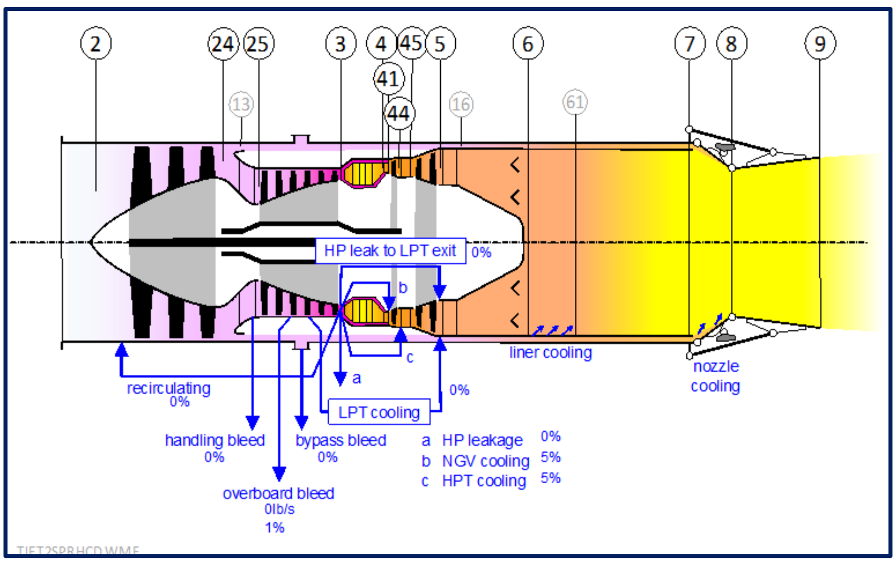

Figure 3.2: Schematic of a Turbojet Engine with Reheat

3.1.1 Overall Characteristics

Major Design Parameters

In a turbojet engine, the two primary design variables are turbine entry temperature (T_4), and overall pressure ratio (*OPR* or P_3/P_2). For two spools the optimum energy division must be determined.

Property	*Unit*	*Value*	*Comment*
Inlet Corr. Flow W2Rstd	lb/s	462.971	
Intake Pressure Ratio		0.937	
LP Compressor Pressure Ratio		4.1	
Compr. Interduct Press. Ratio		0.99	
HP Compressor Pressure Ratio		2.9	
Turb. Interd. Ref. Press. Ratio		0.98	
Burner Exit Temperature	R	2430	
Burner Design Efficiency		0.99	
Burner Partload Constant		1.6	used for off design only
Fuel Heating Value	BTU/lb	18552.4	
Overboard Bleed	lb/s	0	
Power Offtake	hp	100	
HP Spool Mechanical Efficiency		0.99	
LP Spool Mechanical Efficiency		0.99	
Burner Pressure Ratio		0.96	
Turbine Exit Duct Press Ratio		0.98	

Table 3.1: Basic Design Input

Table 3.1 is the “Basic Input” for the design point of a *GasTurb13* model of the *Olympus 593* baseline. Both primary design variables are input, the overall pressure ratio being made up from the LPC, the HPC and the inter-compressor duct loss. T4, as well as the inlet pressure recovery, were obtained from *Reference 2*. To generate an acceptable replica of the engine cycle, a unique combination of the remainder must be estimated iteratively using the net thrust (F_N) and specific fuel consumption (*sfc*) at cruise conditions as targets. By definition, this operating condition also corresponds to the engine design point, the entry point to any component performance maps, and this should be the case for your new engine.

The next four parameters relate to the primary combustor; they are all fairly conventional values by modern standards. The burner efficiency of 99% corresponds to the 1960s and 99.9% is more current. A burner pressure loss of 4% is given up willingly to pay for complete mixing and efficient combustion, so this should be retained. The burner “*part load constant*” is an element in the calculation of burner efficiency discussed in the *GasTurb13 User Guide* in *Reference 3*. Without expert knowledge, this is best left alone!

Secondary Design Parameters

Cooling Air: An overboard bleed is listed in *Table 3.2*. Strictly, this is unnecessary for our non-afterburning design case, but it is needed to cool the afterburner for take-off with reheat. 5% of HPC air is bled form compressor delivery to cool both the HP turbine vane and blade. Fully-compressed air is an expensive commodity, but this is the only source that offers sufficient pressure to permit to coolant to be delivered to the hot vane and blade and emerge from their surfaces. This is aided by the pressure loss through the burner – another reason why we can tolerate combustor pressure losses.

Property	Unit	Value	Comment
Rel. Handling Bleed		0	
Rel. Overboard Bleed W_Bld/W25		0.01	
Rel. Enthalpy of Overb. Bleed		1	
Recirculating Bleed W_reci/W25		0	Off Design Input Only
Rel. Enthalpy of Recirc Bleed		1	
HP Overboard Leak WLk/W25		0	
Number of HP Turbine Stages		1	
HPT NGV 1 Cooling Air / W25		0.05	
HPT Rotor 1 Cooling Air / W25		0.05	
HPT Cooling Air Pumping Dia	in	0	
Number of LP Turbine Stages		1	
LPT Rotor Cooling Air W_Cl/W25		0	
Rel. Enth. of LPT Cooling Air		0.6	
Rel.HP Leakage to LPT exit		0	

Table 3.2: Secondary Air System Input

Turbomachinery Efficiencies: For our baseline model, efficiencies of the LP and HP compressors and HP turbine and LP turbines were entered directly via respective tabs on the input screen. The values are not listed specifically in the tables shown but may be reviewed in the output summary presented later in *Table 3.4*. The designer has the choice of either isentropic or polytropic values, so he or she should be certain of their applicability and their definitions! However, another available option allows *GasTurb13* to calculate efficiencies from data supplied. Compressors use a NASA approach (*Reference 4*) but turbines first estimate prevailing values of stage loading and flow coefficients for use in a *Smith Chart (Reference 5)*, assuming an equal work spilt between stages. This is a most convenient approach to turbine performance since various updated versions of the *Smith Chart* are available. More will be said about this topic in Sub-sections 3.7 and 3.8.

Power Off-take: All engines have power extracted - usually from the HP spool via a tower shaft that passes through an enlarged vane or strut in the main frame – to power aircraft systems. This is often preferred to the use of a separate auxiliary power unit, depending on how much power is required. In the application currently under consideration, considerable auxiliary power may be needed for avionics and passenger equipment and this usage is growing rapidly in modern aircraft. We have selected a nominal power off-take of 100 hp from our baseline engine. Modern engines tend to use a lot of this, so you might like to consider this issue for your engine and mission.

Mixer Efficiency: Since a turbojet has a single flow stream, the Olympus 593 does not require a mixer, but the required new turbofan architecture probably will. Mixer efficiency quantifies the degree of mixing that is achieved at plane 163 between the core flow and the bypass flow. It can be shown analytically that thrust is maximized if the mixing is complete. In order to do this a large and heavy active mixer would be required; therefore an appropriate compromise is arrived at, since a large mixer means a heavier engine that requires more thrust – an uphill spiral! For an exceedingly long mission, the additional mixer weight is justified. In order to optimize whatever mixing is aerodynamically possible, the designer must also ensure that the (static) pressures are (roughly) equalized in the flows leaving the engine core and bypass duct by trading the work balance between the high- and low-speed spools and adjusting annulus areas to effect velocities. The bypass ratio also plays a key role here.

Dimensions: Diameters & Lengths: The engine cycle may be defined purely on the basis of thermodynamics. We define a "rubber engine" initially, where performance is delivered in terms of a net thrust at cruise close to 10,000 lbf given in *Table 1.1* once the engine scale has been determined. For our baseline model, we also had a target dimensional envelope defined in *Figure 1.3*, namely a maximum fan diameter of 47.75 inches and a maximum length of 159 inches,. The diameter is determined from the mass flow rate and the Mach number at the LPC face; the length is a separate issue that is dealt with by manipulation of vane & blade aspect ratios and axial gaps in the turbomachinery and by suitable selection of duct lengths, usually defined as fractions of the corresponding entry radii. Once the correct thrust has been reached, the maximum radius is determined by setting an inlet radius ratio and then varying the Mach number at entry to the LPC. These values are input on the primary input screen under the LP compressor tab, where a Mach number of 0.549 was found to be appropriate - fairly low by today's standards. This sets the general radial dimension for the complete engine, although in fact downstream of the LPC, the entry radius of the HP compressor is also determined by input radius ratios and values of local axial Mach number given in *Table 3.3*.

Name	*Where it is*	*Design Mach No*	*Design Area*
St2	LP Compressor Inlet	Calculated by	LPC Design
St24	LP Compressor Exit	0.4804	0
St25	HP Compressor Inlet	Calculated by	HPC Design
St3	HP Compressor Exit	0.286	0
St4	Burner Exit	0.2516	0
St44	HP Turbine Exit	0.5147	0
St45	LP Turbine Inlet	0.5719	0
St5	LP Turbine Exit	0.6062	0
St6	Gas Generator Exit	0.5	0
St8	*Nozzle Throat*	*0*	*0*
St9	*Nozzle Exit*	*0*	*0*

Table 3.3: Stations Input

The HP & LP turbine radii follow from the exit values of the respective upstream components. For the ducts, radial dimensions are keyed off the inner wall with the blade spans being superimposed. For the overall engine length, early adjustments are made by eye (My personal philosophy is that if it looks right, it's probably OK!), with final manipulations being added as the target dimension is approached. When modeling an existing engine, *GasTurb13* enables an available cross section to be located beneath the model, so that the model can be manipulated via numerical input or sliders assigned to input parameters, until a satisfactory match is achieved. The degree of success can be seen in *Figure 3.4*, where the upper portion of the Olympus 593 cross section from *Figure 1.4* may be seen behind the model.

Materials & Weights: Use was made of the materials database in the *GasTurb13* design code, where, in fact, the default selections were retained for the *Olympus 593*. For proprietary reasons, many advanced materials are not included. Examples of these are: polymeric composites used in cold parts of the engine, such as the inlet and fan; metal matrix composites, which might be expected in the exhaust system; carbon-carbon products, again intended for use in hot sections. All of these materials are considerably lighter than conventional alternatives, Within the component models, material densities can be modified independently of the database.

Component weights are calculated by multiplying the effective volumes by the corresponding material densities. Of course, only the major elements which are explicitly designed are weighed and there are many more constituents. Nuts, bolts, washers, seals and other much larger elements such as fuel lines, oil lines, pumps and control systems still must be accounted for. In industry, this is done by the application of a multiplier or adder to the predicted net mass, whose value is based on decades of experience, to obtain what is designated in the output as the total mass. In general, a multiplication factor of 1.3 is recommended in the *GasTurb13* manual, but I used a "*net mass factor*" of 1.2173 in *Table 3.21* to reach the overall mass target of 7000 lbm (without nozzle) in *Figure 1.3*

A summary of the output for the *Olympus 593* model for the design point at cruise is given in *Table 3.4*. The net thrust is within 0.3% of the target. Unfortunately, the predicted specific fuel consumption of 1.33 is considerably higher than the quoted value of 1.195 in *Figure 1.3*. To be honest, I don't know why. See what you can come up with in your baseline model!

It must also be stated at this point, that my guess for the pressure ratio split between the LP and HP compressors could have been better! It should have been more even. In reaching the data in *Table 3.4*, I sought to make the work and temperature splits roughly equal in achieving the target value of temperature increases ΔT 2–3 = 810 R. This led to a skewing of my efficiency estimates.

A different format of thermodynamic information is contained in *Table 3.5*. Local values of mass flow rate, temperature, pressure, velocity, flowpath area, axial Mach number, and radii - together with their axial locations - are especially useful.

```
              W           T          P        WRstd
Station     lb/s          R         psia       lb/s          FN        =  10031.94 lb
 amb                   389.97      1.456                     TSFC      =    1.3304 lb/(lb*h)
  1       289.345      701.78     11.402                     FN/W2     =   1115.51 ft/s
  2       289.345      701.78     10.684     462.971         WF Burner=    3.70745 lb/s
 24       289.345     1098.53     43.803
 25       289.345     1098.53     43.365     142.705         P2/P1     =    0.9370
  3       289.345     1537.76    125.757      58.221         P25/P24   =    0.9900
 31       257.517     1537.76    125.757                     P3/P2     =   11.7711
  4       261.224     2430.00    120.727      68.828         P45/P44   =    0.9800
 41       275.692     2385.73    120.727      71.975         P6/P5     =    0.9800
 43       275.692     1965.47     47.934
 44       290.159     1944.88     47.934
 45       290.159     1944.88     46.976     175.776         W_NGV/W25=    0.05000
 49       290.159     1589.72     18.800                     WHcl/W25 =    0.05000
  5       290.159     1589.72     18.800     397.100         WLcl/W25 =    0.00000
  6       290.159     1589.72     18.424                     XM6       =   0.50000
  8       290.159     1589.72     18.424     405.204         A8        =   1247.57 in²
Bleed       2.893     1537.76    125.757                     WBld/W2   =   0.01000
---------------------------------------------               Ang8      =     20.00 °
Efficiencies:     isentr   polytr     RNI      P/P           CD8       =   0.96000
 LP Compressor    0.8530   0.8782   0.507    4.100           P8/Pamb   =  12.65199
 HP Compressor    0.8170   0.8402   1.208    2.900           WlkLP/W25=    0.00000
 Burner           0.9900                     0.960           Loading   =    100.00 %
 HP Turbine       0.8900   0.8785   1.378    2.519           e444 th   =   0.85985
 LP Turbine       0.9000   0.8890   0.677    2.499           WlkO/W25 =    0.00000
---------------------------------------------
HP Spool mech Eff 0.9900 Nom Spd     8382 rpm                PWX       =     100.0 hp
LP Spool mech Eff 0.9900 Nom Spd     5819 rpm                Core Eff =     0.5425
---------------------------------------------               Prop Eff =     0.7778
Con-Di Nozzle:                                               A9/A8     =   1.80000
 A9*(Ps9-Pamb)     1536.274                                  CFGid     =   0.95633
---------------------------------------------
hum [%]      war0       FHV      Fuel
   0.0    0.00000    18552.4    Generic

Input Data File:
C:\Concorde Re-Engine Project\GasTurb13 Files\Olympus593_TOC_Scaled_3Aug2020.C2J
 (modified)
```

Table 3.4: Olympus 593 Baseline Engine Output Summary at Cruise

	Units	St 2	St 24	St 25	St 3	St 4	St 44	St 45	St 5	St 6	St 8	St 9
Mass Flow	lb/s	289.345	289.345	289.345	289.345	261.224	290.159	290.159	290.159	290.159	290.159	290.159
Total Temperature	R	701.784	1098.53	1098.53	1537.76	2430	1944.88	1944.88	1589.72	1589.72	1589.72	1589.72
Static Temperature	R	662.116	1052.75	1049.07	1516.12	2406.65	1864.83	1846.95	1496.06	1524.85	1355.59	904.035
Total Pressure	psia	10.6836	43.8026	43.3646	125.757	120.727	47.9343	46.9756	18.7995	18.4236	18.4236	18.4236
Static Pressure	psia	8.70915	37.4947	36.6522	119.047	115.855	40.3644	38.0285	14.7974	15.6331	9.88352	2.1688
Velocity	ft/s	691.704	757.907	787.511	536.003	584.777	1060.21	1172.67	1125.65	936.809	1772.8	2993.21
Area	in^2	1696.69	571.881	561.066	366.785	495.073	674.579	641.136	1390.41	1611.81	1197.67	2155.81
Mach Number		0.549	0.4804	0.499998	0.286	0.2516	0.5147	0.5719	0.6062	0.5	1	2.04626
Density	lb/ft^3	0.035502	0.096129	0.094299	0.211933	0.129932	0.058422	0.055574	0.026696	0.027672	0.019679	6.4752E-3
Spec Heat @ T	BTU/(lb*R)	0.242022	0.251783	0.251783	0.265477	0.292708	0.281946	0.281946	0.272177	0.272177	0.272177	0.272177
Spec Heat @ Ts	BTU/(lb*R)	0.241564	0.250417	0.250312	0.264839	0.292299	0.2799	0.279443	0.269247	0.270148	0.264669	0.24997
Enthalpy @ T	BTU/lb	39.7942	137.553	137.553	251.268	508.221	367.81	367.81	269.341	269.341	269.341	269.341
Enthalpy @ Ts	BTU/lb	30.2328	126.073	125.159	245.526	501.388	345.348	340.329	244.02	251.803	206.535	90.2987
Entropy Function @ T		0.941904	2.54862	2.54862	3.81588	5.76206	4.81934	4.81934	4.00519	4.00519	4.00519	4.00519
Entropy Function @ Ts		0.737572	2.39312	2.38045	3.76105	5.72087	4.64745	4.60804	3.76581	3.84095	3.38243	1.86574
Exergy	BTU/lb	73.2288	165.751	165.482	273.778	477.581	337.678	337.138	235.949	235.408	235.408	235.408
Gas Constant	BTU/(lb*R)	0.068607	0.068607	0.068607	0.068607	0.068606	0.068606	0.068606	0.068606	0.068606	0.068606	0.068606
Fuel-Air-Ratio		0	0	0	0	0.014397	0.012943	0.012943	0.012943	0.012943	0.012943	0.012943
Water-Air-Ratio		0	0	0	0	0	0	0	0	0	0	0
Inner Radius	in	6.51669	10.0332	9.51496	13.6498	14.7944	14.2026	14.2026	10.936	0	0	0
Outer Radius	in	24.1359	16.779	16.4051	17.4383	19.4026	20.4068	20.1443	23.7103	22.6507	19.9277	26.7359
Axial Position	in	17.6488	62.0073	68.6292	106.216	133.552	138.818	138.833	146.651	178.66	195.648	235.503

Table 3.5: Olympus 593 Baseline Engine Detailed Output

A plot of the baseline engine model appears in *Figure 3.3* and as stated earlier, a comparison with the prototype cross section is shown in *Figure 3.4*.

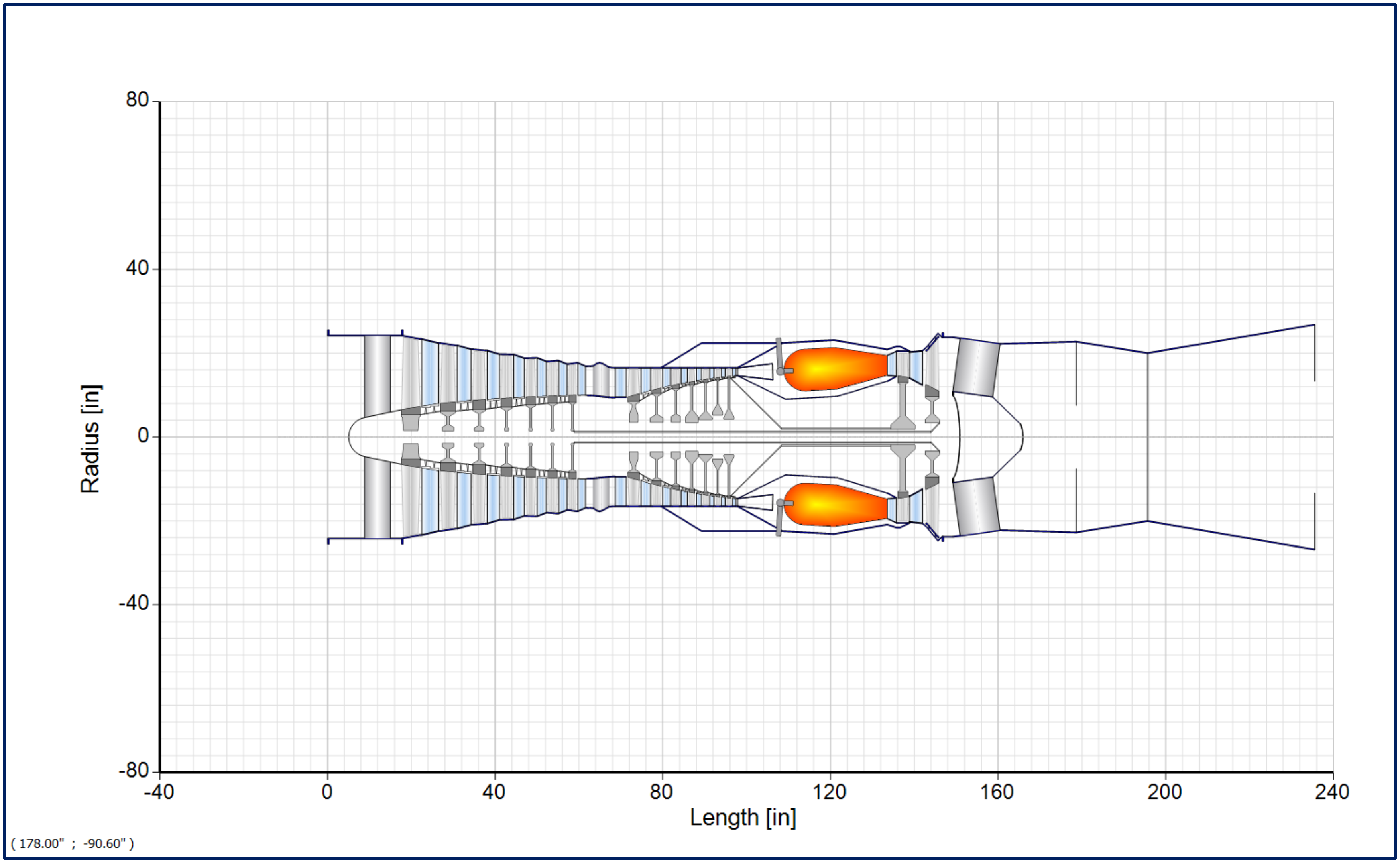

Figure 3.3: Olympus 593 Baseline Engine GasTurb13 Model Cross Section

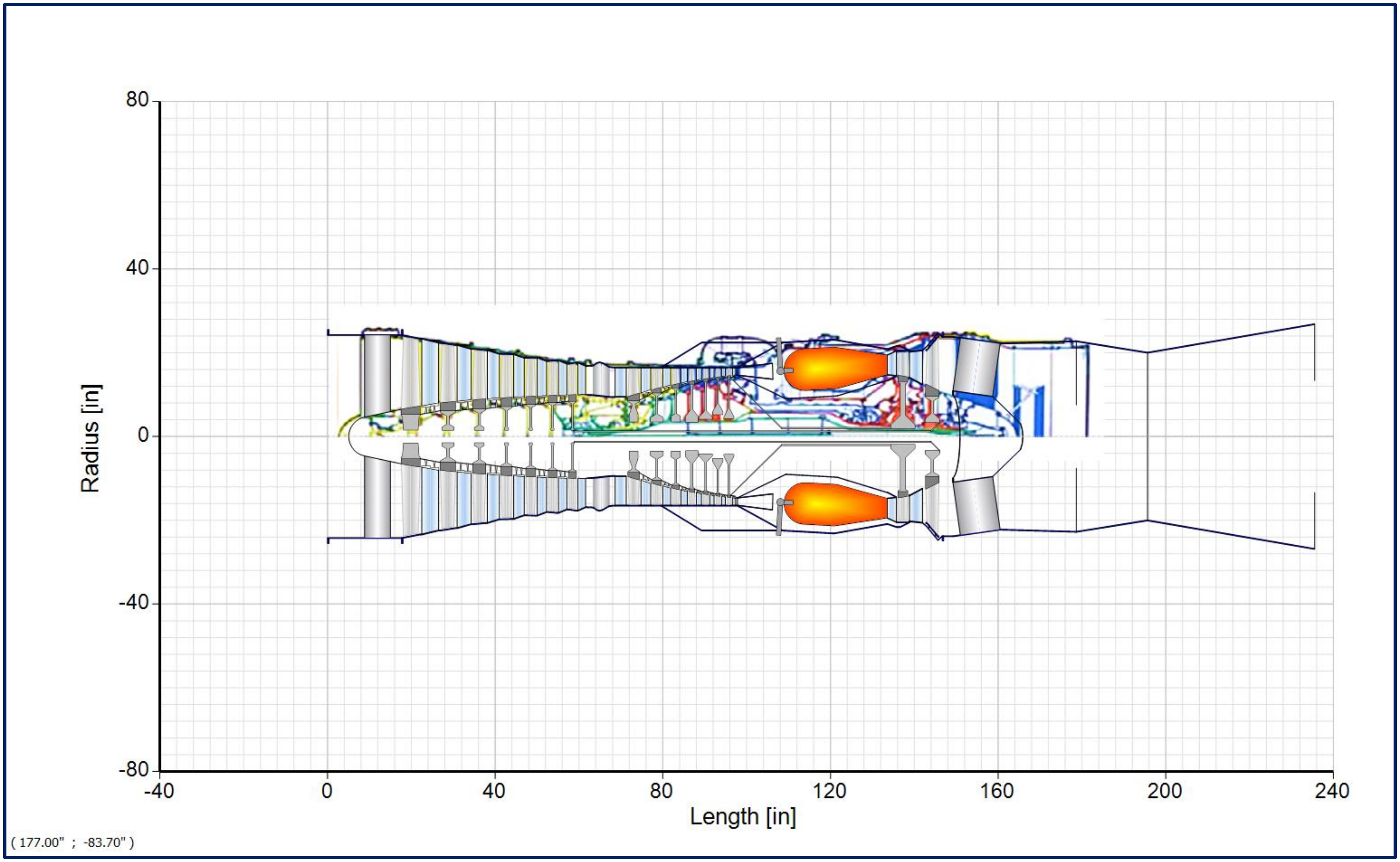

Figure 3.4: Comparison of GasTurb13 Olympus 593Model with Engine Cross Section

Some details of the component models now follow.

3.1.2 Inlet

Note that in this project we are not concerned with the real two-dimensional variable inlet, used in the Concorde to entrain the necessary air flow and reconcile this with the engine. We are currently interested in the hardware downstream of the inlet flange, as in *Figure 1.4*. The inlet is designed with an elliptical center body (*Figure 3.3*). The outer diameter of the inlet has been determined from that of the fan.

Number of Struts		8
Strut Chord/Height		0.34
Gap Width/Height		0.15
Cone Length/Radius		1.25
Cone Angle [deg]		12
Casing Length/Radius		0.6
Casing Thickness	in	0.19685
Casing Material Density	lb/ft³	249.712
Inlet Mass Factor		1

Length	in	17.6488
Cone Length	in	8.82441
Cone Mass	lbm	12.1829
Casing Mass	lbm	76.136
Strut Mass	lbm	53.0005
Total Mass	lbm	141.319

Table 3.6: Inlet Geometry Input & Output

Pertinent geometric characteristics are shown in *Table 3.6*. At 141 lbm, the inlet is fairly light and this is because, based on the density (*Figure 1.4*), we have taken a typical *Ti-Al* alloy as our choice of materials. This should accommodate the dynamic heating effects of Mach 2 operation.

3.1.3 Low Pressure Compressor

The LP compressor characteristics are given in *Tables3.7 and 3.8*. The radius ratio and inlet Mach number are of particular interest because, when taken with mass flow rate, they define the fan tip radius. Based on tip radius the blade tip speed sets the rotational speed of the LP spool. The value of corrected flow per unit area (39.29 lbm/ft^2) is modest by modern standards and corresponds to the input value of Mach number 0.549. Your new design can exceed this.

Input:		
LPC Tip Speed	ft/s	1225.58
LPC Inlet Radius Ratio		0.27000
LPC Inlet Mach Number		0.54900
Engine Inl/LPC Tip Diam Ratio		1.00000
min LPC Inlet Hub Diameter	in	0.00000
Output:		
LPC Tip circumf. Mach No		0.97273
LPC Tip relative Mach No		1.11696
Design LP Spool Speed	[RPM]	5818.76
LPC Inlet Tip Diameter	in	48.27180
LPC Inlet Hub Diameter	in	13.03339
Calculated LPC Radius Ratio		0.27000
LP Spool Torque	lb*ft	0.00000
Aerodynamic Interface Plane	in²	1830.11
Corr.Flow/Area LPC	lb/(s*ft²)	39.29275

Table 3.7: Low Pressure Compressor Aerodynamics Input & Output

Number of Stages		7
Inlet Guide Vanes (IGV) 0,		0
IGV Profile Thickness [%]		5
IGV Material Density	lb/ft³	249.712
Annulus Shape Descr -0.5..		0.26
Given Radius Rat: Inl/Exit 0		0
Inlet Radius Ratio		0.3
Exit Radius Ratio		0.9
Blade and Vane Sweep		0
First Stage Aspect Ratio		4
Last Stage Aspect Ratio		4.2
Blade Gapping: Gap/Chord		0.1
Pitch/Chord Ratio		0.7
Disk Bore / Inner Inlet Radii		0.2
Rel Thickness Inner Air Seal		0.04
IP Compressor Mass Factor		1
Casing Thickness	in	0.19685
Casing Material Density	lb/ft³	249.712
Casing Thermal Exp Coeff	E-6/R	18
Casing Specific Heat	BTU/(lb	0.11950:
Casing Time Constant		10
Blade and Vane Time Const		0.5
Platform Time Constant		1
Design Tip Clearance [%]		1.5
d Flow / d Tip Clear.		2
d Eff / d Tip Clear.		2
d Surge Margin / d Tip Clea		5

Length	in	44.3585
Total Number of Blade and		714
Casing Mass	lbm	164.411
Total Vane Mass	lbm	508.393
Total Blade Mass	lbm	809.569
Inner Air Seal Mass	lbm	48.9754
Rotating Mass	lbm	1170.29
Total Mass	lbm	1843.09
Polar Moment of Inertia	lb*in²	149140

Table 3.8: Low Pressure Compressor Geometry Input & Output

3.1.4 Inter-Compressor Duct

Number of Struts		6
Length/Inlet Inner Radius		0.66
Inner Annulus Slope@Inlet[		0
Inner Annulus Slope@Exit [		0
Relative Strut Length [%]		60
Casing Thickness	in	0.19685
Casing Material Density	lb/ft³	499.424
Compr Interduct Mass Fact		1

Length	in	6.62192
Outer Casing Mass	lbm	39.3917
Strut Mass	lbm	18.5412
Inner Casing Mass	lbm	23.2075
Total Mass	lbm	81.1403

Table 3.9: Inter-Compressor Duct

Notice that in addition to using an overall net mass factor to adjust the engine weight, individual net mass factors may be applied to the components or net mass adders may be used. This remains at a value of unity for the inter-compressor duct at the bottom of the left-hand box in *Table 3.9* since little of the structure is unaccounted for in our simple model.

3.1.5 High Pressure Compressor

```
 Input:
HPC Tip Speed                  ft/s             1200.00
HPC Inlet Radius Ratio                          0.58000
HPC Inlet Mach Number                           0.50000
min HPC Inlet Hub Diameter     in               0.00000
 Output:
HPC Tip circumf. Mach No                        0.76189
HPC Tip relative Mach No                        0.91131
Design HP Spool Speed     [RPM]                 8382.14
HPC Inlet Tip Diameter         in              32.81021
HPC Inlet Hub Diameter         in              19.02992
Calculated HPC Radius Ratio                     0.58000
HP Spool Torque                lb*ft           29507.42
Corr.Flow/Area HPC             lb/(s*ft²)      36.62574
```

Table 3.10: High Pressure Compressor Aerodynamics Input & Output

Again, we set the speed of the HP spool via the tip speed and the corresponding radius. General aerodynamic characteristics of the HP compressor are given in *Table 3.10*, with the geometry defined in *Table 3.11*.

Number of Stages		7
Number of Radial Stages		0
Number of Variable Guide V		0
Inlet Guide Vanes (IGV) 0,		1
IGV Profile Thickness [%]		5
IGV Material Density	lb/ft³	249.712
Annulus Shape Descriptor (		1
Given Radius Rat: Inl/Exit 0		0
Inlet Radius Ratio		0.68
Exit Radius Ratio		0.9
Blade and Vane Sweep		0
First Stage Aspect Ratio		2.6
Last Stage Aspect Ratio		2.4
Blade Gapping: Gap/Chord		0.16
Pitch/Chord Ratio		0.9
Disk Bore / Inner Inlet Radiu		0.3
Diffuser Area Ratio		2.2
Rel Thickness Inner Air Seal		0.04
Compressor Mass Factor		1
Outer Casing Thickness	in	0.19685
Outer Casing Material Densi	lb/ft³	249.712
Casing Thickness	in	0.19685
Casing Material Density	lb/ft³	249.712
Rel Work of Radial End Sta		0.3
Duct Inner Radius Ratio		1
Duct Length/Inlet Inner Ra		0
Number of Duct Struts		8
Relative Duct Strut Length		60
Rad Diffusor/Rotor Blade Le		0.5
Rotor Inlet Swirl Angle		0
Rotor Blade Backsweep An		20
Diffusor Wall Thickness	in	0.09842
Casing Thermal Exp Coeff	E-6/R	18
Casing Specific Heat	BTU/(lb	0.11950
Casing Time Constant		10

Length (w/o Diffusor)	in	29.1713
Number of Inlet Guide Vane		34
Total Number of Blade and		1195
Diffusor Length	in	8.41509
Casing Mass	lbm	85.5355
Outer Casing Mass	lbm	116.424
Total Vane Mass	lbm	64.9342
Total Blade Mass	lbm	143.189
Inner Air Seal Mass	lbm	34.7122
Rotating Mass	lbm	740.08
IGV Mass	lbm	11.8873
Exit Diffusor Mass	lbm	47.1101
Total Mass	lbm	1065.97
Polar Moment of Inertia	lb*in²	72487.3

Table 3.11: High Pressure Compressor Geometry Input & Output

3.1.6 Combustor

A fairly conventional annular combustor is used and geometric details are given in *Table 3.12*. The high density of its material corresponds to the necessary thermal properties. The combustor is a major structural component, linked closely to the HP turbine first vane assembly. This is emphasized by its significant mass.

Reverse Flow Design (0/1)		0
Outer Casing Length/Lengt		2
Exit/Inlet Radius		1.1
Length/Inlet Radius		2.3
Can Width/Can Length		0.4
Inner Casing Thickness	in	0.07874(
Outer Casing Thickness	in	0.19685
Casing Material Density	lb/ft³	499.424
Can Wall Thickness	in	0.19685
Can Material Density	lb/ft³	499.424
Can Thermal Exp Coeff	E-6/R	18
Can Specific Heat	BTU/(lb	0.11950:
Can Time Constant		1
Mass of Fuel Inj. / Fuel Flow		2
Burner Mass Factor		1

Mean Radius, Exit	in	17.0985
Length	in	35.7514
Can Volume	in³	20586.2
Can Mass	lbm	317.245
Can Surface Area / Mass	in²/lbm	35.1535
Fuel Injector Mass	lbm	7.41491
Inner Casing Mass	lbm	61.0622
Outer Casing Mass	lbm	285.193
Total Mass	lbm	670.915
Can Heat Soakage	hp	0

Table 3.12: Combustor Geometry Input & Output

3.1.7 High-Pressure Turbine

Property	*Unit*	*Value*	*Comment*
1. HPT Rotor Inlet Dia	in	38.41	
Last HPT Rotor Exit Dia	in	38.56	
HPT Exit Radius Ratio		0.7248	
HPT Vax.exit / Vax.average		1.29	
HPT Loss Factor [0.3...0.4]		0.35	
HPT 1. Rotor Cooling Constant		0	
Interduct Reference Mach No.		0.5	

Table 3.13: High Pressure Turbine Input to Calculate Efficiency

As stated on page 13, the efficiency of the high pressure turbine was input directly in order to model the *Olympus 593* cycle. However, I also chose to have *GasTurb13* calculate isentropic efficiency based on the data shown in *Table 3.13*, because additional valuable information is then revealed, as shown in *Table 3.14*. It should be noted that this calculated value is based on a modern *Smith Chart* and is therefore higher than that used in the cycle model. Also note that the efficiency contours are expressed as fractions of the maximum value on the chart.

A general summary of the HP turbine is given in *Table 3.14*, followed by the velocity diagrams and *Smith Chart* in *Figure 3.5*. In *Table 3.14*, the value of AN^2, (a measure of the disk rim stress) at almost 69 x 10^9 $in^2\,rpm^2$, is extremely high compared with a typical limit value of 45 x 10^9. That tells me I should have used a much lower rotational speed! This is borne out by the corresponding velocity diagram in *Figure 3.5*, which shows very little turning in the rotor blade. What the *Smith*

Chart tells us is that if we were to use exactly the same vanes and blade metal angles now, the efficiency would be greater than those input to the baseline engine cycle because of the superior aerodynamic design skills!

Input:			
Number of Stages			1
Last HPT Rotor Exit Dia	in		38.56000
HPT Exit Radius Ratio			0.72480
HPT Vax.exit / Vax.average			1.29000
HPT Loss Factor [0.3...0.4]			0.35000
HPT 1. Rotor Cooling Constant			0.00000
Interduct Reference Mach No.			0.50000
Output:			
HPT Inlet Radius Ratio			0.84299
HPT First Stator Exit Angle			70.19141
HPT Exit Mach Number			0.35651
HPT Exit Angle			-0.12933
HPT Last Rotor abs Inl Temp	R		2382.08
HPT First Rotor rel Inl Temp	R		2199.91
HPT First Stage H/T	BTU/(lb*R)		0.04404
HPT First Stage Loading			1.00210
HPT First Stage Vax/u			0.46513
HPT Exit Tip Speed	ft/s		1876.61
HPT Exit A*N*N	in²*RPM²*E-6		68958.62
HPT 1.Rotor Cool.Effectiveness			0.00000
HPT 1.Rotor Bld Metal Temp	R		2199.91
Velocities:			
Stage Inlet Absolute Velocity	V	ft/s	1721.97
Stage Inlet Axial Velocity	Vax	ft/s	583.54
Stage Inlet Relative Velocity	W	ft/s	583.54
Circumferential Velocity	U	ft/s	1618.39
Stage Exit Absolute Velocity	V	ft/s	752.77
Stage Exit Axial Velocity	Vax	ft/s	752.77
Stage Exit Relative Velocity	W	ft/s	1786.43

Table 3.14: High Pressure Turbine Aerodynamics Output

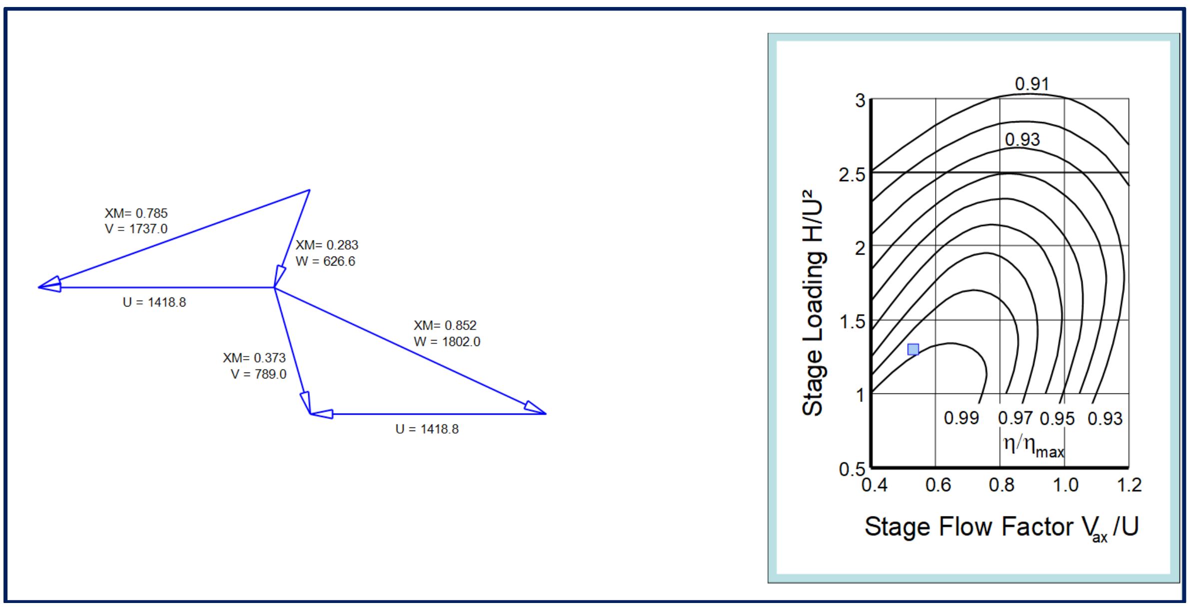

Figure 3.5: High Pressure Turbine Velocity Diagram & Smith Chart

Number of Stages = 1		no input
Unshrouded/Shrouded Bla		0
Inner Radius: R,exit / R,inle		0.96
Inner Annulus Slope@Inlet[		30
Inner Annulus Slope@Exit [		30
First Stage Aspect Ratio		2.1
Last Stage Aspect Ratio		3
Blade Gapping: Gap/Chord		0.2
Pitch/Chord Ratio		1
Disk Bore / Inner Inlet Radi		0.1
Rel Thickness Inner Air Seal		0.04
Turbine Mass Factor		1
Outer Casing Thickness	in	0.19685
Outer Casing Material Dens	lb/ft³	499.424
Casing Thickness	in	0.19685
Casing Cooling Effectivenes		0.5
Casing Material Density	lb/ft³	499.424
Casing Thermal Exp Coeff	E-6/R	18
Casing Specific Heat	BTU/(lb	0.11950:
Casing Time Constant		20
Blade and Vane Time Const		2
Platform Time Constant		5
Design Tip Clearance [%]		1.5
d Eff / d Tip Clear.		2

Length	in	5.26652
Total Number of Blade and		99
Casing Mass	lbm	38.1484
Outer Casing Mass	lbm	42.2267
Total Vane Mass	lbm	29.3291
Total Blade Mass	lbm	73.7744
Inner Air Seal Mass	lbm	0
Rotating Mass	lbm	315.661
Total Mass	lbm	425.365
Polar Moment of Inertia	lb*in²	41102.2

Table 3.15: High Pressure Turbine Geometry Input & Output

HP turbine geometric details are shown in *Table 3.15*.

3.1.8 Low-Pressure Turbine

Characteristics of the low pressure turbine are presented in *Tables 3.16 to* 3.18 and *Figure 3.6*. Except for the comments about excessive disk rim stress, the discussion is the same as for the HP turbine.

Property	*Unit*	*Value*	*Comment*
Number of LPC Stages		7	
LPC Loss Corr Factor		1	
IPC Exit Mach No		0.42	
IPC Exit Hub/Tip Radius Ratio		0.62	
IPC Last Stage Tip Clear.	mil	11.811	
% IPC Eff Change for % Clear		2	

Table 3.16: Low Pressure Turbine Input to Calculate Efficiency

Input:			
Number of Stages			1
LPT with EGV's [0/1]			1.00000
Last LPT Rotor Exit Dia	in		37.74000
LPT Exit Radius Ratio			0.49390
LPT Vax.exit / Vax.average			0.98000
LPT Loss Factor [0.3...0.4]			0.35000
LPT 1. Rotor Cooling Constant			0.00000
Output:			
LPT Inlet Radius Ratio			0.70797
LPT First Stator Exit Angle			65.89362
LPT Exit Mach Number			0.58860
LPT Exit Angle			-47.26265
LPT Last Rotor abs Inl Temp	R		1991.95
LPT First Rotor rel Inl Temp	R		1827.01
LPT First Stage H/T	BTU/(lb*R)		0.04619
LPT First Stage Loading			2.80652
LPT First Stage Vax/u			0.83459
LPT Exit Tip Speed	ft/s		1212.56
LPT Exit A*N*N	in²*RPM²*E-6		45858.58
LPT 1.Rotor Cool.Effectiveness			0.00000
LPT 1.Rotor Bld Metal Temp	R		1827.01
LPT Torque	lb*ft		36087.67
Velocities:			
Stage Inlet Absolute Velocity	V	ft/s	1888.53
Stage Inlet Axial Velocity	Vax	ft/s	771.34
Stage Inlet Relative Velocity	W	ft/s	1124.39
Circumferential Velocity	U	ft/s	905.72
Stage Exit Absolute Velocity	V	ft/s	1113.87
Stage Exit Axial Velocity	Vax	ft/s	755.91
Stage Exit Relative Velocity	W	ft/s	1882.28

Table 3.17: Low Pressure Turbine Aerodynamics Input & Output

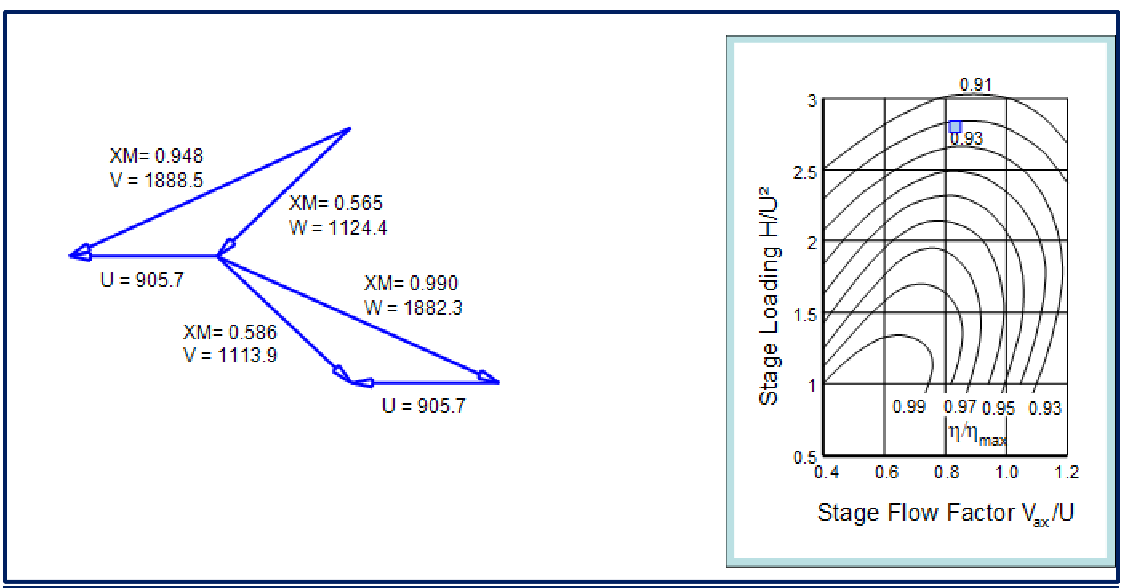

Figure 3.6: Low Pressure Turbine Velocity Diagram & Smith Chart

Number of Stages = 1		no input
Unshrouded/Shrouded Blac		1
Inner Radius: R,exit / R,inle		0.77
Inner Annulus Slope@Inlet[		25
Inner Annulus Slope@Exit [		25
First Stage Aspect Ratio		1.9
Last Stage Aspect Ratio		1.8
Blade Gapping: Gap/Chord		0.25
Pitch/Chord Ratio		1
Disk Bore / Inner Inlet Radi		0.2
Rel Thickness Inner Air Seal		0.04
LP Turbine Mass Factor		1
Casing Thickness	in	0.19685
Casing Cooling Effectivenes		0.5
Casing Material Density	lb/ft³	499.424
Casing Thermal Exp Coeff	E-6/R	18
Casing Specific Heat	BTU/(lb	0.11950:
Casing Time Constant		20
Blade and Vane Time Const		2
Platform Time Constant		5
Design Tip Clearance [%]		1.5
d Eff / d Tip Clear.		2

Length	in	7.81806
Total Number of Blade and		68
Casing Mass	lbm	67.3546
Total Vane Mass	lbm	67.5859
Total Blade Mass	lbm	231.874
Inner Air Seal Mass	lbm	0
Rotating Mass	lbm	335.46
Total Mass	lbm	470.401
Polar Moment of Inertia	lb*in²	64251.7

Table 3.18: Low Pressure Turbine Geometry Input & Output

3.1.9 Exhaust and Nozzle

The core exhaust is directly downstream of the low pressure turbine. It is comprised of an outer casing, an inner casing, and an inner cone that closes off the inner casing, and a strut or frame, which supports the rear bearing and centers the rotating assembly. *Table 3.19* contains the input and output details of the exhaust geometry.

Number of Struts		8
Strut Chord/Height		0.75
Strut Lean Angle		8
Gap Width/Height		0.2
Cone Angle [deg]		50
Cone Length/Inlet Radius		0.6
Casing Length/Inlet Radius		1.35
Inner Casing Thickness	in	0.07874(
Outer Casing Thickness	in	0.19685
Casing Material Density	lb/ft³	499.424
Exhaust Duct Mass Factor		1

Length	in	32.0089
Cone Length	in	6.5616
Outer Casing Mass	lbm	246.115
Strut Mass	lbm	111.408
Cone Mass	lbm	25.2371
Front Cover Mass	lbm	7.70538

The cone ends in the exhaust duct

Table 3.19: Exhaust Geometry Input & Output

The convergent-divergent nozzle is defined in *Table 3.20.* In both subsonic and supersonic operations, nozzle performance has a far larger impact on that of the overall system than any other component. The throat area A8 is usually choked and controls the flow through the whole engine. The expansion ratio A9/A8 determines how well the exhaust jet is expanded or how closely its static pressure matches the prevailing ambient value. Optimum thrust is produced when the pressure term in the thrust equation is slightly above zero. In *Table 3.4*, A9/A8 = 1.8 and the pressure term of the thrust equation is 1536 lbf, which tells us that A9 could have been larger except that the local diameter would then have exceeded that of the fan, leading to a non-cylindrical nacelle. So I left A9/A8 at 1.8, even though the jet Mach number of 2.046 in *Table 3.5* is rather meagre for a flight Mach number of 2.0. (OK, A9 should have been bigger!)

Geometry and mass are presented in *Table 3.20*. A net mass factor of 1.2 accounts for the specific controls and accessories used to activate the variable geometry in the nozzle, in keeping with normal industrial practice and is additional to the mass factor applied to the whole engine in *Sub-section 3.1.10*.

Standard/Plug Nozzle 1/2		1
Inl Section Length/Outer R		inactive
Conv Length/Inl Section R		0.75
Cone Angle [deg]		inactive
Cone Length/Inlet Radius		inactive
Inlet Section Area Ratio		inactive
Divergent Length/Throat R		1
Inner Casing Thickness	in	0.07874
Outer Casing Thickness	in	0.19685
Casing Material Density	lb/ft³	499.424
Nozzle Mass Factor		1.2

Overall Length	in	56.8435
Inlet Section Length	in	0
Convergent Length	in	16.988
Divergent Length	in	39.8555
Convergent Cone Angle [d		9.10632
Divergent Cone Angle [deg		9.69372
Inlet Section Mass	lbm	0
Convergent Section Mass	lbm	130.934
Divergent Section Mass	lbm	337.228
Inner Casing Mass	lbm	0
Outer Casing Mass	lbm	468.162
Total Mass	lbm	561.795

Table 3.20: Nozzle Input & Output

3.1.10 Overall Engine

LP Shaft Thickness	in	0.19685
HP Shaft Thickness	in	0.19685
Shaft Material Density	lb/ft³	499.424
LP Spool Design Spd Incr [		0
HP Spool Design Spd Incr [		0
Net Mass Factor		1.2173
Net Mass Adder	lbm	0

Front LP Shaft Cone Lengt	in	0.20195
Middle LP Shaft Length	in	85.0715
Middle LP Shaft Radius	in	1.34147
Rear LP Shaft Cone Length	in	2.00747
Front HP Shaft Cone Lengt	in	0
Rear HP Shaft Cone Length	in	12.1781
Rear HP Shaft Length	in	26.0843
Rear HP Shaft Radius	in	1.91105
Engine Length	in	235.503
Max Engine Diameter	in	56.5587
LP Shaft Mass	lbm	42.6904
HP Shaft Mass	lbm	54.7591
Net Mass	lbm	5749.89
Total Mass	lbm	6999.34
LP Spool Inertia	lb*in²	213392
HP Spool Inertia	lb*in²	113589

Table 3.21: Overall Engine Input & Output

Geometric details of the overall engine are provided in *Table 3.21*. Here we can see that application of a net mass factor of 1.2173 results in our overall target mass of 7000 lbm, when the nozzle is neglected. The net mass factor is reasonable allowance for the sub-systems and other miscellaneous items not included in our preliminary engine design.

3.2 Take-Off Conditions: Off-Design Operation

In *Section 1, Table 1.2,* the second "off-design" point was specified to be "End of Runway" take-off at sea level/Mach 0.3 (ISA +10°C) with a net thrust of 33,600 lbf. To address this, the design point cycle model with no reheat was run in the off-design mode to generate performance maps for the LPC, HPC, HPT and LPT. Reheat does not affect the maps. The operating conditions were then changed to rolling take-off and the model was run again. At that point, it was noticed that the LPC and HPC operating points beyond the scope of their maps, so I reverted to the design point in the off-design mode and scaled both compressor maps by moving the respective operating points to a more central location. On returning to the rolling take-off conditions, new maps were generated as shown in *Figures 3.7, 3.8, 3.9* and *3.10*. The original operating points at cruise are indicated by open round symbols and the off-design are represented by yellow squares. Aerodynamically, the turbines are more stable so no changes are needed to their maps for off-design operation of the *Olympus 593* engine model.

Reheat was then activated in the cycle design point model, using a nominal value of T7. It is now available to use at off-design. Returning to the off-design mode, the expansion ratio of the nozzle (A9/A8) was adjusted until optimum expansion was reached. Finally, T7 was adjusted until the net thrust target was achieved.

The resulting output summary for the rolling take-off case is shown in *Table 3.22*.

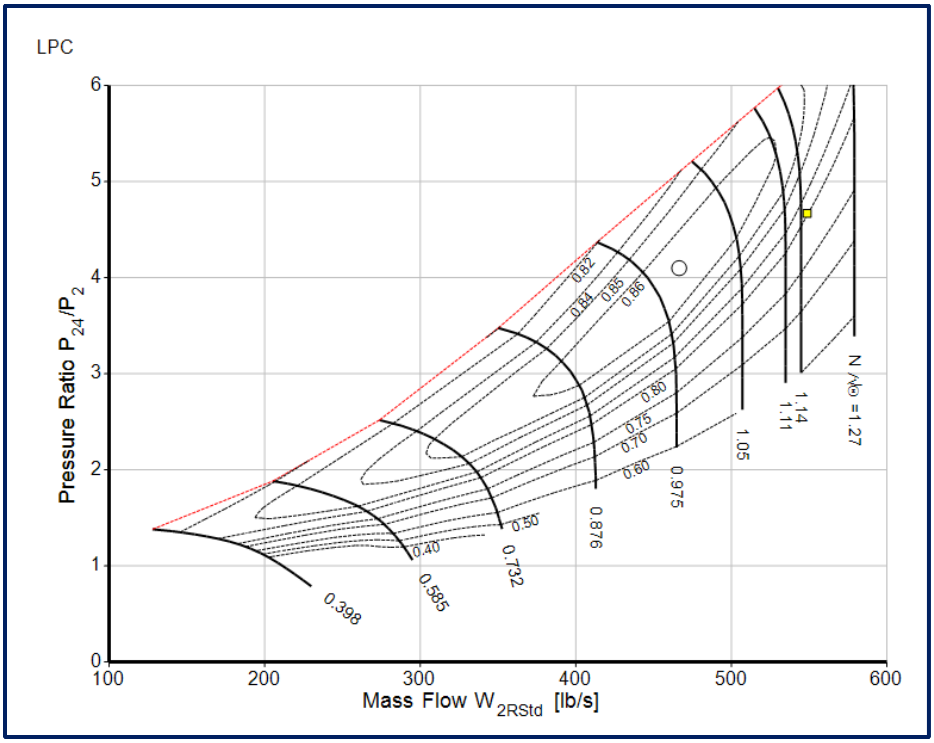

Figure 3.7: Olympus 593 Baseline Engine LPC Map at EoR Take-Off

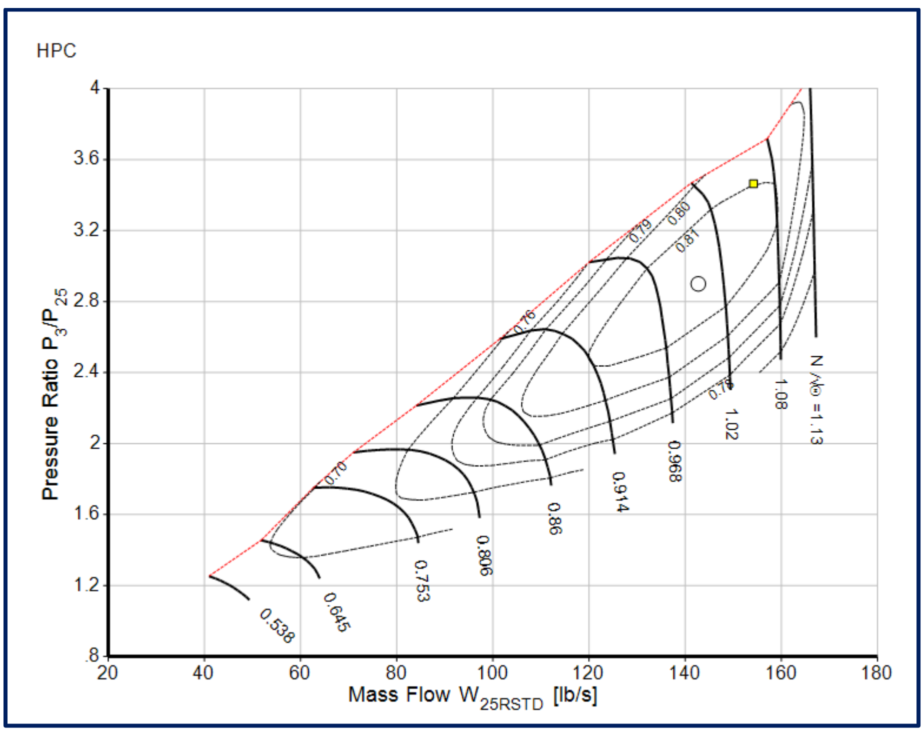

Figure 3.8: Olympus 593 Baseline Engine HPC Map at EoR Take-Off

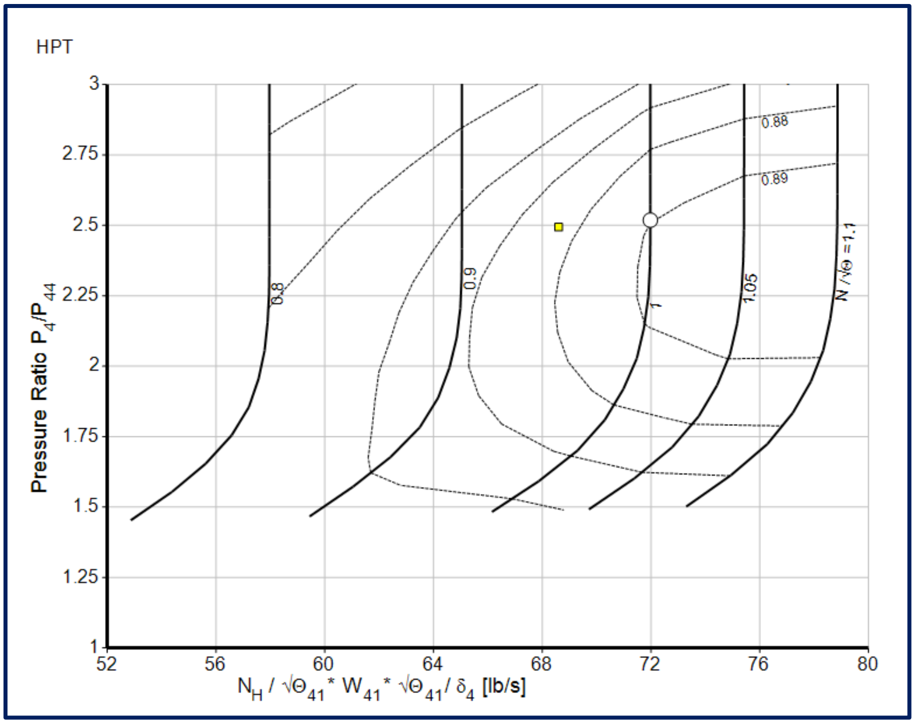

Figure 3.9: Olympus 593 Baseline Engine HPT Map at EoR Take-Off

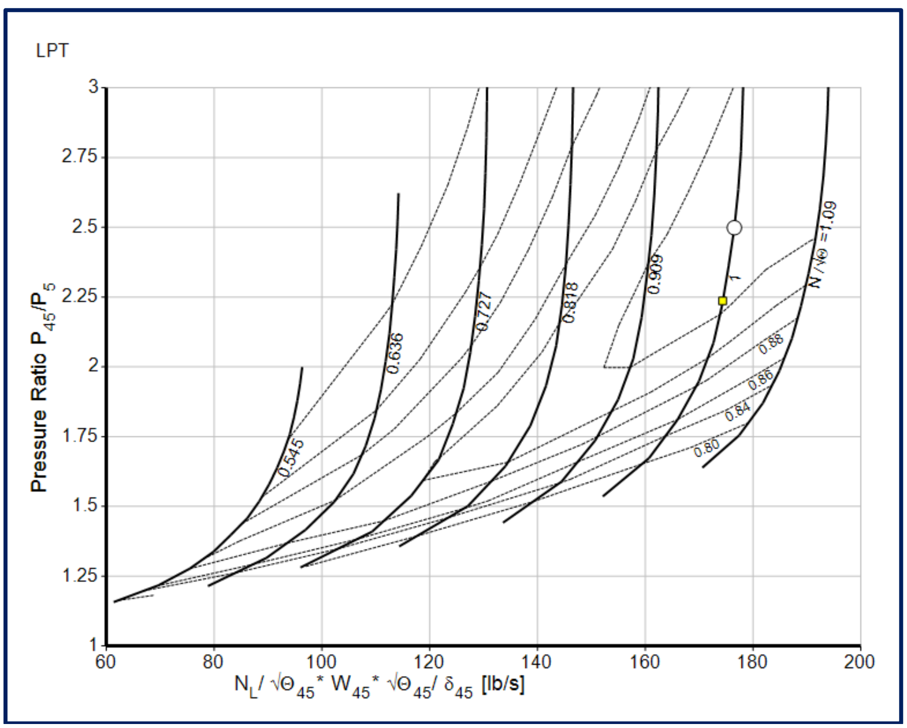

Figure 3.10: Olympus 593 Baseline Engine LPT Map at EoR Take-Off

```
        W          T          P          WRstd         Reheat on
Station  lb/s       R         psia        lb/s          FN          =   33611.07 lb
 amb               518.67    14.696                    TSFC        =     1.2576 lb/(lb*h)
  1    542.651    528.15    15.656                    FN/W2       =    1992.82 ft/s
  2    542.651    528.14    14.670     548.571        WF Burner=     8.50050 lb/s
 24    542.651    888.91    68.505                    WF total =    11.74174
 25    542.651    888.91    67.708     154.192        P2/P1       =     0.9370
  3    542.651   1330.38   234.550      54.453        P25/P24     =     0.9884
 31    482.959   1330.38   234.550                    P3/P2       =    15.9889
  4    491.460   2433.78   226.343      69.121        P45/P44     =     0.9803
 41    518.592   2380.00   226.343      72.127        P6/P5       =     0.9838
 43    518.592   1971.83    90.759                    P16/P6      =     1.7503
 44    545.725   1941.49    90.759                    P16/Ps6     =     1.9800
 45    545.725   1941.49    88.971     174.400        W_NGV/W25=     0.05000
 49    545.725   1626.84    39.782                    WHcl/W25 =     0.05000
  5    545.725   1626.84    39.782     357.035        WLcl/W25 =     0.00000
  6    545.725   1626.84    39.139                    XM6         =     0.43286
 61    545.725    903.80    39.139                    XM61        =     0.43286
  7    249.007   1977.50    37.935                    XM7         =     0.52284
  8    548.966   1977.50    37.935     415.254        A8          =    1284.74 in²
 13      0.000    888.91    68.505                    BypBld      =     0.00000 lb/s
 16      0.000    888.91    68.505                    WclNozzle=     0.00000 lb/s
Bleed    5.427   1330.38   234.550                    WBld/W2     =     0.01000
                                                       BPR         =     0.0000
-------------------------------------------           Ang8        =      19.47 °
Efficiencies:    isentr   polytr    RNI     P/P       CD8         =     0.96106
 LP Compressor   0.8018   0.8388   0.977   4.670      P8/Pamb     =     2.58132
 HP Compressor   0.8109   0.8391   2.424   3.464      WlkLP/W25=     0.00000
 Burner          0.9916                     0.965      Loading     =      89.67 %
 HP Turbine      0.8789   0.8666   2.591   2.494      e444 th     =     0.85388
 LP Turbine      0.9024   0.8931   1.285   2.236      WlkO/W25 =     0.00000
 Reheat          0.9327                     0.969      far7        =     0.02186
-------------------------------------------           P5/P2       =     2.7119 EPR
HP Spool mech Eff 0.9900   Speed     9737 rpm         PWX         =      100.0 hp
LP Spool mech Eff 0.9900   Speed     5812 rpm         Core Eff =     0.3320
-------------------------------------------           Prop Eff =     0.2554
Con-Di Nozzle:                                         A9/A8       =     1.05000
 A9*(Ps9-Pamb)      364.338                            CFGid       =     0.96034
-------------------------------------------
hum [%]     war0       FHV      Fuel
   0.0    0.00000    18552.4    Generic

Input Data File:
C:\Concorde Re-Engine Project\GasTurb13 Files\Olympus593_RTO_Scaled_Reheat_5Aug2
020.C2J
```

Table 3.22: Olympus 593 Baseline Engine Output Summary at EoR Take-Off

4. Hints & Suggestions

- Even though this document has been prepared in Imperial units, you may carry out the project and submit your proposal in SI units if you prefer.

- You should first replicate the *Olympus 593* baseline engine model with whatever software that you will use for your new engine design. Your results may not match the baseline model exactly but will enable you to make a valid comparison of weights and performance for your new concept.

- The efficiencies of the turbomachinery components may be assumed to be the same as those of the baseline engine and be input directly or obtained via the "calculate efficiency" mode of whatever software you are using.

- Use **military specification MIL-E-5007** a current general estimate of the characteristics of an oblique shock system, to determine inlet recovery in your new engine design

$$\frac{P_2}{P_1} = 1.0 - 0.075(M - 1)^{1.35}$$

where M is the flight Mach number.

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

Even though the date for submission of *Letters of Intent* is stated as November 1, 2020 on pages 34 and 36, it is recommended that teams who know that they will enter the competition inform AIAA and Dr. Ian Halliwell (ianhalliwell@earthlink.net) as soon as possible, so that assistance may be given and access to design codes may be arranged, where appropriate (See page 33).

Questions will be taken by volunteers from the *AIAA Air Breathing Propulsion Technical Group*, whose contact information will be provided to teams who submit a *letter of intent*.

5. Competition Expectations

The existing rules and guidelines for the *AIAA Foundation Student Design Competition* should be observed and these are provided in *Appendix 2*. 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).

- 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.

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.

<u>Proposals should be limited to fifty pages, which will not include the administrative/contents or the "signature" pages.</u>

References

1. *"A Case Study by Aerospatiale and British Aerospace on the Concorde"*
   Jean Rech and Clive S. Leyman
   AIAA Professional Study Series.

2. "*Future SST Engines with particular reference to Olympus 593 Evolution and Concorde Experience."*
   P.H. Calder and P.C Gupta
   SAE 751056, presented at National Aerospace Engineering and Manufacturing Meeting, Culver City, Los Angeles. November 1975.

3. *"GasTurb 13: A Design & Off-Design Performance Program for Gas Turbines"*
   <http://www.gasturb.de>
   Joachim Kurzke, GasTurb GmbH. 2018.

4. "*Users' Manual for Updated Computer Code for Axial Flow Compressor Conceptual Design*"
   Arthur J. Glassman
   NASA Contractor Report 189171, 1992

5. "*A Simple Correlation of Turbine Efficiency*"
   S. F. Smith
   Journal of the Royal Aeronautical Society. Volume 69. 1965.

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. "*Aircraft Propulsion – Second Edition*"
Saeed Farokhi
Wiley, 2014.

7. "*The Jet Engine*"
Rolls-Royce plc. 2005.

8. "*Propulsion and Power – An Exploration of Gas Turbine Performance Modeling*"
Joachim Kurzke and Ian Halliwell
Springer, 2018.

Available Software and Additional Reference Material

"NPSS® Academic Edition (www.npssconsortium.org): Numerical Propulsion System Simulation® (NPSS®) proudly sponsors the AIAA Undergraduate Engine Design Competition, with the hope to help students develop valuable skills for the aerospace industry. An academic version of the NPSS software is available for free to all students throughout the world. NPSS is the industry standard for aerospace engine cycle design, analysis, and system integration. Primary applications include aerospace systems, but it can also be used for modeling rocket propulsion cycles, Rankine and Brayton cycles, refrigeration cycles, and electrical systems. A copy of the newly released NPSS Integrated Development Environment (IDE) is available for students participating in the AIAA Undergraduate Engine Design Competition." ***NPSS®***

GasTurb13 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 performance model to be generated, with stressed disks and component weights. A student license for this code is available directly strictly for academic work. A free 30-day license may also be down-loaded. **(http://www.gasturb.de)**

AxSTREAM EDU™ by SoftInWay Inc. **(http://www.softinway.com)** AxSTREAM® is a turbomachinery design, analysis, and optimization software suite used by many of the world's leading aerospace companies developing new and innovative aero engine technology. AxSTREAM EDU™ enables students to work on the design of propulsion and power generation systems. AxCYCLE™, an add-on to AxSTREAM EDU™ addresses cycle design and analysis. Participants in the AIAA Undergraduate Team Engine Design Competition can acquire an AxSTREAM EDU™ license via the following steps:

- Submit a Letter of Intent to AIAA
- Once the letter of intent has been received and approved, names of team members will be recognized as being eligible to be granted access to the AxSTREAM EDU™ software by AIAA.
- Students must then contact the RFP author, who will then arrange for SoftInWay to grant the licenses.

In addition to the software, students will also gain free access to STU, SoftInWay's online self-paced video course platform with various resources and video tutorials on both turbomachinery fundamentals.

The offers above are subject to *ITAR* restrictions.

Appendix 1. Letter of Intent

2020/2021
Joint AIAA–IGTI Undergraduate Team Engine Design Competition

Request for Proposal:

Candidate Engines for a Supersonic Business jet

Title of Design Proposal: ____

Name of School: ____

Designer's Name	**AIAA or ASME**	**Graduation Date**	**Degree**
______________	__________	__________	____________
Team Leader			
Team Leader E-mail			
______________	__________	__________	____________
______________	__________	__________	____________
______________	__________	__________	____________
______________	__________	__________	____________

In order to be eligible for the 2020/21 AIAA Engine Design Competition for Undergraduate Teams, you must complete this form, the "Letter of Intent", and return it by February 12, 2021 via www.aiaa-awards, as noted in *Appendix 2, Section III*, "Schedule and Activity Sequences." For any non-member listed above, a student member application and member dues payment to AIAA should also be included with this form or submitted to ASME, with a note attached.

____

Signature of Faculty Advisor   Signature of Project Advisor   Date

____

Faculty Advisor – Printed   Project Advisor – Printed   Date

Appendix 2. Rules and Guidelines

I. General Rules

1. All undergraduate AIAA branch or at-large Student Members are eligible and encouraged to participate.

2. Teams will be groups of **not more than four** AIAA branch or at-large Student Members per entry.

3. Proposals must be submitted in MS Word or Adobe PDF format also via www.aiaa-awards. Total size of the file(s) cannot exceed 60 MB, which must also fit on 100 pages when printed. The file title should include the team name and/or university. **A "Signature" page must be included in the report and indicate all participants, including faculty and project advisors, along with their AIAA member numbers**. Designs that are submitted must be the work of the students, but guidance may come from the Faculty/Project Advisor and should be accurately acknowledged. **Graduate student participation in any form is prohibited**.

4. Design projects that are used as part of an organized classroom requirement are eligible and encouraged for competition.

5. More than one design may be submitted from students at any one school.

6. If a design group withdraws their project from the competition, the team chairman must notify AIAA Headquarters immediately!

7. Judging will be in two parts.
   - First, the written proposals will be assessed by the judging panel comprised of members of AIAA organizing committees from industrial and government communities.
   - Second, the best three teams will be invited to present their work to a second judging panel at a special technical session at the AIAA Propulsion and Energy Forum, Denver, CO, August 9 - 11, 2021. The results of the presentations will be combined with the earlier scores to determine first, second and third places.

8. Certificates will be presented to the winning design teams for display at their university and a certificate will also be presented to each team member and the faculty/project advisor. The finishing order will be announced immediately following the three presentations.

II. 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.

III. Schedule and Sequence of Activities

Significant activities, dates, and addresses for submission of proposal and related materials are as follows:

A. Letter of Intent – February 12, 2021
B. Receipt of Proposal – May 14, 2021
C. Proposal evaluations completed - June 30, 2021
D. Round 2 Proposal Presentations & Announcement of Winners at the AIAA Propulsion and Energy Forum; August 9 - 11, 2021.

Teams intending to submit a proposal must submit a one page Letter of Intent along with the signed attached Intent Form (Item A) on or before the date specified above by February 12, 2021 to: www.aiaa-awards.

For further information, please contact Michael Lagana, AIAA University Programs Manager at **MichaelL@AIAA.org.**

A pdf file of the proposal must be received at the same address on or before the date specified above for the Receipt of Proposal (Item B).

IV. Proposal Requirements

The technical proposal is the most important criterion 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:

1. Demonstrate a thorough understanding of the Request for Proposal (RFP) requirements.

2. 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.

3. 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.

4. Include tradeoff studies performed to arrive at the final design.

5. Provide a description of automated design tools used to develop the design.

V. Basis for Judging

Round 1: Proposal

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 independence of thinking or a fresh approach to the project. Does the method and treatment of the problem show imagination? Does the approach 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.

Round 2: Presentation

Each team will have 30 minutes to present a summary of its proposal to the judging panel. In addition to the categories above, the presentations will be assessed for clarity, effectiveness and the ability to sell the teams' ideas. Scores from the presentation will be added to those from the proposal. The presentation score will be adjusted so that it is worth 30% of the overall value.