AIAA Foundation Student Design Competition 2019 - 2020
Undergraduate Team – Engine

Candidate Engines for a Supersonic Business Jet

Request for Proposal

March 10, 2019

Abstract

New engine designs are solicited for a supersonic business jet that will travel from Europe to North America and back within one business day. Entry into service is expected to be around 2030. The new aircraft must cruise at Mach 1.15 over land, without producing a sonic boom on the ground, but it also must operate at Mach 0.98 in order to offer a cost-per-mile similar to competing subsonic private jets. Over water, it must be possible to cruise at Mach 2.1 and at this speed, with a range 4,600 miles, the aircraft will cross the Atlantic in less than two hours and burn less than 96,000 pounds of fuel.

The supersonic business jet is currently powered by two low bypass ratio turbofan engines, each with a nominal net thrust of 21,700 lbf (96.53 kN) at sea level take-off. The challenges of successful commercial operation are quite substantial for any gas turbine engine, however, light weight, low take-off noise, reduced emissions especially at high altitude and affordable fares are paramount. Candidate engines should be lighter than the current power plant and have an improved fuel burn so that the payload and/or operating altitude may be increased and the range may be extended.

A generic model of the current power plant is supplied. Responders should generate a typical, multi-segment, mission that addresses the above-listed general improvements specifically and covers design point and off-design engine operations. The performance and total fuel consumption of the candidate engine should be estimated over the mission and stated clearly in the proposal. Special attention should be paid to engine mass, dimensions & integration with the aircraft. Technical feasibility and operating costs should also be addressed.

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1. Introduction

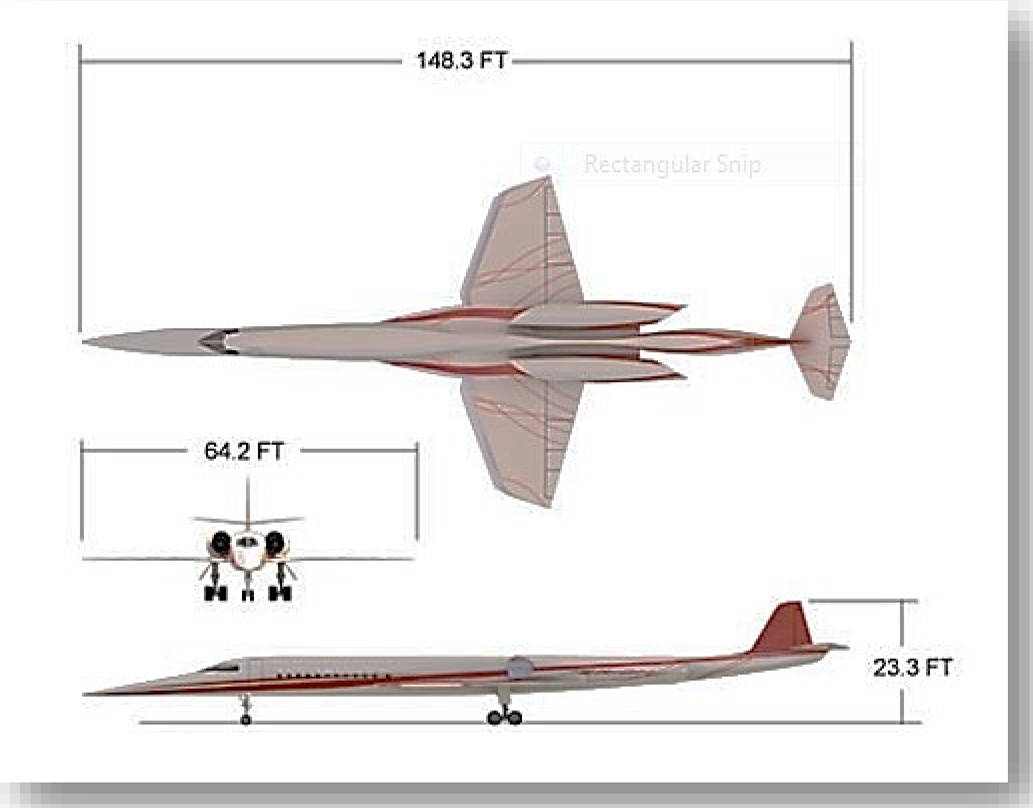


Figure 1: The Supersonic Business Jet

The supersonic business jet under consideration should allow non-stop travel from Europe to North America and back within one business day. It has a plan form which is similar in wing and tail shape and arrangement to the *F-104 Starfighter*. It will cruise at Mach 1.15 over land without producing a sonic boom on the ground. The plane can also cruise at Mach 0.98, offering a similar cost-per-mile to subsonic private jets. Over water, however, Mach 2.4 should be achievable. At such a speed, with a range 4,600 miles and burning 97,400 pounds of fuel, the aircraft should be able to cross the Atlantic in less than two hours. Some relevant aircraft characteristics are given in *Table 1*.

Aircraft Specifications

General characteristics	
Crew	2
Capacity	8 – 12 passengers
Length	135.6 ft (41.33 m)
Wing span	64.2 ft (19.57 m)
Height	21.2 ft (6.46 m)
Wing area	1,200 ft² (111.5 m²)
Max. take-off weight	146,000 lbm (40,823 kg)
Power plant	2 × low bypass ratio turbofans; 21,700 lbf (96.53 kN) each
Performance	
Maximum speed	1,720 knots (Mach 3; 1980 mph; 3186 km/h)
Cruise speed	1204 knots (Mach 2.1; 1386 mph; 2230 km/h) @ 40kft
Range	At Mach 0.95: 4,600 nm (5,300 mi; 8,500 km)
Service ceiling	51,000 ft (16,000 m)

Table 1: Some General Characteristics of the Supersonic Business Jet Aircraft

The radical design of the wings also brings much lower noise emissions, and according to the manufacturer, the plane will operate within the most stringent noise limitations. The current engines are described in a generic model, given in Section 3. Aircraft dimensions are given in Figure 1, from which the overall nacelle length may be estimated to be 34 feet.

At take-off the total thrust needed from the two engines is 43,400 lbf (193.06 kN). In-flight engine thrust requirements in kN are summarized in *Figure 2*. Additional data may be generated between Mach 0.2 and 0.98.

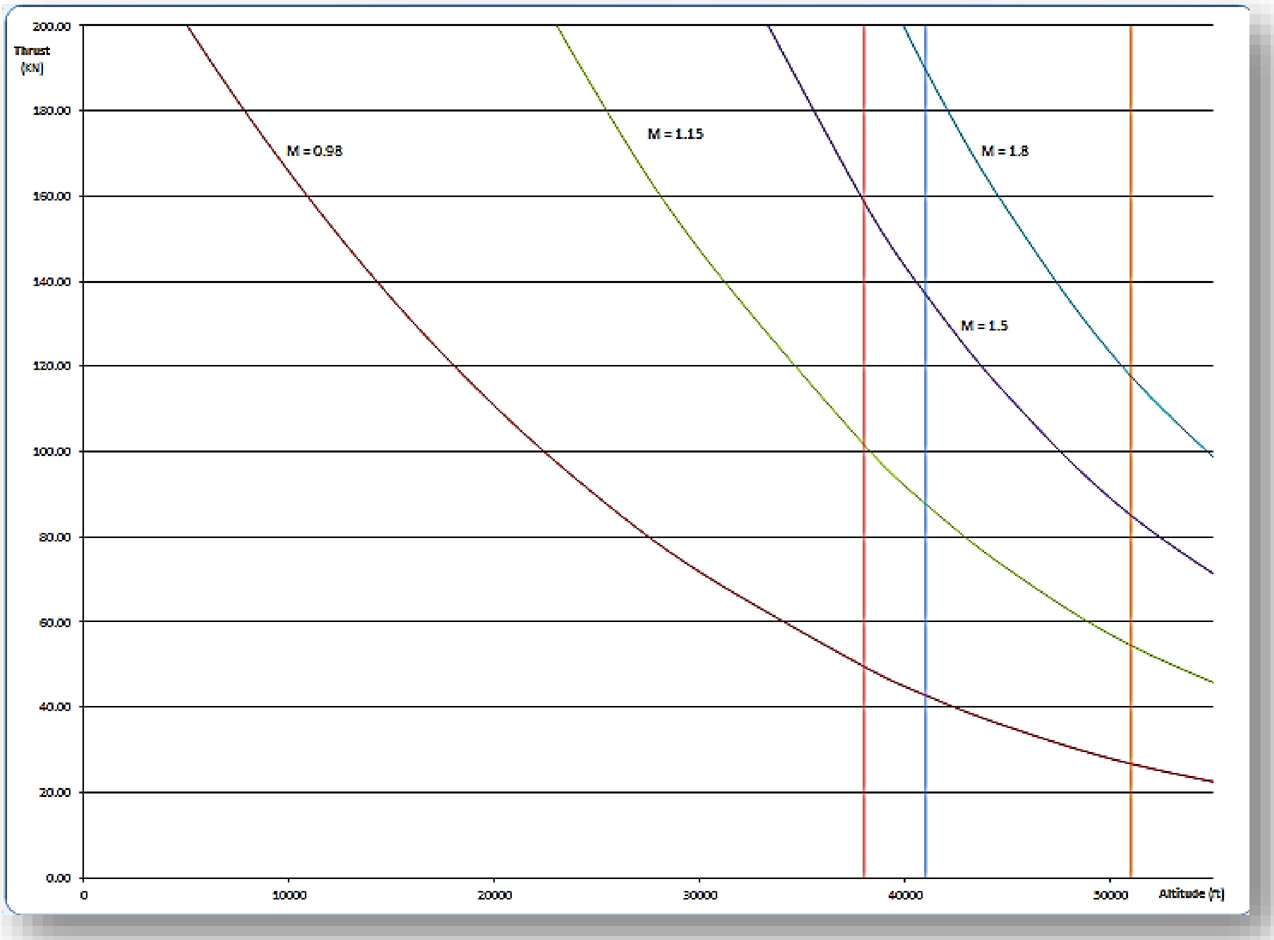


Figure 2: In-Flight Thrust Requirements

2. Design Objectives & Requirements

- A new engine design is required for a future version of the reference supersonic business jet, with an entry-into-service date of 2025.

- The current flight envelope ranges from take-off at static sea-level conditions to supersonic cruise at 40,000 feet/Mach 2.1. This is to be retained for the new engine, so these two flight conditions should be used as the principal design points for candidate engines. Take-off thrust should match that of the baseline engine described later. It is hoped that the endurance might be extended by reducing the fuel consumption and minimizing engine mass.

- The generic baseline engine model should be used as a starting point, and the new design should be optimized for minimum engine mass & fuel burn, based on trade studies to determine the best combination of fan pressure ratio, bypass ratio, overall pressure ratio and turbine entry temperature. 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.

- Based on the entry into service date, the development of new materials and an increase in design limits may be assumed. So let us set a new limit of 2840 R for turbine entry temperature. The development and potential application of carbon matrix composites is of particular interest (*Reference 1*). 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.

- Different engine architecture is permitted, but accommodation within the existing nacelle envelope is preferred.

- An appropriate inlet must be designed. A 2-ramp, either axisymmetric or 2-dimensuional 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.

- 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.

- Since reduced specific fuel consumption does not necessarily lead to reduced fuel consumption, additional credit will be awarded for determination of fuel burn over an assumed mission by dividing it into suitable segments in terms of time at altitude and Mach number and summing the incremental fuel burn estimates.

3. Baseline Engine Model

As stated previously, the baseline engine is a low bypass ratio turbofan. A generic model has been generated from publically-available information *(Reference 2)* using *GasTurb12*. 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.

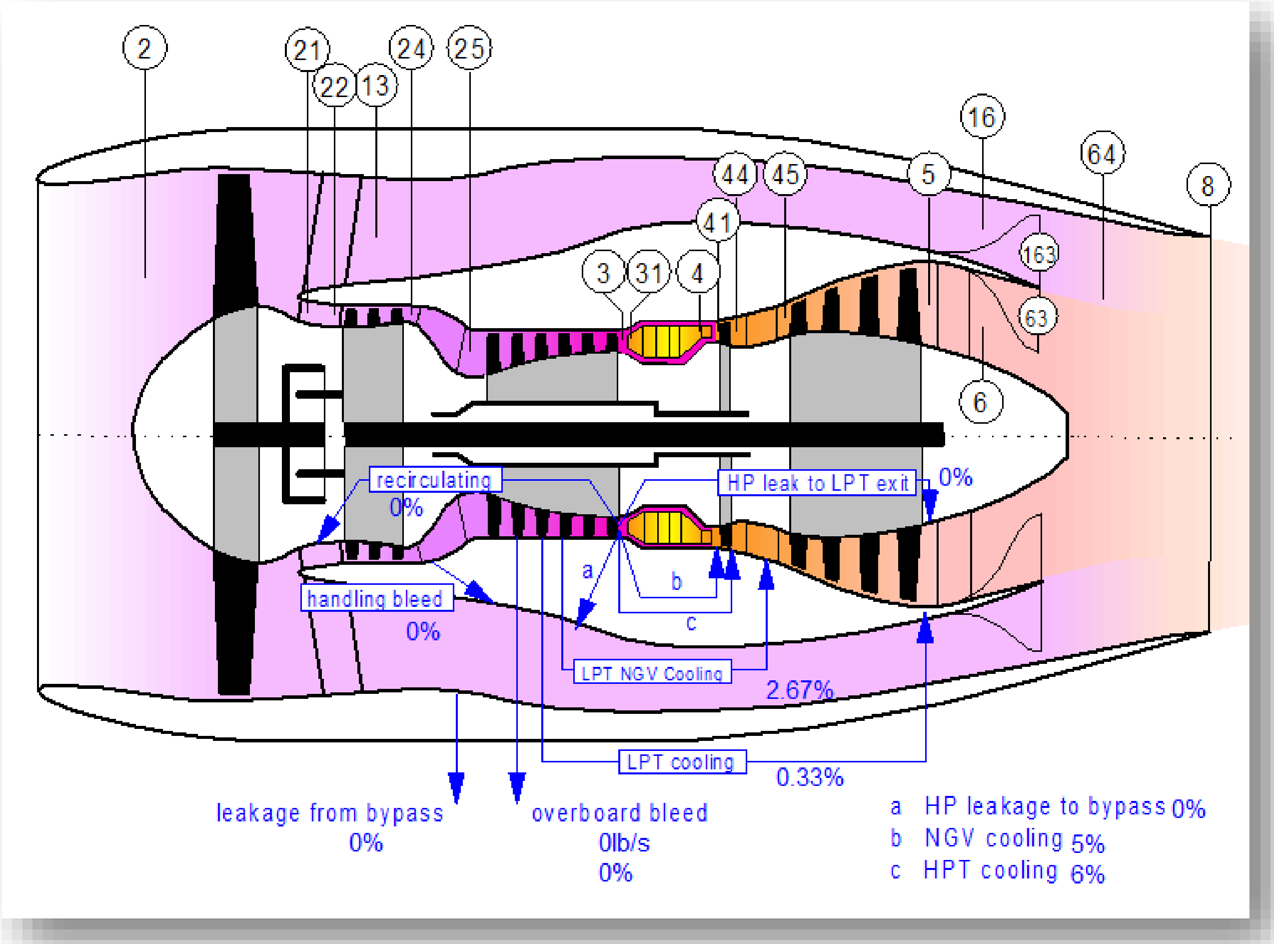


Figure 3: A Mixed Flow, High Bypass Ratio, Turbofan Engine Schematic with Calculation Stations & Cooling Flows

Figure 3 contains a general schematic with relevant station numbers. (In *GasTurb12*, setting the gear ratio to unity eliminates the gear box.)

3.1 Overall Characteristics

Table 2 contains a summary of basic engine characteristics, taken directly from *Reference 2*.

Design Features of the Baseline Engine	
Engine Type	Axial, turbofan
Number of fan/booster/compressor stages	2, 6, 7
Number of HP/LP turbine stages	1, 3
Combustor type	Annular
Maximum net thrust at sea level	21,700 lbf
Specific fuel consumption at max. power	0.519 lbm/hr/lbf
Overall pressure ratio at max. power	21.0
Max. envelope diameter	49.2 inches
Max. envelope length	154.1 inches
Dry weight less tail-pipe	4,515 lbm

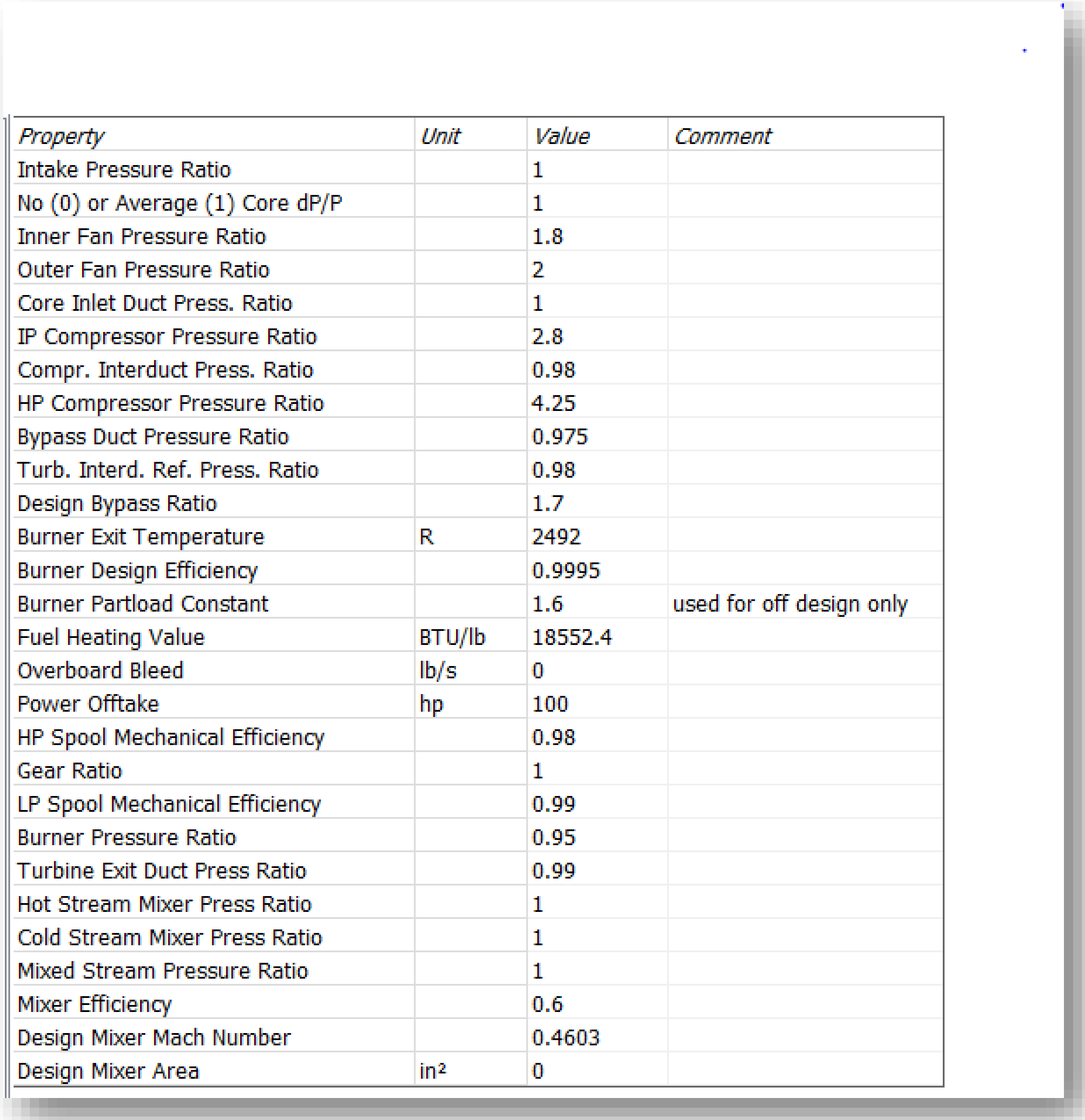
Table 2: Baseline Engine: Basic Data, Overall Geometry & Performance

Major Design Parameters

In a turbofan engine, the primary four design variables are turbine entry temperature (T_4), overall pressure ratio (*OPR* or P_3/P_2), fan pressure ratio (*FPR* or P_{21}/P_2) and bypass ratio (*BPR*). We usually differentiate between the fan pressure ratios in the core & bypass streams.

Table 3 is the "Basic Input" for the *GasTurb12* model of the baseline engine. Of the four primary design variables, only the overall pressure ratio is given explicitly in *Table 3*. To generate an acceptable replica of the engine, a unique combination of the remainder must be estimated iteratively using performance figures which are provided – namely the net thrust (F_N) and specific fuel consumption (*sfc*) at static sea level take-off conditions - as targets. By definition, this operating condition also corresponds to the engine design point, but this may not be the case for your new engine.

Table 3 above contains most of the primary input parameters for the engine cycle. Some of the secondary inputs are also discussed here while the rest are covered below. The first row of *Table 3* assumes negligible total pressure loss between the inlet leading edge and the fan face. The inner and outer fan pressure ratios are then selected separately; there is more blade speed at the fan tip than at its hub, so the inner & outer fan pressure ratios have been set at 1.8 & 2.0 respectively – fairly aggressive but not unreasonable for a modern single-stage machine. A 2% total pressure loss is then accounted for in the duct between the fan and the IP compressor or booster. Knowing that the required overall pressure ratio is 21.0, results in a pressure ratio across the remainder of the compression system of 11.667, allowing for losses. This is distributed between the booster and the HP compressor with 2.8 across the former (over 6 stages) and 4.25 across the latter (over 7 stages). Next a 2.5% total pressure loss is assumed in the bypass duct, followed by an inter-turbine duct loss of 2%.



Property	Unit	Value	Comment
Intake Pressure Ratio		1	
No (0) or Average (1) Core dP/P		1	
Inner Fan Pressure Ratio		1.8	
Outer Fan Pressure Ratio		2	
Core Inlet Duct Press. Ratio		1	
IP Compressor Pressure Ratio		2.8	
Compr. Interduct Press. Ratio		0.98	
HP Compressor Pressure Ratio		4.25	
Bypass Duct Pressure Ratio		0.975	
Turb. Interd. Ref. Press. Ratio		0.98	
Design Bypass Ratio		1.7	
Burner Exit Temperature	R	2492	
Burner Design Efficiency		0.9995	
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		0.98	
Gear Ratio		1	
LP Spool Mechanical Efficiency		0.99	
Burner Pressure Ratio		0.95	
Turbine Exit Duct Press Ratio		0.99	
Hot Stream Mixer Press Ratio		1	
Cold Stream Mixer Press Ratio		1	
Mixed Stream Pressure Ratio		1	
Mixer Efficiency		0.6	
Design Mixer Mach Number		0.4603	
Design Mixer Area	in²	0	

Table 3: Basic Input

Continuing with the input description, the design bypass ratio was set at 1.7. A value of 2492˚R for the turbine entry temperature was taken as being reasonable for a relatively modern version of this engine with limited cooling capacity and an expected long life for the HP turbine (say 5,000 hours). In fact, this value of T_4 was the result of an iterative process that involved turbomachinery efficiencies and the target thrust. The next four parameters relate to the primary combustor; they are all fairly conventional values by modern standards. The burner "*part load constant*" is an element in the calculation of burner efficiency that is discussed in the *GasTurb12 User Guide* in *Reference 4*. Without expert knowledge, this is best left alone! The remaining parameters in *Table 3* may be considered as secondary influences and are discussed briefly below.

Secondary Design Parameters

Cooling Air: Mention has already been made of bleed and cooling air flows – the secondary flows. Only the overboard bleed is listed in *Table 3* (although this is in fact zero), however the secondary flows indicated in *Figure 2* have been set via another "air system" tab on the input screen as fractions of W_{25}, the HP compressor entry flow.

Pressure Losses: A number of total pressure losses, mentioned earlier, are also specified in *Table 3* by inserting the appropriate pressure ratios across the inter-compressor duct, the inter-turbine duct, the mixer and the primary combustor.

Turbomachinery Efficiencies: Efficiencies of the fan, HP compressor, HP turbine and LP turbine are entered via their respective tabs on the input screen. The values are not listed specifically in *Table 3*, but may be reviewed in the output summary presented later in *Table 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! Both values appear in the output summary in *Table 4*. However, another option is available that has been used here. It allows *GasTurb12* to estimate turbine efficiencies from data supplied - values of stage loading and flow coefficients - which are then used in a *Smith Chart (Reference 4)*, assuming an equal work spilt between stages. It is recommended that this be used.

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. This is often preferred to the use of a separate auxiliary power unit, depending on how much power is required for airframe use. 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 engine. This may be excessive and you may choose to reduce it for your engine and mission!

Mixer Efficiency: 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 & 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 mechanically 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.

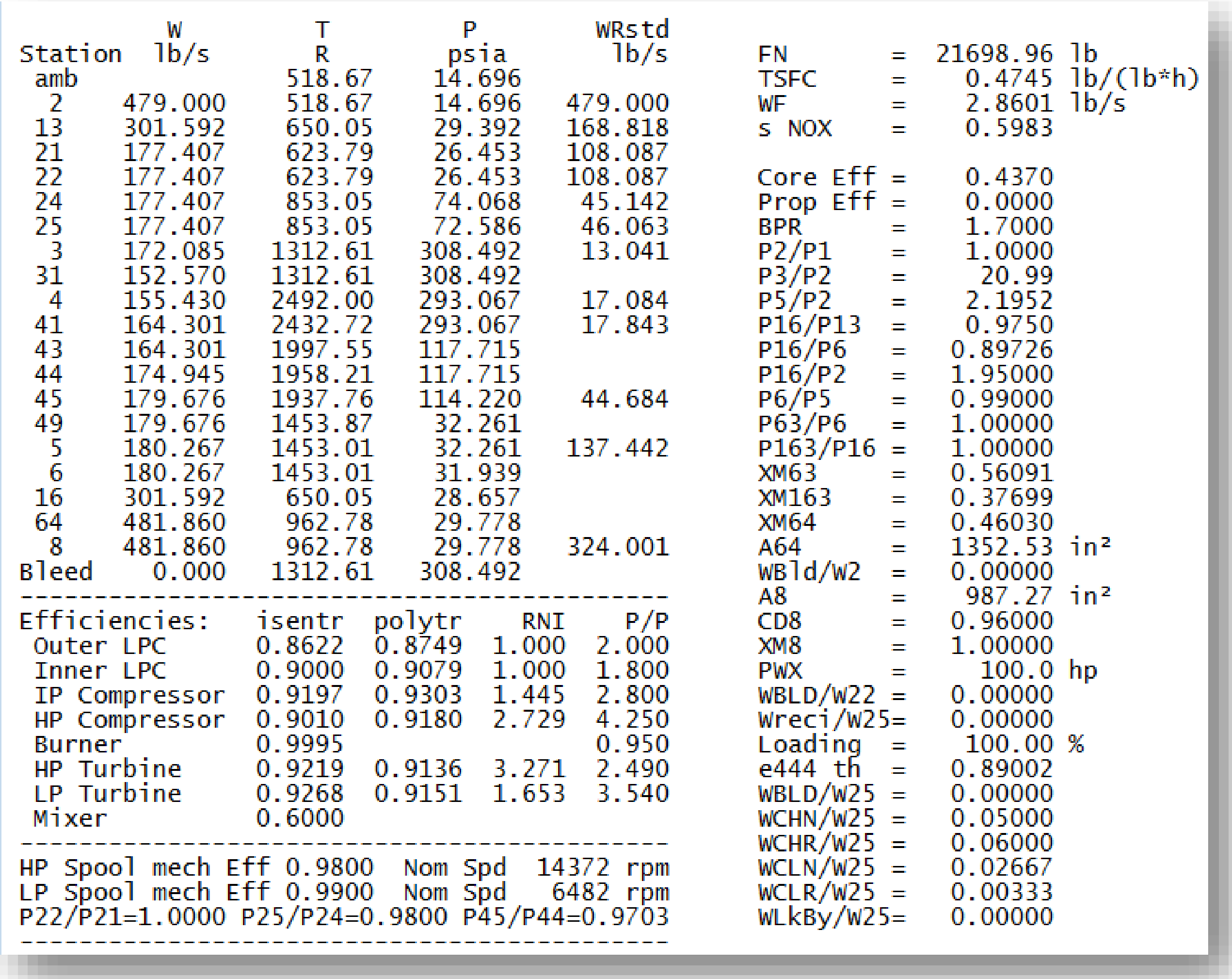
A limited study has been made of the influence of a number of secondary parameters and it was determined that the default values present in the *GasTurb12* generic model should be retained, based on the known expertise of the author of the code.

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 of 21,700 lbf given in *Table 2* once the engine scale has been determined. We also have a target dimensional envelope to fit into, namely a maximum casing diameter of 49.2 inches and a maximum length of 154.1 inches. The diameter can be determined via the mass flow rate; 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 fan. These values are input on the primary input screen under the LP compressor tab, where a fairly aggressive Mach number of 0.619 was found to be appropriate. This sets the general radial dimension for the complete engine, although in fact downstream of the fan, the entry radius of the compressor is also determined by an input radius ratio. 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. The overall diameter of the model turned out to be 143.3 inches – 4 inches too large. The engine model length of 143.3 inches appears to be slightly shorter than the target but definition of this dimension, taken from *Reference 3*, is open to interpretation!

Materials & Weights

As far as possible, use was made of the materials database in the *GasTurb12* design code. 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, although it should be noted they may not yet have found their way into the baseline engine, where long life and reliability are critical. However, within the component models, material densities can be modified independently of the database and I have taken advantage of this feature in some cases where I believe that "advanced" materials of lower density are appropriate. Use has also been made of the materials data in *Reference 6*, interpolating and extrapolating where necessary.



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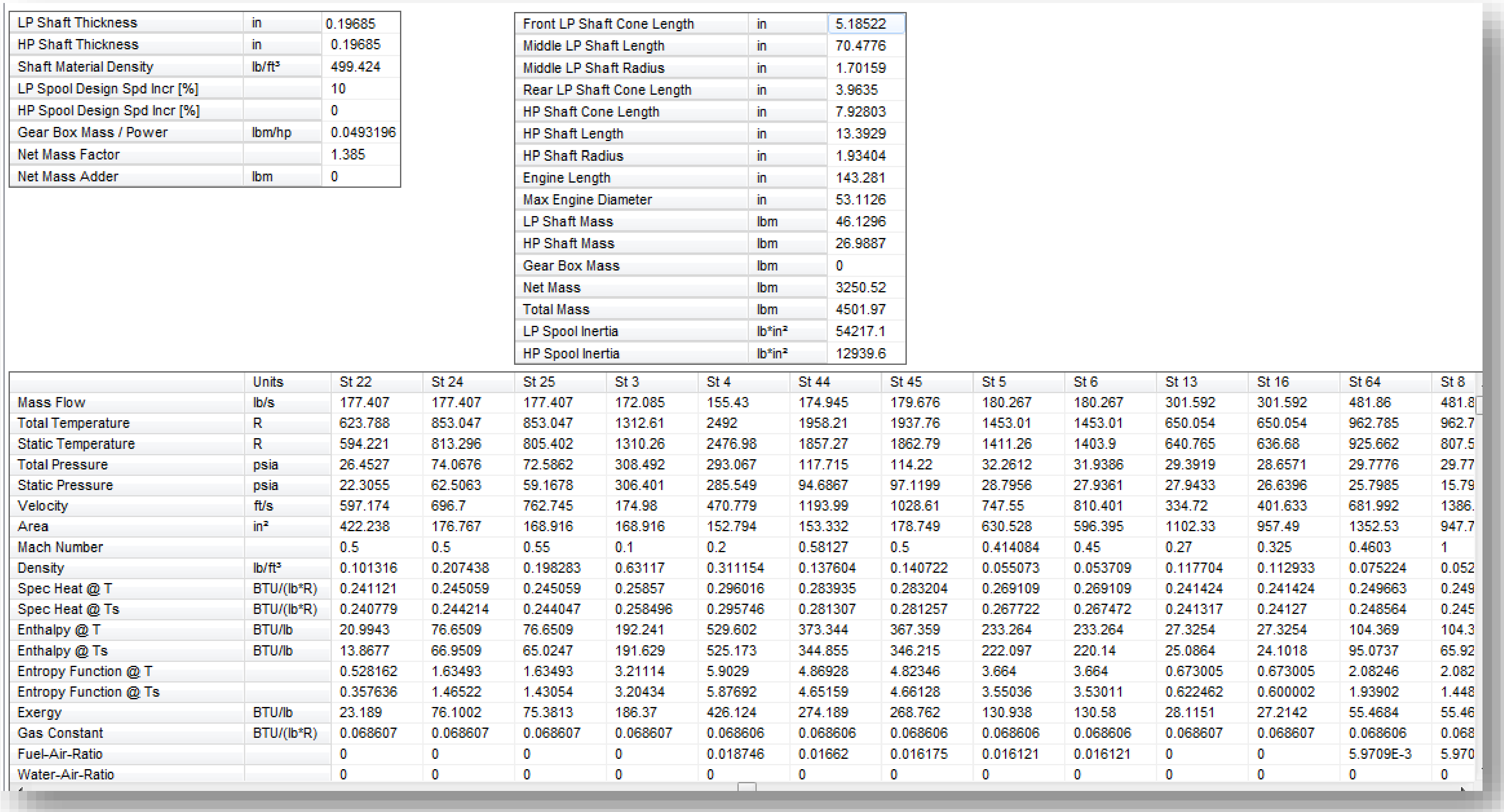
             W         T         P      WRstd
Station   lb/s         R      psia       lb/s      FN       =  21698.96 lb
 amb              518.67    14.696                 TSFC     =    0.4745 lb/(lb*h)
  2    479.000    518.67    14.696   479.000      WF       =    2.8601 lb/s
 13    301.592    650.05    29.392   168.818      s NOX    =    0.5983
 21    177.407    623.79    26.453   108.087
 22    177.407    623.79    26.453   108.087      Core Eff =    0.4370
 24    177.407    853.05    74.068    45.142      Prop Eff =    0.0000
 25    177.407    853.05    72.586    46.063      BPR      =    1.7000
  3    172.085   1312.61   308.492    13.041      P2/P1    =    1.0000
 31    152.570   1312.61   308.492                P3/P2    =     20.99
  4    155.430   2492.00   293.067    17.084      P5/P2    =    2.1952
 41    164.301   2432.72   293.067    17.843      P16/P13  =    0.9750
 43    164.301   1997.55   117.715                P16/P6   =   0.89726
 44    174.945   1958.21   117.715                P16/P2   =   1.95000
 45    179.676   1937.76   114.220    44.684      P6/P5    =   0.99000
 49    179.676   1453.87    32.261                P63/P6   =   1.00000
  5    180.267   1453.01    32.261   137.442      P163/P16 =   1.00000
  6    180.267   1453.01    31.939                XM63     =   0.56091
 16    301.592    650.05    28.657                XM163    =   0.37699
 64    481.860    962.78    29.778                XM64     =   0.46030
  8    481.860    962.78    29.778   324.001      A64      =   1352.53 in²
Bleed    0.000   1312.61   308.492                WBld/W2  =   0.00000
--------------------------------------------      A8       =    987.27 in²
Efficiencies:    isentr  polytr    RNI    P/P     CD8      =   0.96000
 Outer LPC       0.8622  0.8749  1.000  2.000     XM8      =   1.00000
 Inner LPC       0.9000  0.9079  1.000  1.800     PWX      =     100.0 hp
 IP Compressor   0.9197  0.9303  1.445  2.800     WBLD/W22 =   0.00000
 HP Compressor   0.9010  0.9180  2.729  4.250     Wreci/W25=   0.00000
 Burner          0.9995                 0.950     Loading  =    100.00 %
 HP Turbine      0.9219  0.9136  3.271  2.490     e444 th  =   0.89002
 LP Turbine      0.9268  0.9151  1.653  3.540     WBLD/W25 =   0.00000
 Mixer           0.6000                           WCHN/W25 =   0.05000
--------------------------------------------      WCHR/W25 =   0.06000
HP Spool mech Eff 0.9800  Nom Spd  14372 rpm      WCLN/W25 =   0.02667
LP Spool mech Eff 0.9900  Nom Spd   6482 rpm      WCLR/W25 =   0.00333
P22/P21=1.0000 P25/P24=0.9800 P45/P44=0.9703      WLkBy/W25=   0.00000
--------------------------------------------
```

Table 4: Baseline Engine Output Summary

In *GasTurb12* component weights are calculated by multiplying the effective volumes by the corresponding material densities. Of course, only the major elements which are directly 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 the engine industry, this is done by the application of a multiplier or adder whose value is based on decades of experience. In general, a multiplication factor of 1.3 is recommended in the *GasTurb12* manual, but I increased this to a "*net mass factor*" of 1.385 in *Table 5*, and I also used additional factors for individual components mainly because it got me closer to the gross engine weight I was looking for! The total mass of the engine shown in *Table 5* (4502.2 lbm) corresponds reasonably closely to the 4515 lbm target in *Table 2*, when the mass of the tail pipe is accounted for.

Table 4 is the “Output Summary Table” from *Gasturb12* for the baseline engine and *Table 5* is a more detailed “overall output table”.



LP Shaft Thickness	in	0.19685
HP Shaft Thickness	in	0.19685
Shaft Material Density	lb/ft³	499.424
LP Spool Design Spd Incr [%]		10
HP Spool Design Spd Incr [%]		0
Gear Box Mass / Power	lbm/hp	0.0493196
Net Mass Factor		1.385
Net Mass Adder	lbm	0

Front LP Shaft Cone Length	in	5.18522
Middle LP Shaft Length	in	70.4776
Middle LP Shaft Radius	in	1.70159
Rear LP Shaft Cone Length	in	3.9635
HP Shaft Cone Length	in	7.92803
HP Shaft Length	in	13.3929
HP Shaft Radius	in	1.93404
Engine Length	in	143.281
Max Engine Diameter	in	53.1126
LP Shaft Mass	lbm	46.1296
HP Shaft Mass	lbm	26.9887
Gear Box Mass	lbm	0
Net Mass	lbm	3250.52
Total Mass	lbm	4501.97
LP Spool Inertia	lb*in²	54217.1
HP Spool Inertia	lb*in²	12939.6

	Units	St 22	St 24	St 25	St 3	St 4	St 44	St 45	St 5	St 6	St 13	St 16	St 64	St 8
Mass Flow	lb/s	177.407	177.407	177.407	172.085	155.43	174.945	179.676	180.267	180.267	301.592	301.592	481.86	481.8
Total Temperature	R	623.788	853.047	853.047	1312.61	2492	1958.21	1937.76	1453.01	1453.01	650.054	650.054	962.785	962.7
Static Temperature	R	594.221	813.296	805.402	1310.26	2476.98	1857.27	1862.79	1411.26	1403.9	640.765	636.68	925.662	807.5
Total Pressure	psia	26.4527	74.0676	72.5862	308.492	293.067	117.715	114.22	32.2612	31.9386	29.3919	28.6571	29.7776	29.77
Static Pressure	psia	22.3055	62.5063	59.1678	306.401	285.549	94.6867	97.1199	28.7956	27.9361	27.9433	26.6396	25.7985	15.79
Velocity	ft/s	597.174	696.7	762.745	174.98	470.779	1193.99	1028.61	747.55	810.401	334.72	401.633	681.992	1386.
Area	in²	422.238	176.767	168.916	168.916	152.794	153.332	178.749	630.528	596.395	1102.33	957.49	1352.53	947.7
Mach Number		0.5	0.5	0.55	0.1	0.2	0.58127	0.5	0.414084	0.45	0.27	0.325	0.4603	1
Density	lb/ft³	0.101316	0.207438	0.198283	0.63117	0.311154	0.137604	0.140722	0.055073	0.053709	0.117704	0.112933	0.075224	0.052
Spec Heat @ T	BTU/(lb*R)	0.241121	0.245059	0.245059	0.25857	0.296016	0.283935	0.283204	0.269109	0.269109	0.241424	0.241424	0.249663	0.249
Spec Heat @ Ts	BTU/(lb*R)	0.240779	0.244214	0.244047	0.258496	0.295746	0.281307	0.281257	0.267722	0.267472	0.241317	0.24127	0.248564	0.245
Enthalpy @ T	BTU/lb	20.9943	76.6509	76.6509	192.241	529.602	373.344	367.359	233.264	233.264	27.3254	27.3254	104.369	104.3
Enthalpy @ Ts	BTU/lb	13.8677	66.9509	65.0247	191.629	525.173	344.855	346.215	222.097	220.14	25.0864	24.1018	95.0737	65.92
Entropy Function @ T		0.528162	1.63493	1.63493	3.21114	5.9029	4.86928	4.82346	3.664	3.664	0.673005	0.673005	2.08246	2.082
Entropy Function @ Ts		0.357636	1.46522	1.43054	3.20434	5.87692	4.65159	4.66128	3.55036	3.53011	0.622462	0.600002	1.93902	1.448
Exergy	BTU/lb	23.189	76.1002	75.3813	186.37	426.124	274.189	268.762	130.938	130.58	28.1151	27.2142	55.4684	55.46
Gas Constant	BTU/(lb*R)	0.068607	0.068607	0.068607	0.068607	0.068606	0.068606	0.068606	0.068606	0.068606	0.068607	0.068607	0.068606	0.068
Fuel-Air-Ratio		0	0	0	0	0.018746	0.01662	0.016175	0.016121	0.016121	0	0	5.9709E-3	5.970
Water-Air-Ratio		0	0	0	0	0	0	0	0	0	0	0	0	0

Table 5: Baseline Engine Detailed Output

A cutaway of the baseline engine is shown in *Figure 3*.

A plot of the *GasTurb12* baseline engine model appears in *Figure 4*.

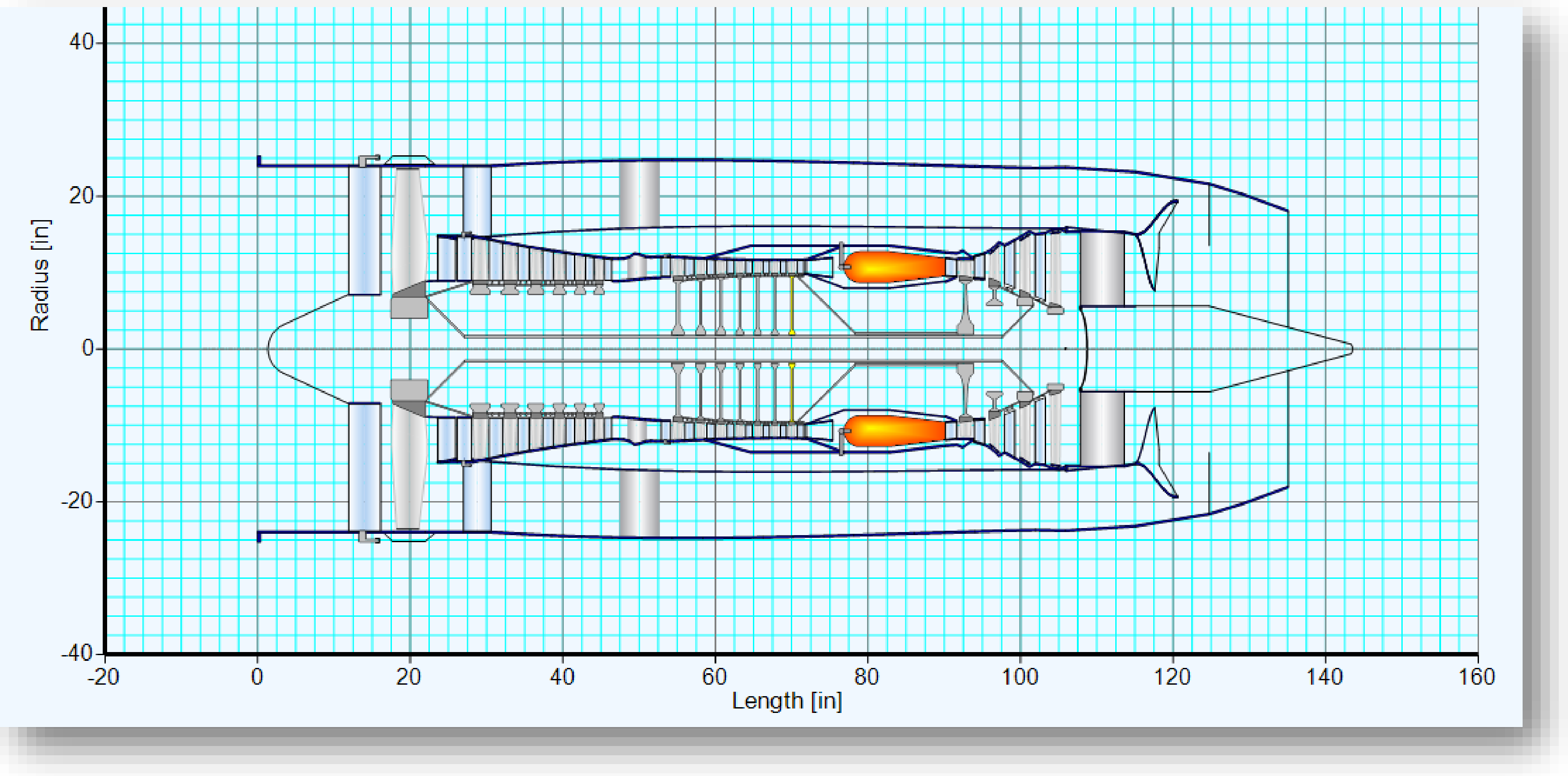
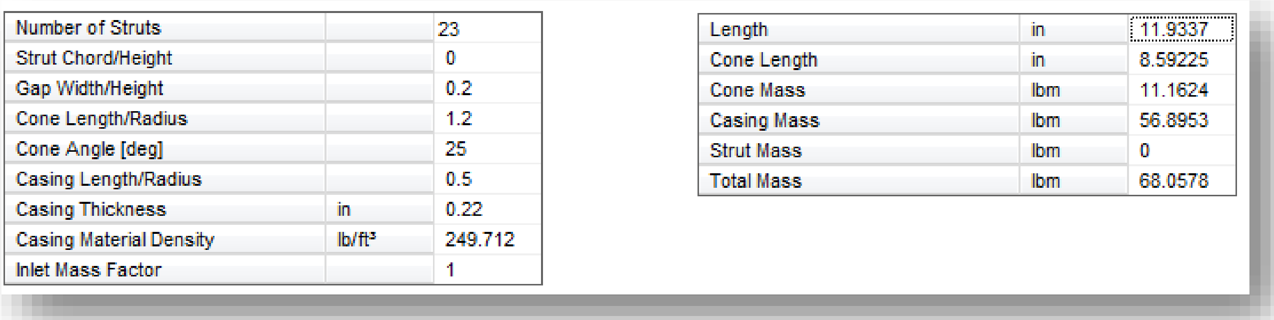


Figure 4: GasTurb12 Model of the Baseline Engine

Some details of the component models now follow.

3.2 Inlet

The inlet is designed with a rounded center body (see *Figure 4*). In practice, a single-stage fan can be cantilevered from a bearing located in the main frame of the engine. The outer diameter of the inlet has been determined from that of the fan.



Number of Struts		23
Strut Chord/Height		0
Gap Width/Height		0.2
Cone Length/Radius		1.2
Cone Angle [deg]		25
Casing Length/Radius		0.5
Casing Thickness	in	0.22
Casing Material Density	lb/ft³	249.712
Inlet Mass Factor		1

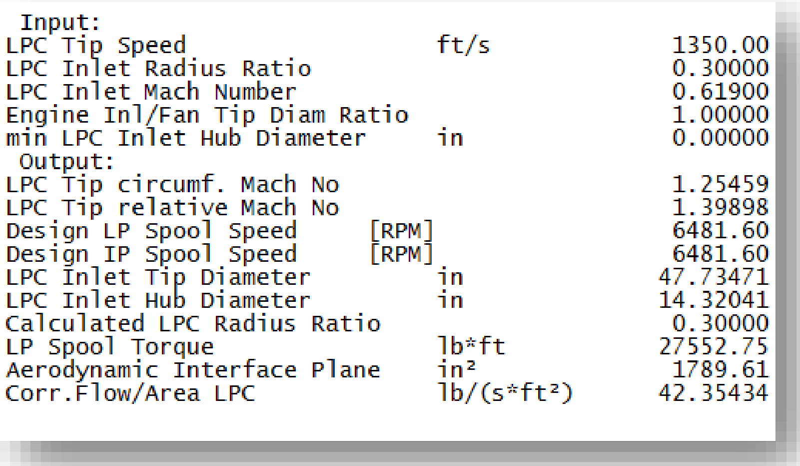
Length	in	11.9337
Cone Length	in	8.59225
Cone Mass	lbm	11.1624
Casing Mass	lbm	56.8953
Strut Mass	lbm	0
Total Mass	lbm	68.0578

Table 6: Inlet Design

Pertinent characteristics of the inlet are shown in *Table 6*. At 68 lbm, the inlet is fairly light and this is because, based on the density, we have taken a typical *Ti-Al* alloy as our choice of materials. This should accommodate the dynamic heating effects of high-speed operation. It is noteworthy that the *GasTurb* "inlet" is merely the portion of the casing (plus center body) immediately upstream of the fan. The *GasTurb12* model begins at the "upstream flange".

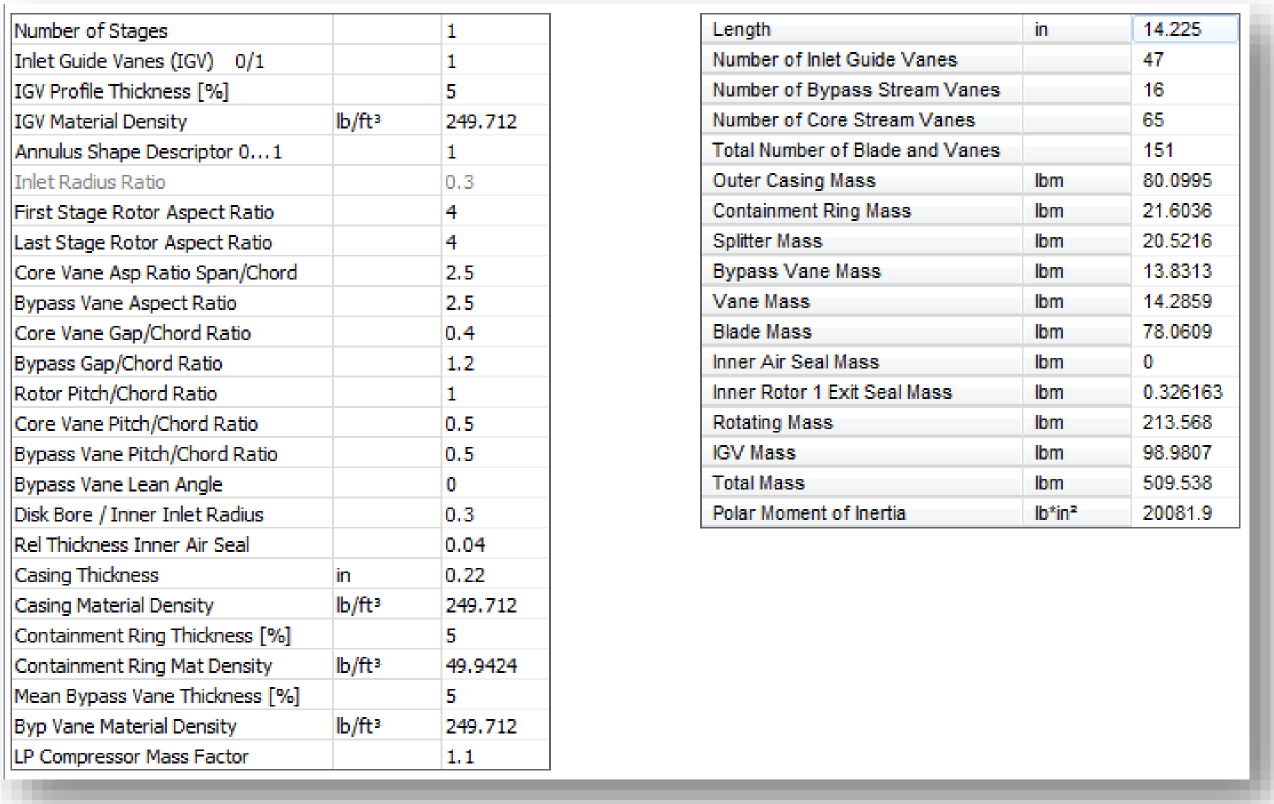
3.3 Fan

The fan characteristics are given in *Tables 7 & 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 (42.35 lbm/ft^2) is fairly aggressive and corresponds to the input value of Mach number (0.61).



```
 Input:
LPC Tip Speed                  ft/s          1350.00
LPC Inlet Radius Ratio                       0.30000
LPC Inlet Mach Number                        0.61900
Engine Inl/Fan Tip Diam Ratio                1.00000
min LPC Inlet Hub Diameter     in            0.00000
 Output:
LPC Tip circumf. Mach No                     1.25459
LPC Tip relative Mach No                     1.39898
Design LP Spool Speed     [RPM]              6481.60
Design IP Spool Speed     [RPM]              6481.60
LPC Inlet Tip Diameter         in           47.73471
LPC Inlet Hub Diameter         in           14.32041
Calculated LPC Radius Ratio                  0.30000
LP Spool Torque                lb*ft        27552.75
Aerodynamic Interface Plane    in²           1789.61
Corr.Flow/Area LPC             lb/(s*ft²)   42.35434
```

Table 7: Fan: Detailed Overview

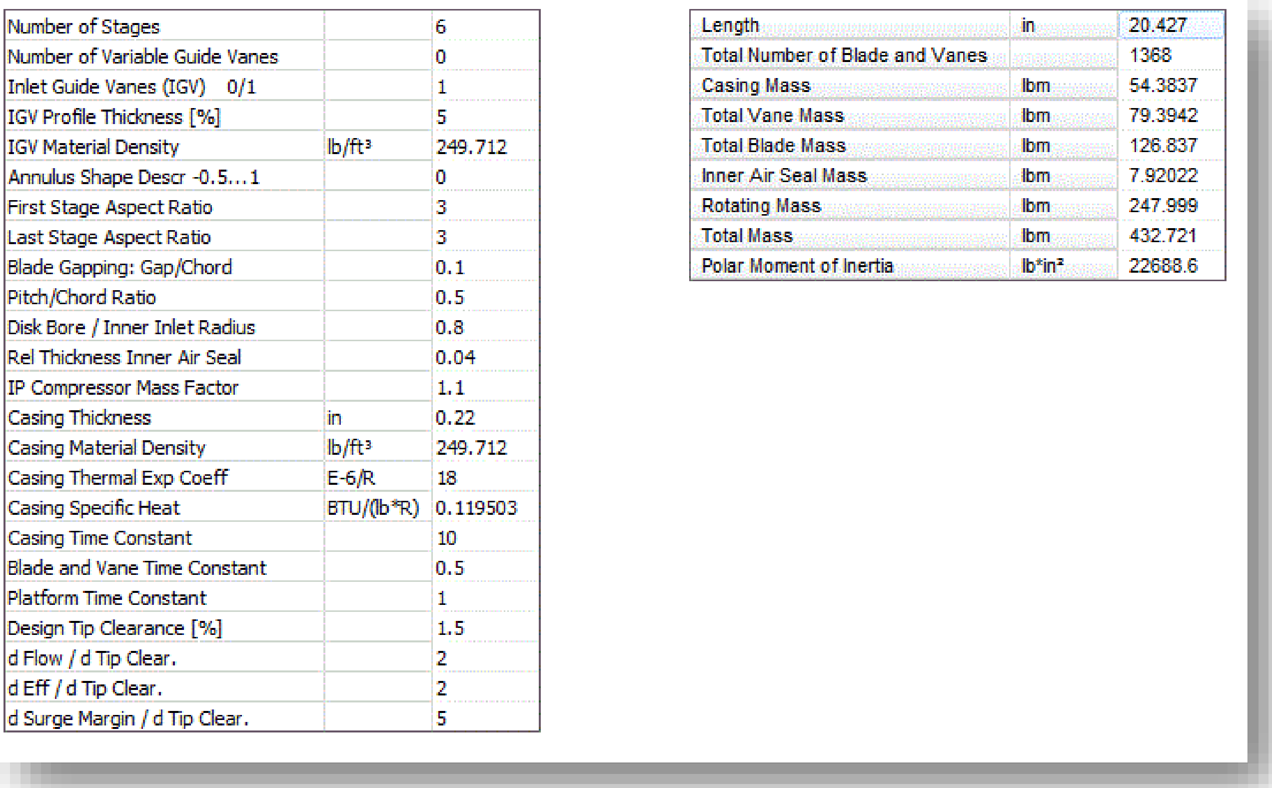


Number of Stages		1
Inlet Guide Vanes (IGV) 0/1		1
IGV Profile Thickness [%]		5
IGV Material Density	lb/ft³	249.712
Annulus Shape Descriptor 0…1		1
Inlet Radius Ratio		0.3
First Stage Rotor Aspect Ratio		4
Last Stage Rotor Aspect Ratio		4
Core Vane Asp Ratio Span/Chord		2.5
Bypass Vane Aspect Ratio		2.5
Core Vane Gap/Chord Ratio		0.4
Bypass Gap/Chord Ratio		1.2
Rotor Pitch/Chord Ratio		1
Core Vane Pitch/Chord Ratio		0.5
Bypass Vane Pitch/Chord Ratio		0.5
Bypass Vane Lean Angle		0
Disk Bore / Inner Inlet Radius		0.3
Rel Thickness Inner Air Seal		0.04
Casing Thickness	in	0.22
Casing Material Density	lb/ft³	249.712
Containment Ring Thickness [%]		5
Containment Ring Mat Density	lb/ft³	49.9424
Mean Bypass Vane Thickness [%]		5
Byp Vane Material Density	lb/ft³	249.712
LP Compressor Mass Factor		1.1

Length	in	14.225
Number of Inlet Guide Vanes		47
Number of Bypass Stream Vanes		16
Number of Core Stream Vanes		65
Total Number of Blade and Vanes		151
Outer Casing Mass	lbm	80.0995
Containment Ring Mass	lbm	21.6036
Splitter Mass	lbm	20.5216
Bypass Vane Mass	lbm	13.8313
Vane Mass	lbm	14.2859
Blade Mass	lbm	78.0609
Inner Air Seal Mass	lbm	0
Inner Rotor 1 Exit Seal Mass	lbm	0.326163
Rotating Mass	lbm	213.568
IGV Mass	lbm	98.9807
Total Mass	lbm	509.538
Polar Moment of Inertia	lb*in²	20081.9

Table 8: Fan General Output

3.4 Booster

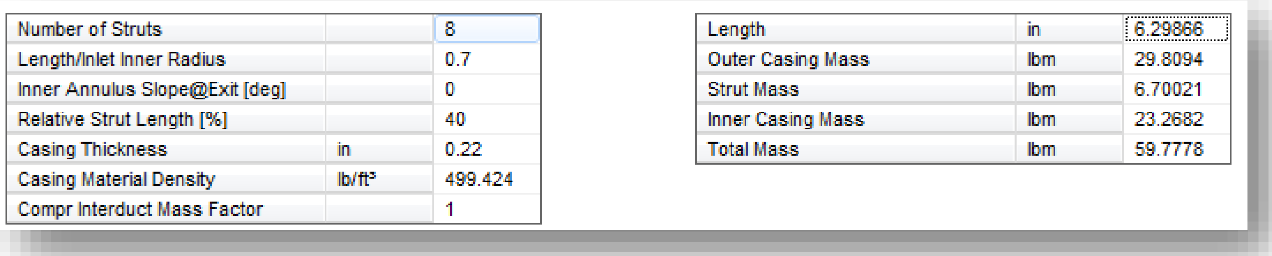


Number of Stages		6
Number of Variable Guide Vanes		0
Inlet Guide Vanes (IGV) 0/1		1
IGV Profile Thickness [%]		5
IGV Material Density	lb/ft³	249.712
Annulus Shape Descr -0.5...1		0
First Stage Aspect Ratio		3
Last Stage Aspect Ratio		3
Blade Gapping: Gap/Chord		0.1
Pitch/Chord Ratio		0.5
Disk Bore / Inner Inlet Radius		0.8
Rel Thickness Inner Air Seal		0.04
IP Compressor Mass Factor		1.1
Casing Thickness	in	0.22
Casing Material Density	lb/ft³	249.712
Casing Thermal Exp Coeff	E-6/R	18
Casing Specific Heat	BTU/(lb*R)	0.119503
Casing Time Constant		10
Blade and Vane Time Constant		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 Clear.		5

Length	in	20.427
Total Number of Blade and Vanes		1368
Casing Mass	lbm	54.3837
Total Vane Mass	lbm	79.3942
Total Blade Mass	lbm	126.837
Inner Air Seal Mass	lbm	7.92022
Rotating Mass	lbm	247.999
Total Mass	lbm	432.721
Polar Moment of Inertia	lb*in²	22688.6

Table 9: Booster - General Output

3.5 Inter-Compressor Duct



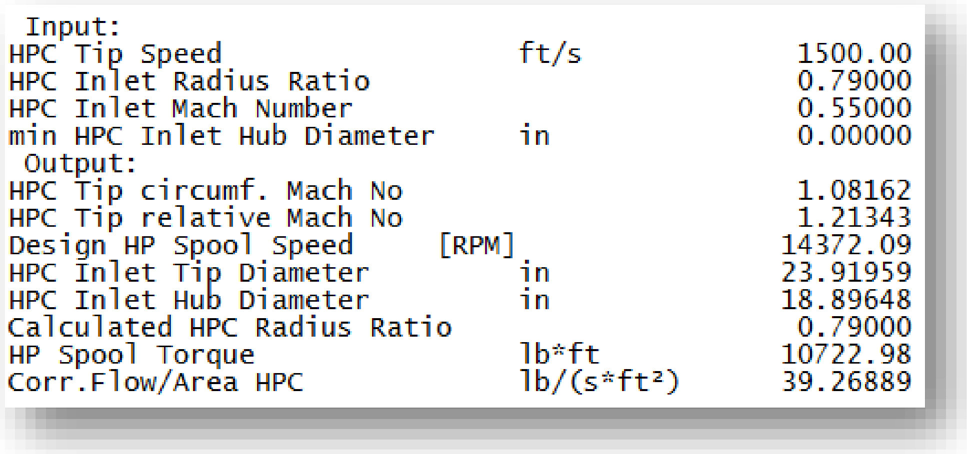
Number of Struts		8
Length/Inlet Inner Radius		0.7
Inner Annulus Slope@Exit [deg]		0
Relative Strut Length [%]		40
Casing Thickness	in	0.22
Casing Material Density	lb/ft³	499.424
Compr Interduct Mass Factor		1

Length	in	6.29866
Outer Casing Mass	lbm	29.8094
Strut Mass	lbm	6.70021
Inner Casing Mass	lbm	23.2682
Total Mass	lbm	59.7778

Table 10: 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, although this remains at a value of unity for the inter-compressor duct since very little of the structure is left unaccounted for in the simple model.

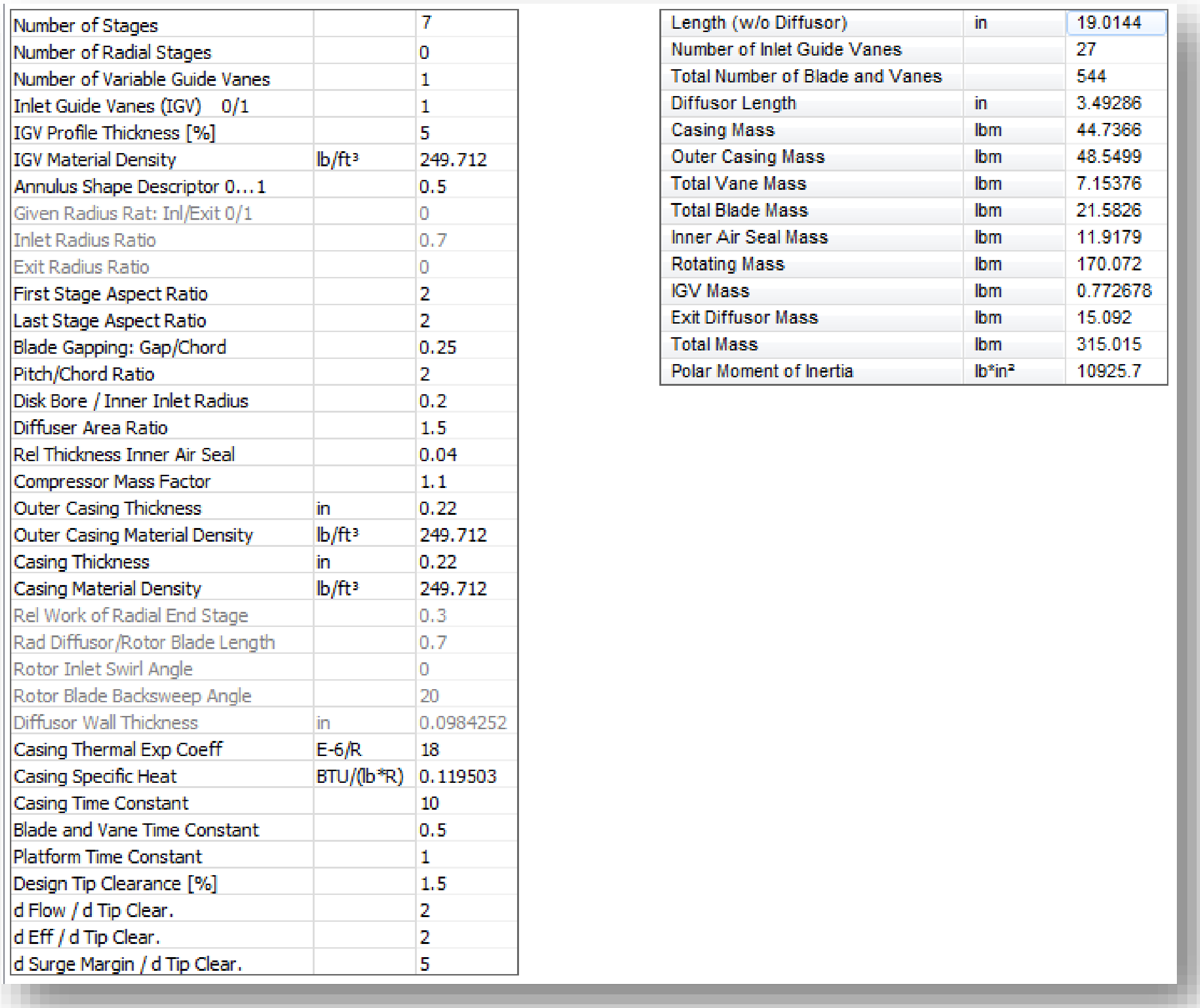
3.6 High Pressure Compressor



```
 Input:
HPC Tip Speed                  ft/s            1500.00
HPC Inlet Radius Ratio                         0.79000
HPC Inlet Mach Number                          0.55000
min HPC Inlet Hub Diameter     in              0.00000
 Output:
HPC Tip circumf. Mach No                       1.08162
HPC Tip relative Mach No                       1.21343
Design HP Spool Speed     [RPM]               14372.09
HPC Inlet Tip Diameter         in             23.91959
HPC Inlet Hub Diameter         in             18.89648
Calculated HPC Radius Ratio                    0.79000
HP Spool Torque                lb*ft          10722.98
Corr.Flow/Area HPC             lb/(s*ft²)     39.26889
```

Table 11: High Pressure Compressor - Detailed Overview

Again, we set the speed of the HP spool via the tip speed and the corresponding radius. The general characteristics of the HP compressor are given in *Table 11*.



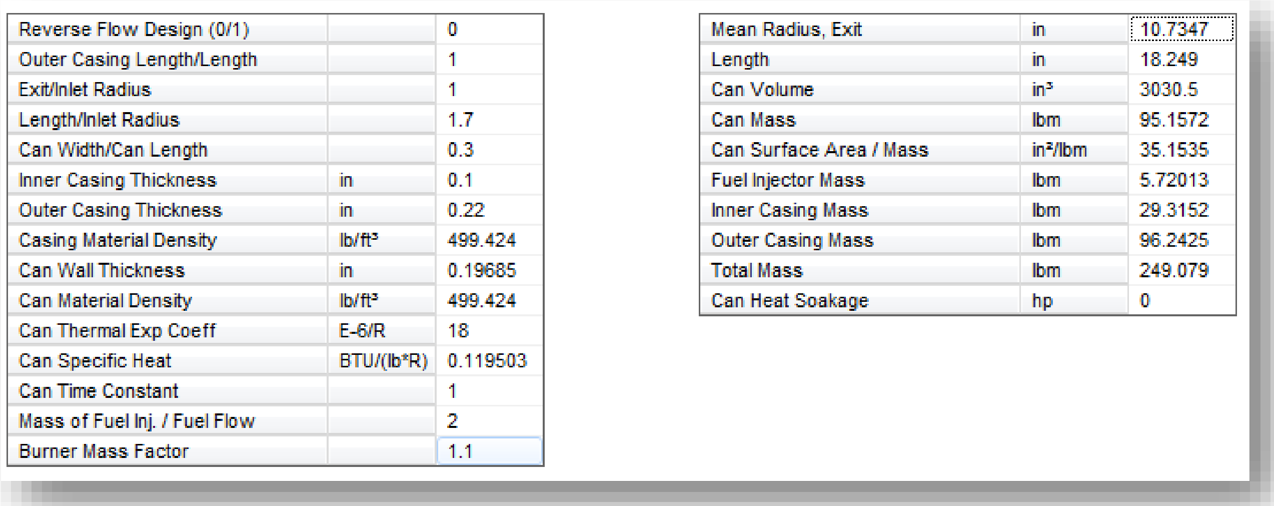
Number of Stages		7
Number of Radial Stages		0
Number of Variable Guide Vanes		1
Inlet Guide Vanes (IGV) 0/1		1
IGV Profile Thickness [%]		5
IGV Material Density	lb/ft³	249.712
Annulus Shape Descriptor 0...1		0.5
Given Radius Rat: Inl/Exit 0/1		0
Inlet Radius Ratio		0.7
Exit Radius Ratio		0
First Stage Aspect Ratio		2
Last Stage Aspect Ratio		2
Blade Gapping: Gap/Chord		0.25
Pitch/Chord Ratio		2
Disk Bore / Inner Inlet Radius		0.2
Diffuser Area Ratio		1.5
Rel Thickness Inner Air Seal		0.04
Compressor Mass Factor		1.1
Outer Casing Thickness	in	0.22
Outer Casing Material Density	lb/ft³	249.712
Casing Thickness	in	0.22
Casing Material Density	lb/ft³	249.712
Rel Work of Radial End Stage		0.3
Rad Diffusor/Rotor Blade Length		0.7
Rotor Inlet Swirl Angle		0
Rotor Blade Backsweep Angle		20
Diffusor Wall Thickness	in	0.0984252
Casing Thermal Exp Coeff	E-6/R	18
Casing Specific Heat	BTU/(lb*R)	0.119503
Casing Time Constant		10
Blade and Vane Time Constant		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 Clear.		5

Length (w/o Diffusor)	in	19.0144
Number of Inlet Guide Vanes		27
Total Number of Blade and Vanes		544
Diffusor Length	in	3.49286
Casing Mass	lbm	44.7366
Outer Casing Mass	lbm	48.5499
Total Vane Mass	lbm	7.15376
Total Blade Mass	lbm	21.5826
Inner Air Seal Mass	lbm	11.9179
Rotating Mass	lbm	170.072
IGV Mass	lbm	0.772678
Exit Diffusor Mass	lbm	15.092
Total Mass	lbm	315.015
Polar Moment of Inertia	lb*in²	10925.7

Table 12: High Pressure Compressor - General Output

3.7 Combustor

A fairly conventional annular combustor is used and details are given in *Table 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.

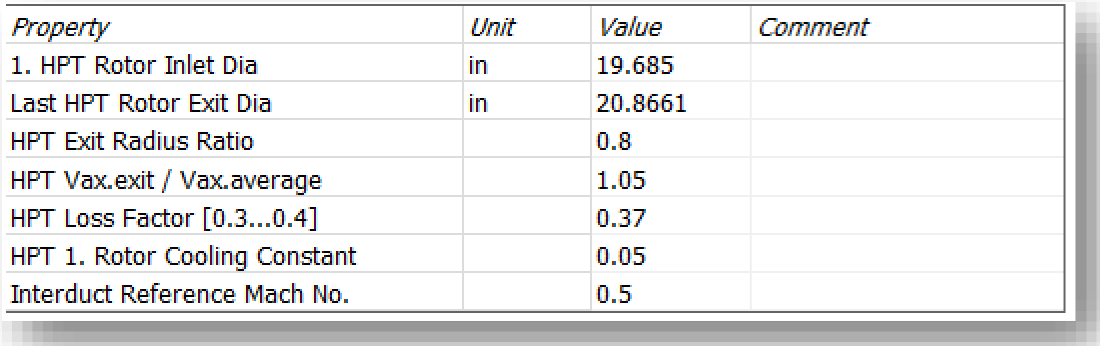


Reverse Flow Design (0/1)		0
Outer Casing Length/Length		1
Exit/Inlet Radius		1
Length/Inlet Radius		1.7
Can Width/Can Length		0.3
Inner Casing Thickness	in	0.1
Outer Casing Thickness	in	0.22
Casing Material Density	lb/ft^3	499.424
Can Wall Thickness	in	0.19685
Can Material Density	lb/ft^3	499.424
Can Thermal Exp Coeff	E-6/R	18
Can Specific Heat	BTU/(lb*R)	0.119503
Can Time Constant		1
Mass of Fuel Inj. / Fuel Flow		2
Burner Mass Factor		1.1

Mean Radius, Exit	in	10.7347
Length	in	18.249
Can Volume	in^3	3030.5
Can Mass	lbm	95.1572
Can Surface Area / Mass	in^2/lbm	35.1535
Fuel Injector Mass	lbm	5.72013
Inner Casing Mass	lbm	29.3152
Outer Casing Mass	lbm	96.2425
Total Mass	lbm	249.079
Can Heat Soakage	hp	0

Table 13: Combustor

3.8 High-Pressure Turbine

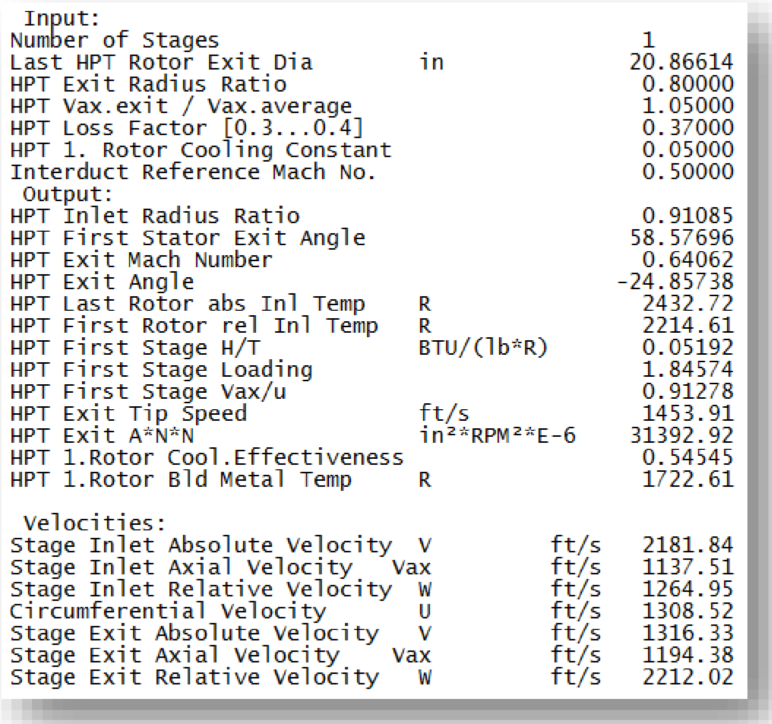


Property	*Unit*	*Value*	*Comment*
1. HPT Rotor Inlet Dia	in	19.685	
Last HPT Rotor Exit Dia	in	20.8661	
HPT Exit Radius Ratio		0.8	
HPT Vax.exit / Vax.average		1.05	
HPT Loss Factor [0.3...0.4]		0.37	
HPT 1. Rotor Cooling Constant		0.05	
Interduct Reference Mach No.		0.5	

Table 14: High Pressure Turbine – Basis for Efficiency Estimate

As stated in *Section 3.1*, the efficiency of the high pressure turbine was estimated by *GasTurb12* on the basis of the data shown in *Table 13*, which is made available once that efficiency option is selected.

The useful summary of the HP turbine presented in *Table 14* appears as a result of that selection.

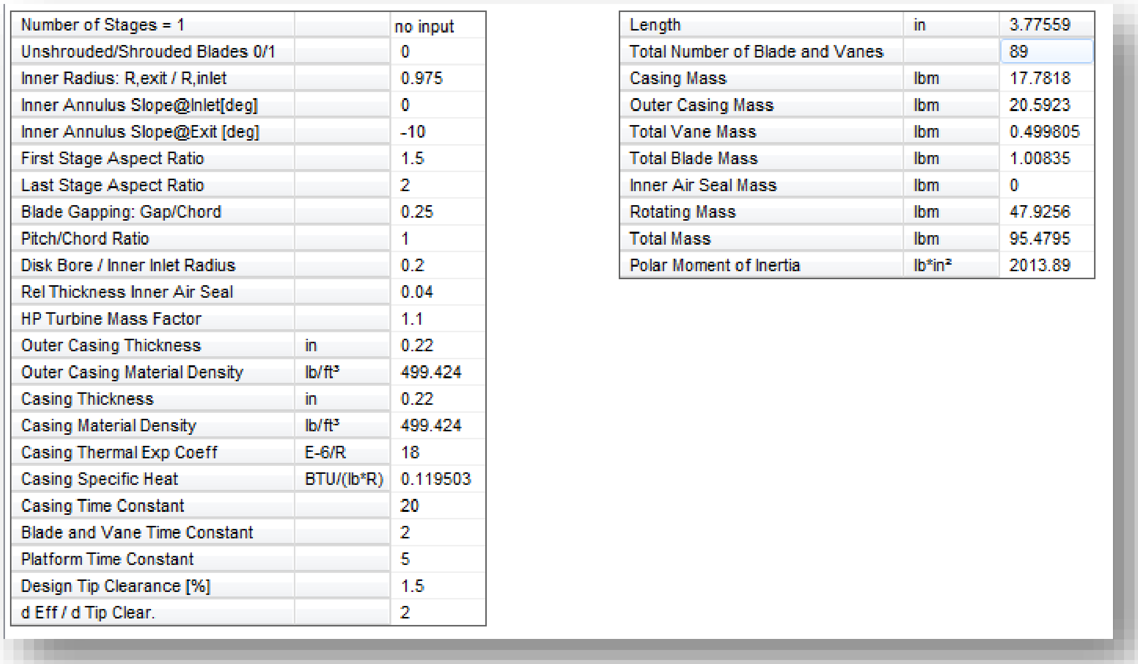


```
 Input:
Number of Stages                                   1
Last HPT Rotor Exit Dia        in           20.86614
HPT Exit Radius Ratio                        0.80000
HPT Vax.exit / Vax.average                   1.05000
HPT Loss Factor [0.3...0.4]                  0.37000
HPT 1. Rotor Cooling Constant                0.05000
Interduct Reference Mach No.                 0.50000
 Output:
HPT Inlet Radius Ratio                       0.91085
HPT First Stator Exit Angle                 58.57696
HPT Exit Mach Number                         0.64062
HPT Exit Angle                             -24.85738
HPT Last Rotor abs Inl Temp    R             2432.72
HPT First Rotor rel Inl Temp   R             2214.61
HPT First Stage H/T            BTU/(lb*R)    0.05192
HPT First Stage Loading                      1.84574
HPT First Stage Vax/u                        0.91278
HPT Exit Tip Speed             ft/s          1453.91
HPT Exit A*N*N                 in²*RPM²*E-6 31392.92
HPT 1.Rotor Cool.Effectiveness               0.54545
HPT 1.Rotor Bld Metal Temp     R             1722.61

 Velocities:
Stage Inlet Absolute Velocity  V        ft/s  2181.84
Stage Inlet Axial Velocity   Vax        ft/s  1137.51
Stage Inlet Relative Velocity  W        ft/s  1264.95
Circumferential Velocity       U        ft/s  1308.52
Stage Exit Absolute Velocity   V        ft/s  1316.33
Stage Exit Axial Velocity    Vax        ft/s  1194.38
Stage Exit Relative Velocity   W        ft/s  2212.02
```

Table 15: HPT Summary

A general summary of the HP turbine is given in *Table 15*, followed by the velocity diagrams and *Smith Chart* in *Figure 5*.



Number of Stages = 1		no input
Unshrouded/Shrouded Blades 0/1		0
Inner Radius: R,exit / R,inlet		0.975
Inner Annulus Slope@Inlet[deg]		0
Inner Annulus Slope@Exit [deg]		-10
First Stage Aspect Ratio		1.5
Last Stage Aspect Ratio		2
Blade Gapping: Gap/Chord		0.25
Pitch/Chord Ratio		1
Disk Bore / Inner Inlet Radius		0.2
Rel Thickness Inner Air Seal		0.04
HP Turbine Mass Factor		1.1
Outer Casing Thickness	in	0.22
Outer Casing Material Density	lb/ft^3	499.424
Casing Thickness	in	0.22
Casing Material Density	lb/ft^3	499.424
Casing Thermal Exp Coeff	E-6/R	18
Casing Specific Heat	BTU/(lb*R)	0.119503
Casing Time Constant		20
Blade and Vane Time Constant		2
Platform Time Constant		5
Design Tip Clearance [%]		1.5
d Eff / d Tip Clear.		2

Length	in	3.77559
Total Number of Blade and Vanes		89
Casing Mass	lbm	17.7818
Outer Casing Mass	lbm	20.5923
Total Vane Mass	lbm	0.499805
Total Blade Mass	lbm	1.00835
Inner Air Seal Mass	lbm	0
Rotating Mass	lbm	47.9256
Total Mass	lbm	95.4795
Polar Moment of Inertia	lb*in^2	2013.89

Table 16: High Pressure Turbine – General Output

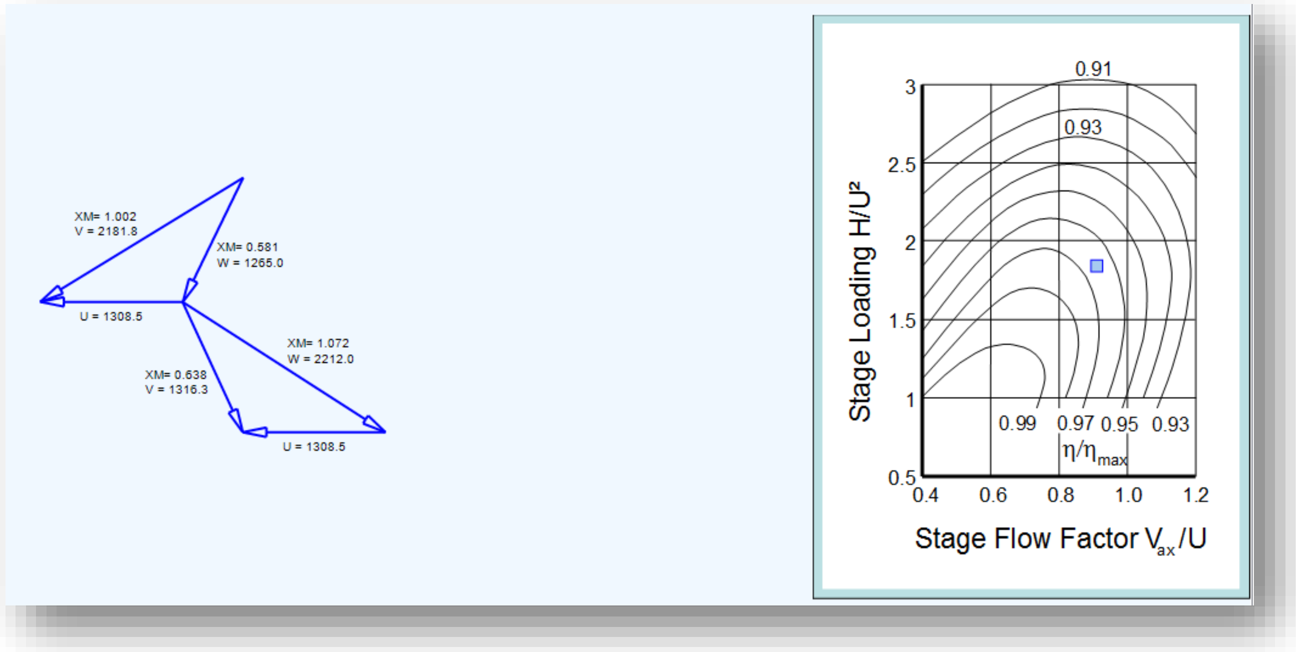
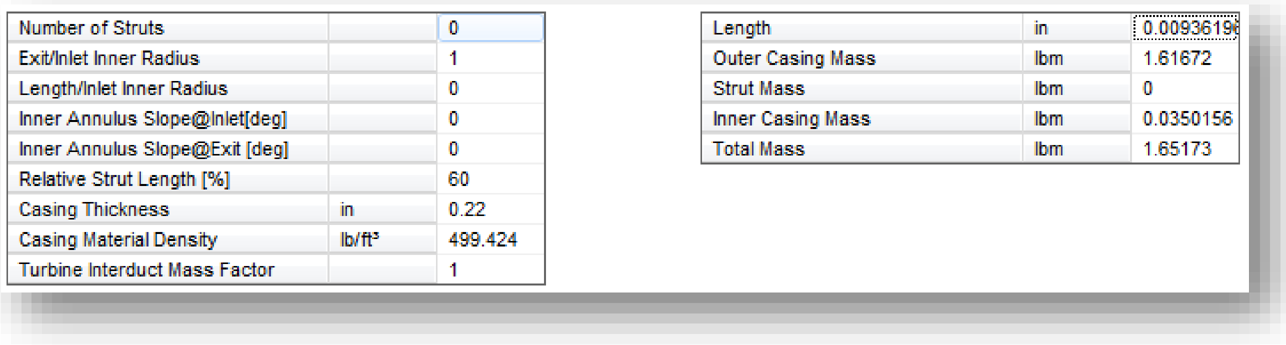


Figure 5: High Pressure Turbine Velocity Diagrams & Smith Chart

3.9 Inter-Turbine Duct

Table 16 contains details of the inter-turbine duct. Its relatively short length allows the two turbines to be close-coupled and the exit-to-inlet radius ratio of 1.0 emphasizes this.



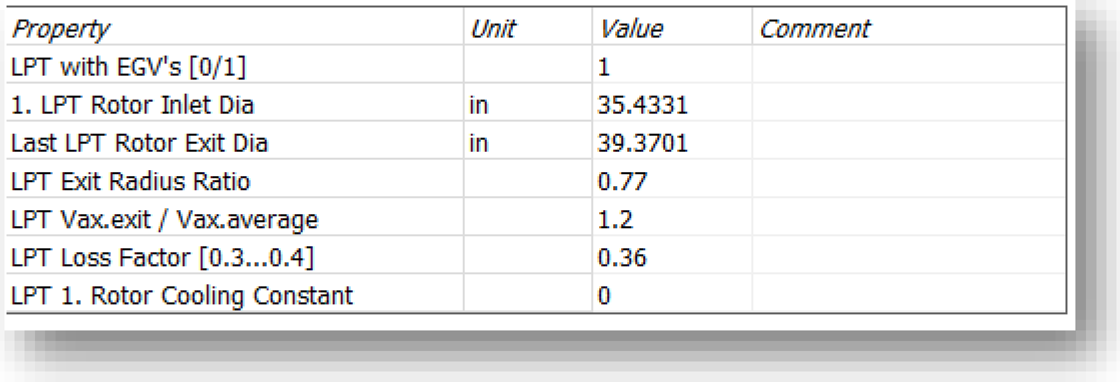
Number of Struts		0
Exit/Inlet Inner Radius		1
Length/Inlet Inner Radius		0
Inner Annulus Slope@Inlet[deg]		0
Inner Annulus Slope@Exit [deg]		0
Relative Strut Length [%]		60
Casing Thickness	in	0.22
Casing Material Density	lb/ft³	499.424
Turbine Interduct Mass Factor		1

Length	in	0.0093619
Outer Casing Mass	lbm	1.61672
Strut Mass	lbm	0
Inner Casing Mass	lbm	0.0350156
Total Mass	lbm	1.65173

Table 17: Inter-Turbine Duct

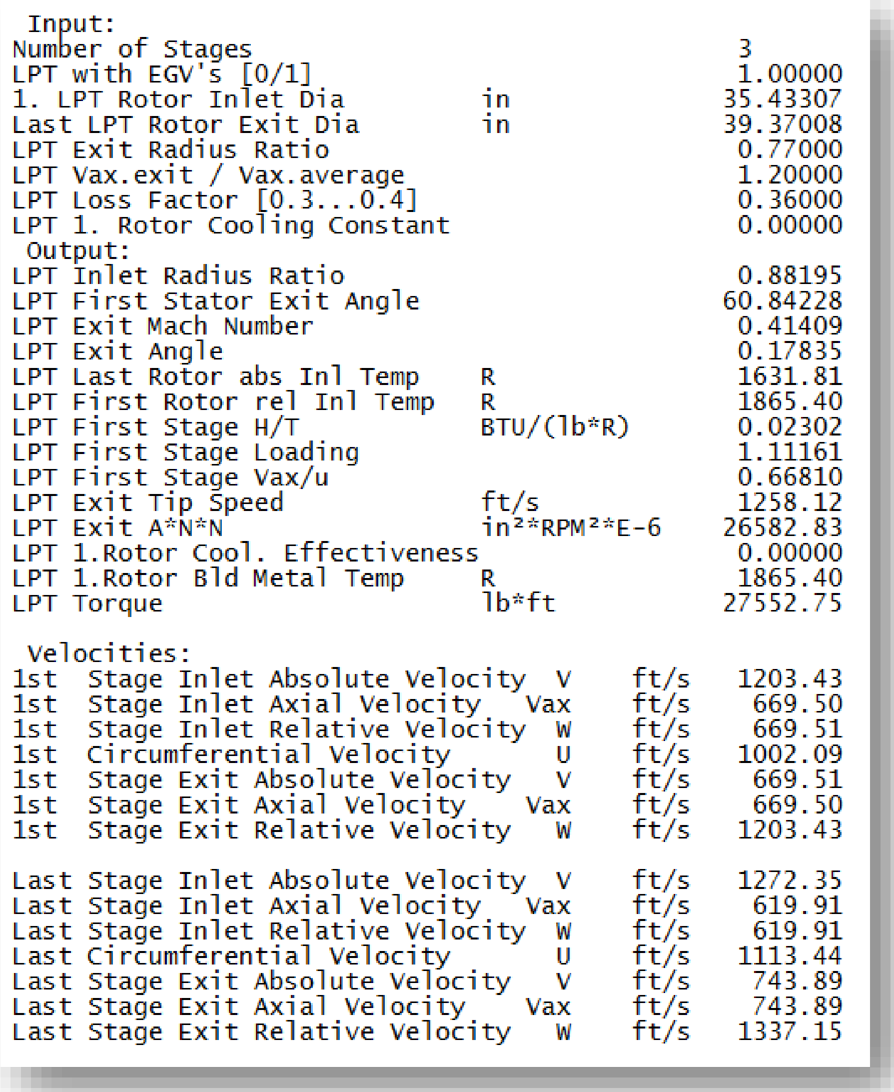
3.10 Low-Pressure Turbine

Characteristics of the low pressure turbine are presented in *Tables 17* - 19 and *Figure 6*. *Figure 6* shows velocity diagrams for the first and last stages only. The flared nature of the LP turbine flowpath ensures that meanline radii are maximized, stage loading coefficients are minimized and stage efficiencies are optimized. This may be observed in *Figure 6*, where the common design point for all three stages is nicely centered on the *Smith Chart* due mainly to the high mean blade speed. It should be noted that the efficiency contours in *Figure 6* (and *Figure 5*) are expressed as fractions of the maximum value on the chart! The true value of the average stage efficiency is 92.68%, which corresponds to the value in the engine performance summary in *Table 4*.



Property	*Unit*	*Value*	*Comment*
LPT with EGV's [0/1]		1	
1. LPT Rotor Inlet Dia	in	35.4331	
Last LPT Rotor Exit Dia	in	39.3701	
LPT Exit Radius Ratio		0.77	
LPT Vax.exit / Vax.average		1.2	
LPT Loss Factor [0.3...0.4]		0.36	
LPT 1. Rotor Cooling Constant		0	

Table 18: Basis for LP Turbine Calculated Efficiency

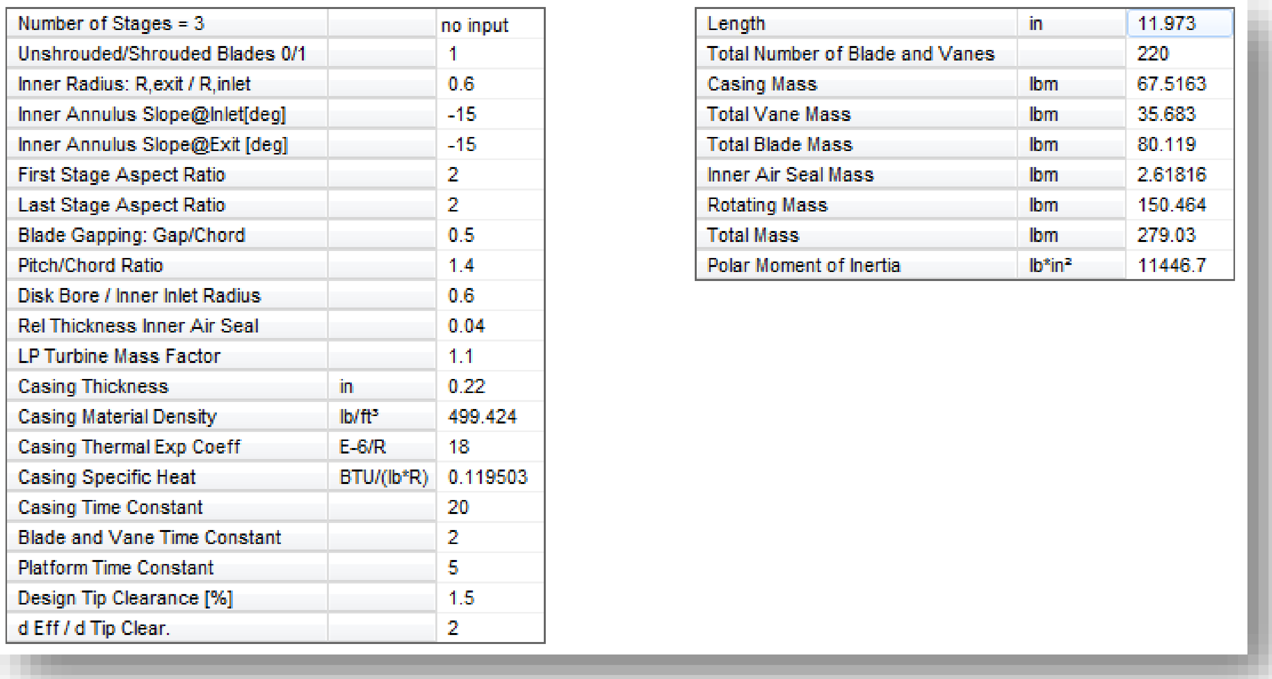


```
 Input:
Number of Stages                                3
LPT with EGV's [0/1]                            1.00000
1. LPT Rotor Inlet Dia          in             35.43307
Last LPT Rotor Exit Dia         in             39.37008
LPT Exit Radius Ratio                           0.77000
LPT Vax.exit / Vax.average                      1.20000
LPT Loss Factor [0.3...0.4]                     0.36000
LPT 1. Rotor Cooling Constant                   0.00000
 Output:
LPT Inlet Radius Ratio                          0.88195
LPT First Stator Exit Angle                    60.84228
LPT Exit Mach Number                            0.41409
LPT Exit Angle                                  0.17835
LPT Last Rotor abs Inl Temp     R               1631.81
LPT First Rotor rel Inl Temp    R               1865.40
LPT First Stage H/T             BTU/(lb*R)      0.02302
LPT First Stage Loading                         1.11161
LPT First Stage Vax/u                           0.66810
LPT Exit Tip Speed              ft/s            1258.12
LPT Exit A*N*N                  in²*RPM²*E-6   26582.83
LPT 1.Rotor Cool. Effectiveness                 0.00000
LPT 1.Rotor Bld Metal Temp      R               1865.40
LPT Torque                      lb*ft          27552.75

 Velocities:
1st  Stage Inlet Absolute Velocity  V    ft/s   1203.43
1st  Stage Inlet Axial Velocity   Vax    ft/s    669.50
1st  Stage Inlet Relative Velocity  W    ft/s    669.51
1st  Circumferential Velocity       U    ft/s   1002.09
1st  Stage Exit Absolute Velocity   V    ft/s    669.51
1st  Stage Exit Axial Velocity    Vax    ft/s    669.50
1st  Stage Exit Relative Velocity   W    ft/s   1203.43

Last Stage Inlet Absolute Velocity  V    ft/s   1272.35
Last Stage Inlet Axial Velocity   Vax    ft/s    619.91
Last Stage Inlet Relative Velocity  W    ft/s    619.91
Last Circumferential Velocity       U    ft/s   1113.44
Last Stage Exit Absolute Velocity   V    ft/s    743.89
Last Stage Exit Axial Velocity    Vax    ft/s    743.89
Last Stage Exit Relative Velocity   W    ft/s   1337.15
```

Table 19: LPT Summary



Number of Stages = 3		no input
Unshrouded/Shrouded Blades 0/1		1
Inner Radius: R,exit / R,inlet		0.6
Inner Annulus Slope@Inlet[deg]		-15
Inner Annulus Slope@Exit [deg]		-15
First Stage Aspect Ratio		2
Last Stage Aspect Ratio		2
Blade Gapping: Gap/Chord		0.5
Pitch/Chord Ratio		1.4
Disk Bore / Inner Inlet Radius		0.6
Rel Thickness Inner Air Seal		0.04
LP Turbine Mass Factor		1.1
Casing Thickness	in	0.22
Casing Material Density	lb/ft³	499.424
Casing Thermal Exp Coeff	E-6/R	18
Casing Specific Heat	BTU/(lb*R)	0.119503
Casing Time Constant		20
Blade and Vane Time Constant		2
Platform Time Constant		5
Design Tip Clearance [%]		1.5
d Eff / d Tip Clear.		2

Length	in	11.973
Total Number of Blade and Vanes		220
Casing Mass	lbm	67.5163
Total Vane Mass	lbm	35.683
Total Blade Mass	lbm	80.119
Inner Air Seal Mass	lbm	2.61816
Rotating Mass	lbm	150.464
Total Mass	lbm	279.03
Polar Moment of Inertia	lb*in²	11446.7

Table 20: Low Pressure Turbine: General Output

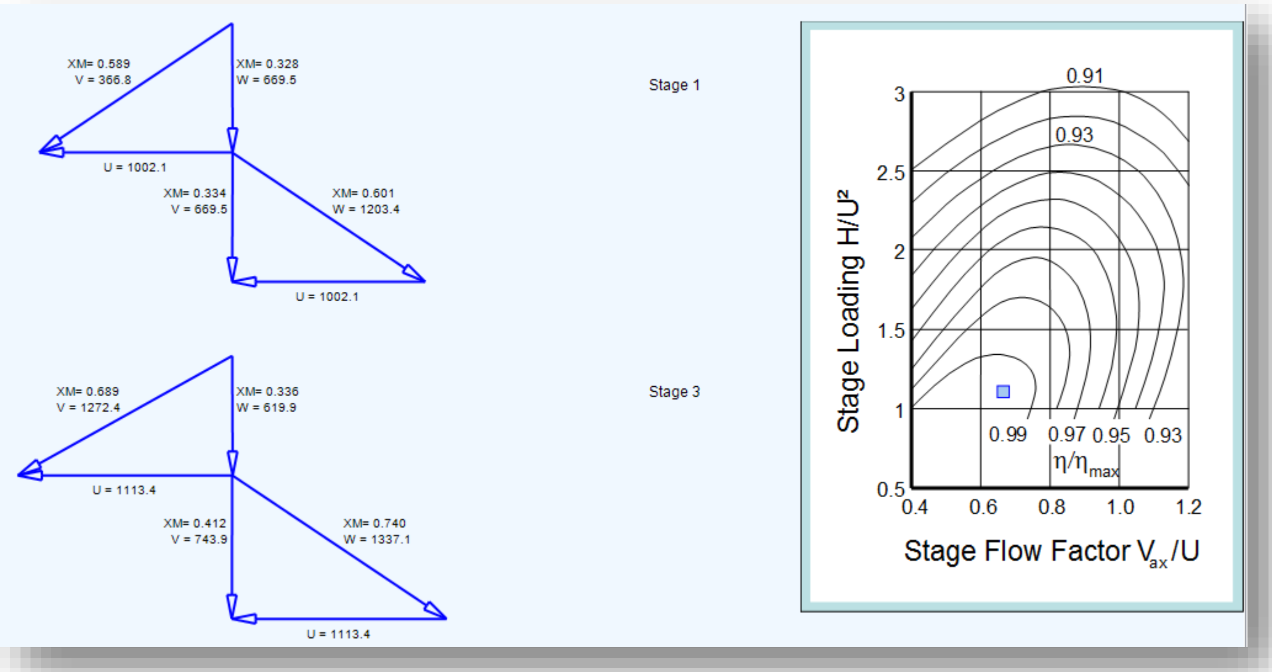
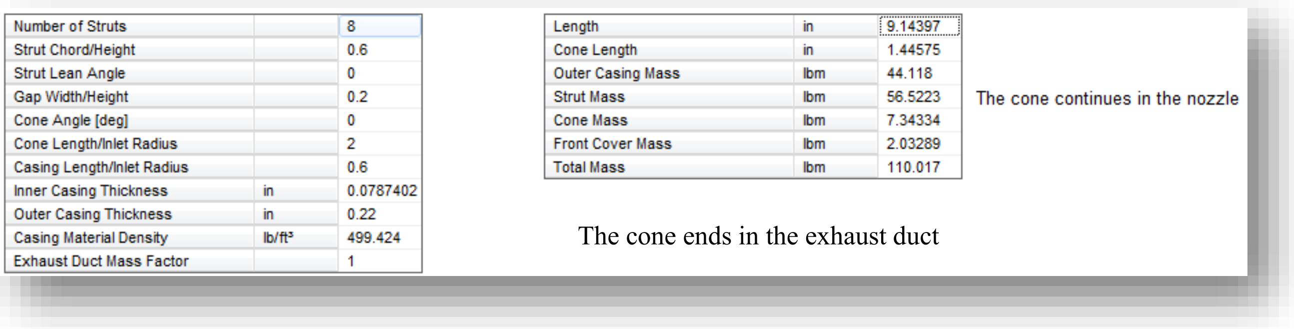


Figure 6: Low Pressure Turbine Velocity Diagrams & Smith Chart

3.11 Core Exhaust & Core 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. In *Figure 4* on page 16, the core exhaust extends to about 125 inches. It is important to note that in *GasTurb12* the core exhaust does not include the convergent portion or core nozzle. *Table 20* below contains the input and output details of the core exhaust.



Number of Struts		8
Strut Chord/Height		0.6
Strut Lean Angle		0
Gap Width/Height		0.2
Cone Angle [deg]		0
Cone Length/Inlet Radius		2
Casing Length/Inlet Radius		0.6
Inner Casing Thickness	in	0.0787402
Outer Casing Thickness	in	0.22
Casing Material Density	lb/ft^3	499.424
Exhaust Duct Mass Factor		1

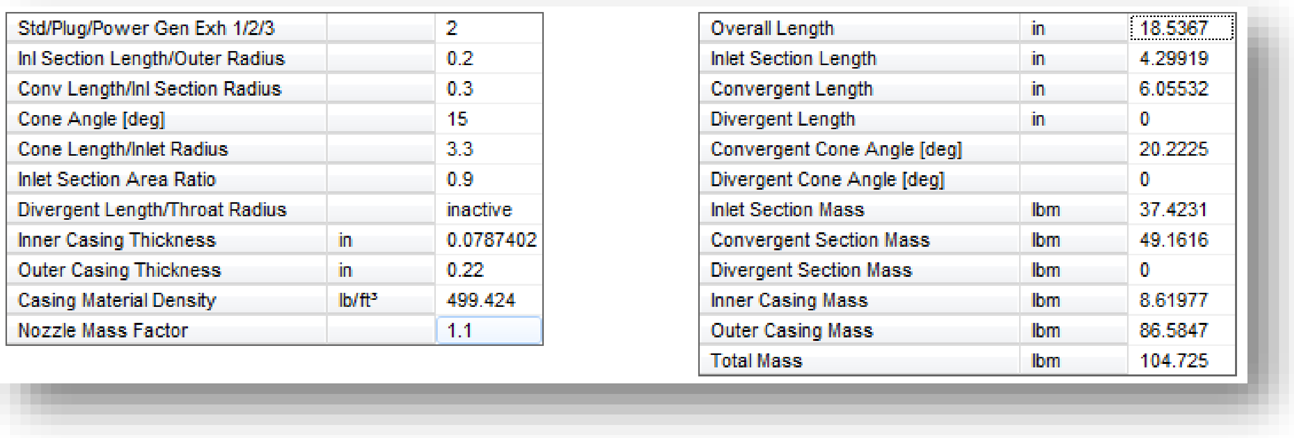
Length	in	9.14397
Cone Length	in	1.44575
Outer Casing Mass	lbm	44.118
Strut Mass	lbm	56.5223
Cone Mass	lbm	7.34334
Front Cover Mass	lbm	2.03289
Total Mass	lbm	110.017

The cone continues in the nozzle

The cone ends in the exhaust duct

Table 21: Core Exhaust

The core nozzle is the part of the engine that converges to its exit area at 135 inches in *Figure 4*. The casing material density in the core nozzle is the same as that for the core exhaust, although a lighter material most likely could have been used owing to the prevailing value of mixed temperature. Core nozzle data is found in *Table 21*.



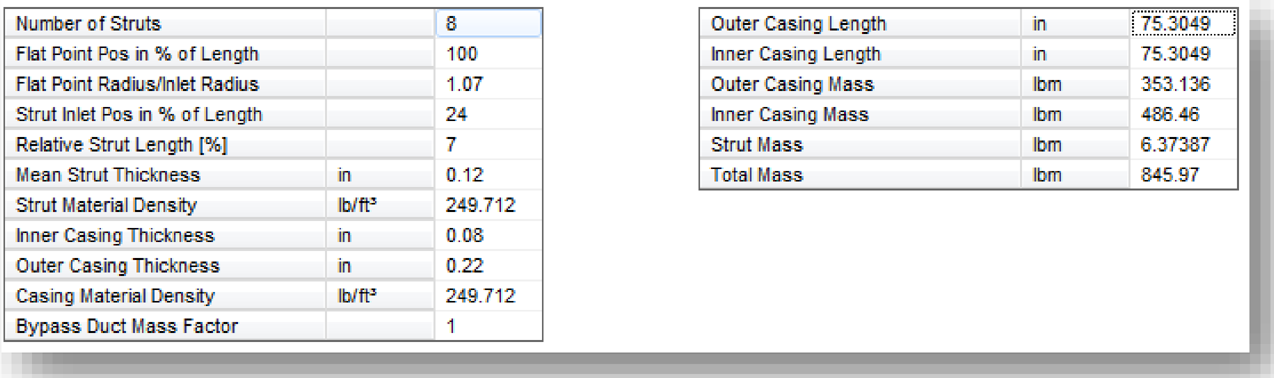
Std/Plug/Power Gen Exh 1/2/3		2
Inl Section Length/Outer Radius		0.2
Conv Length/Inl Section Radius		0.3
Cone Angle [deg]		15
Cone Length/Inlet Radius		3.3
Inlet Section Area Ratio		0.9
Divergent Length/Throat Radius		inactive
Inner Casing Thickness	in	0.0787402
Outer Casing Thickness	in	0.22
Casing Material Density	lb/ft^3	499.424
Nozzle Mass Factor		1.1

Overall Length	in	18.5367
Inlet Section Length	in	4.29919
Convergent Length	in	6.05532
Divergent Length	in	0
Convergent Cone Angle [deg]		20.2225
Divergent Cone Angle [deg]		0
Inlet Section Mass	lbm	37.4231
Convergent Section Mass	lbm	49.1616
Divergent Section Mass	lbm	0
Inner Casing Mass	lbm	8.61977
Outer Casing Mass	lbm	86.5847
Total Mass	lbm	104.725

Table 22: Core Nozzle

3.12 Bypass Duct & Mixer

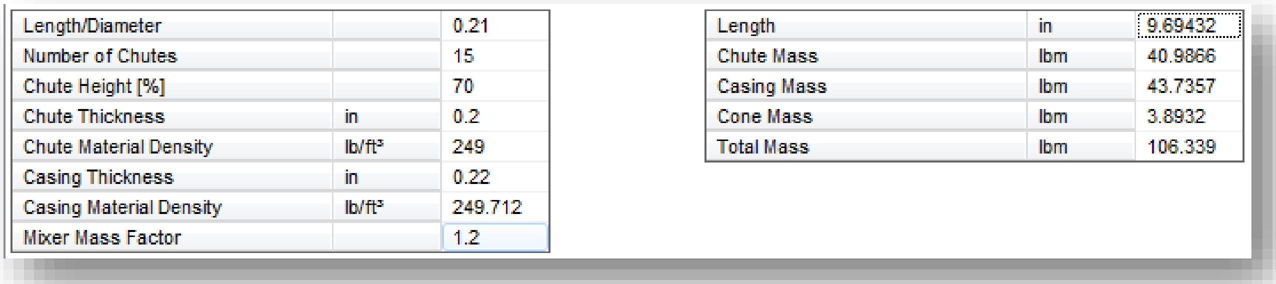
Tables 22 and *23* define the input and output parameters for the bypass duct and the mixer. Recall that the mixer input parameters appeared with the basic input in *Table 3*.



Number of Struts		8
Flat Point Pos in % of Length		100
Flat Point Radius/Inlet Radius		1.07
Strut Inlet Pos in % of Length		24
Relative Strut Length [%]		7
Mean Strut Thickness	in	0.12
Strut Material Density	lb/ft^3	249.712
Inner Casing Thickness	in	0.08
Outer Casing Thickness	in	0.22
Casing Material Density	lb/ft^3	249.712
Bypass Duct Mass Factor		1

Outer Casing Length	in	75.3049
Inner Casing Length	in	75.3049
Outer Casing Mass	lbm	353.136
Inner Casing Mass	lbm	486.46
Strut Mass	lbm	6.37387
Total Mass	lbm	845.97

Table 23: Bypass Duct



Length/Diameter		0.21
Number of Chutes		15
Chute Height [%]		70
Chute Thickness	in	0.2
Chute Material Density	lb/ft^3	249
Casing Thickness	in	0.22
Casing Material Density	lb/ft^3	249.712
Mixer Mass Factor		1.2

Length	in	9.69432
Chute Mass	lbm	40.9866
Casing Mass	lbm	43.7357
Cone Mass	lbm	3.8932
Total Mass	lbm	106.339

Table 24: Mixer

4. 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 assumed to be the same as those of the baseline engine, and be input directly or the “calculate efficiency” mode of *GasTurb12* may be invoked.

- This is not an aircraft design competition, so credit will not be given for derivation of aircraft flight characteristics. Thrust requirements for the mission are given in *Figure 2*.

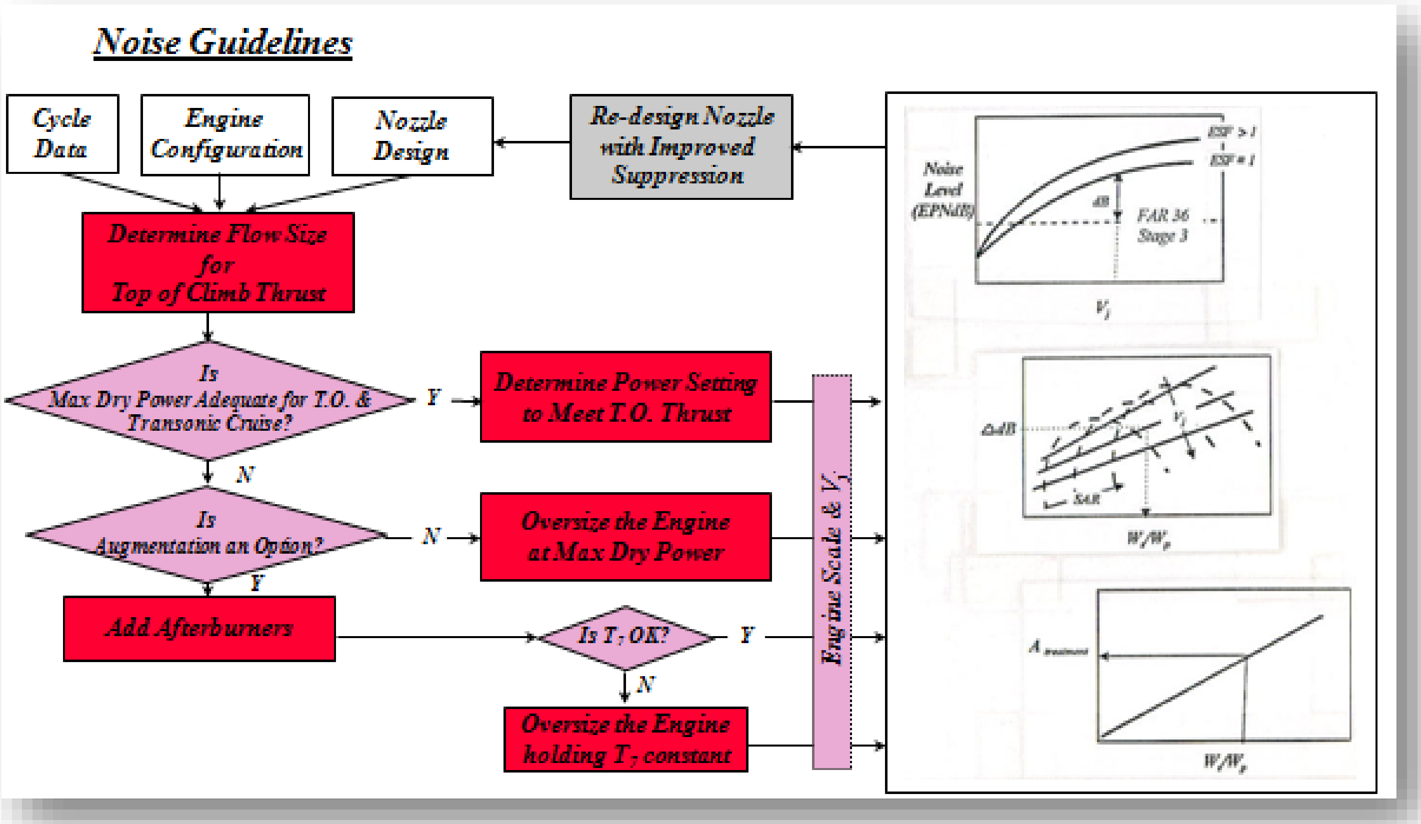


Figure 7: Nozzle Design for Multi Mission Applications

- Examples of nozzle design processes are given in *Figures 7* and *8*. These are not intended to define a specific sequence of activities for this competition but merely to provide some ideas for issues that could be considered. *Figure 7* addresses the use of a mixer-ejector nozzle and its right hand side contains curves of (1) noise level as a function of jet velocity, (2) noise attenuation (ΔdB) as a function of secondary-to-primary flow, suppressor area ratio & jet velocity, and (3) required area of noise absorbent needed to offset the primary/secondary flow mixing noise. *Figure 8* outlines a general approach.

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

5. 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:

- Completion of the compliance matrices and required trade studies listed on **Error! Reference source not found.**, **Error! Reference source not found.**, and **Error! Reference source not found.**, 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.
    - CFD (Computational Fluid Dynamics) & FEA (Finite Element Analysis) will be excluded from judging and is encouraged not to be used.
    - If a CAD model is shown it must be consistent with Analysis provided.

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.

Proposals page limits will not include the administrative/ contents or the "signature" pages.

6. References

1. "*GE Tests CMCs for Future Engine*", Aviation Week & Space Technology, July 30, 2012.
2. "*Aerospace Source Book",* Aviation Week & Space Technology, January 15, 2007.
3. *"GasTurb 12: A Design & Off-Design Performance Program for Gas Turbines",* http://www.gasturb.de, Joachim Kurzke, 2012.
4. "*A Simple Correlation of Turbine Efficiency*", S. F. Smith, Journal of the Royal Aeronautical Society, Volume 69, 1965.
5. *"Aeronautical Vest Pocket Handbook"*, *Pratt & Whitney Aircraft,* Circa 1980.
6. Roux, Elodie, "*Turbofan and Turbojet Engines: Database Handbook*", 2007, ISBN: 978-2-9529380-1-3
7. *TPE331-10 Turboprop Engine*, Honeywell International Inc., 2006, aerocontent.honeywell.com/aero/common/documents/myaerospacecatalog-documents/BA_brochures-documents/TPE331.10.pdf.
8. "*Overview of NASA Electrified Aircraft Propulsion Research for Large Subsonic Transports*", https://ntrs.nasa.gov/archive/nasa/casi.ntrs.nasa.gov/20170012222.pdf Jansen, Ralph H., Bowman, Dr. Cheryl, Jankovsky, Amy, Dyson, Dr. Rodger, Felder, James L., NASA Glenn Research Center, Cleveland, OH

7. 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.

8. 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)
- NPSS® Learning Edition
  - www.npssconsortium.org

  Numerical Propulsion System Simulation (NPSS®) is an object oriented, multi-physics, engineering design and simulation environment used by many of the major aerospace companies. Primary application areas for NPSS include aerospace systems (i.e. engine performance models for aircraft propulsion), thermodynamic system analysis such as Rankine and Brayton cycles, various rocket propulsion cycles, and industry standardization for model sharing and integration. However, since it is fundamentally a flow-network solver, it has also been applied to a variety of other fluid/thermal subjects such as multiphase heat transfer systems, refrigeration cycles, variations of common power cycles (i.e. Brayton), and overall vehicle emission analyses. NPSS is available for free to academia throughout the world in support of the AIAA engine design competition, and comes with an example model ready for use in the contest.

- 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. By utilizing the educational version of the software (AxSTREAM EDU™), students will have the opportunity to work with real-world design tools for practical experience in topics including, but not limited to, propulsion, energy, and power generation. AxSTREAM EDU™ allows students to work through the entire design process including, but not limited to:
    - Preliminary design
    - Meanline (1D) and axisymmetric (2D) analysis
    - Profiling and 3D blade design

  The software can be utilized for axial, radial, mixed-flow, and diagonal configurations for turbines, compressors and fans. In addition, students also have the option of utilizing AxCYCLE™ as an add-on to AxSTREAM EDU™ for thermodynamic 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** (LOI) 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.
    - From there, students **must** contact the AIAA Student competition Chair, listed with the abstract, who will then contact 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 as well as use of AxSTREAM EDU™.