Joint AIAA Foundation/ASME.IGTI Student Design Competition 2012/13
Undergraduate Team – Engine

An Improved Engine for a High Altitude Long Endurance Unmanned Air Vehicle

- Request for Proposal -

August 30, 2012

Abstract

The *Northrop Grumman RQ4 Global Hawk* is an unmanned aerial vehicle used by the United States Air Force and Navy as a high altitude platform for surveillance and security. The capabilities of such an aircraft allow more precise targeting of weapons and better protection of forces than conventional air vehicles. In typical operations the Global Hawk has a cruise speed of 357 mph (310 kn; 575 km/h), a range of 8,700 mi (7,560 nmi; 14,001 km), a service ceiling of 60,000 feet (18,288 m), and may fly for up to 28 hours. It is powered by a single *Rolls-Royce AE3007H* turbofan engine with a nominal net thrust of 7,050 lbf (31.4 kN) at sea level take-off. The challenges of successful operation at extremely high altitudes are quite substantial for any gas turbine engine, however, and improved performance is sought continually. This Request For Proposal is seeking a new design as a potential replacement for the current engine. Candidate engines must be lighter & smaller in order that the payload and/or operating altitude can be increased and have an improved fuel burn so that range can be extended.

A generic model of the current power plant, the *Rolls-Royce AE3007H* turbofan, is supplied. Responders should generate a typical, multi-element, mission that specifically addresses the general improvements listed above and covers design point and off-design engine operations. The performance and total fuel consumption of the current and candidate engines should be estimated over the mission and compared in the proposal. Special attention should be paid to dimensions & integration with the aircraft, engine mass, technical feasibility and operating costs.

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1. **Introduction**

The Northrop Grumman RQ-4 Global Hawk is an unmanned aerial vehicle (UAV) and has been used for many years by the United States Air Force, Navy and by the German Luftwaffe as a surveillance aircraft. The Global Hawk is a modern version of the Lockheed U-2 of the 1950s in terms of its role and operation and is used by war-theater commanders to provide a broad overview and systematic target surveillance. For this purpose, the Global Hawk is able to provide high resolution Synthetic Aperture Radar (SAR) that can penetrate cloud-cover and sandstorms and Electro-Optical/Infrared (EO/IR) imagery at long range with long loiter times over target areas. Its in-flight endurance can exceed 30 hours. This enables it to survey as much as 40,000 square miles (103,600 square kilometers) of terrain a day.

On 21 March 2001, aircraft number 982003, the third Advanced Concept Technology Demonstration aircraft, set an official world endurance record for UAVs, at 30 hours, 24 minutes and 1 second, flying from Edwards Air Force Base. During the same flight, it set an absolute altitude record of 19,928 meters (65,381 feet) in its FAI class category. On 24 April 2001 a Global Hawk flew non-stop from Edwards in the US to the Royal Australian Air Force Base, Edinburgh in Australia, making history by being the first pilotless aircraft to cross the Pacific Ocean. The flight took 22 hours, and set a world record for absolute distance flown by a UAV of 13,219.86 kilometers (8,214.44 mi) (*Reference 1*).

The Global Hawk costs about US$35 million to procure each aircraft. The unit cost rises to US$218 million when development costs are included.

RQ-4A

General characteristics

- **Crew:** 0
- **Length:** 44 ft 5 in (13.54 m)
- **Wingspan:** 116 ft 2 in (35.41 m)
- **Height:** 15 ft 2 in (4.62 m)
- **Empty weight:** 8,490 lb (3,851 kg)
- **Gross weight:** 22,900 lb (10,387 kg)
- **Powerplant:** 1 × Allison Rolls-Royce AE3007H turbofan engine, 7,050 lbf (31.4 kN) thrust

Performance

- **Maximum speed:** 497.1 mph (800 km/h; 432 kn)
- **Cruise speed:** 404 mph (351 kn; 650 km/h)
- **Range:** 15,525 mi (13,491 nmi; 24,985 km)
- **Endurance:** 36 hours
- **Service ceiling:** 65,000 ft (19,812 m)

Table 1: Some General Characteristics of the RQ-4A Global Hawk Aircraft

Table 1 contains some general characteristics of the airplane, also obtained from *Reference 1*.

The *Global Hawk* is powered by a single *Rolls-Royce AE3007H* turbofan - designated as the *F137* in military service - with a nominal net thrust of 7,050 lbf (31.4 kN) at sea level take-off. *Figure 1* shows the installation of the engine in the aircraft.

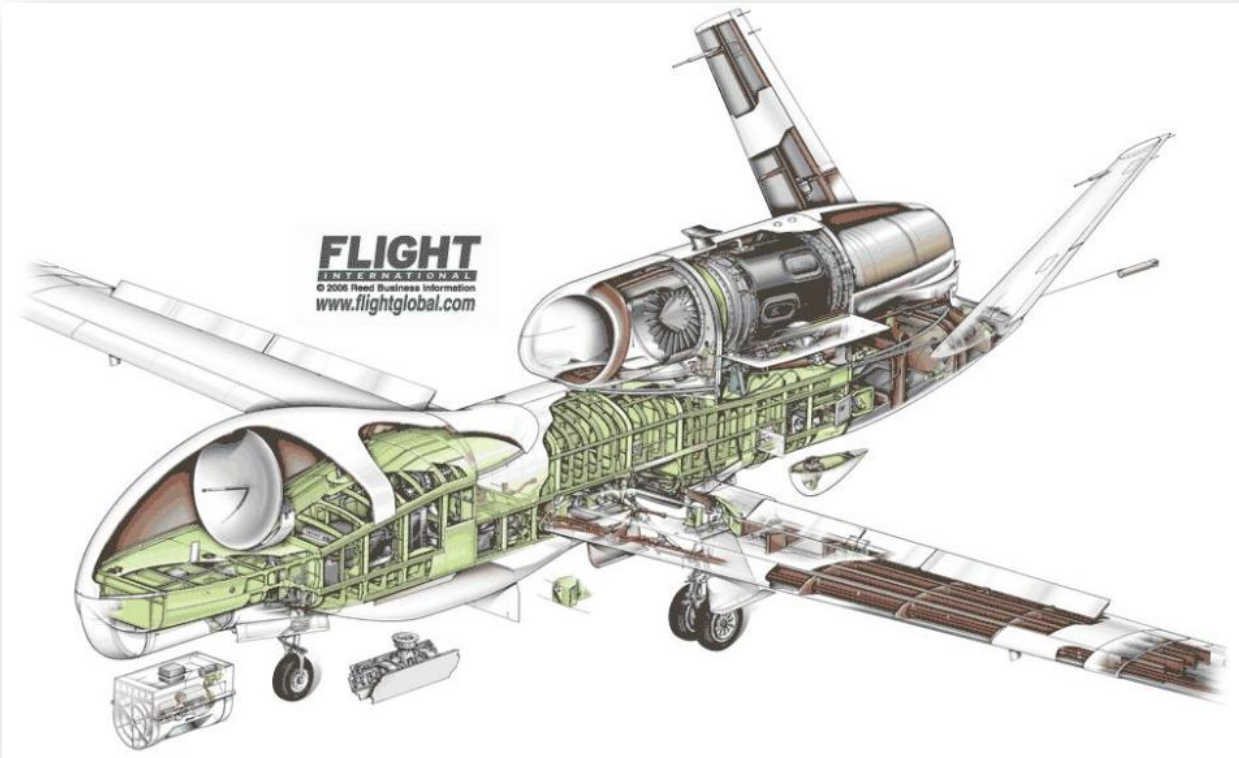

Figure 1: Cutaway Illustration of the Global Hawk & Power Plant

2. Design Objectives & Requirements

- A new engine design is required for the *Northrop Grumman RQ4 Global Hawk* unmanned air vehicle.

- While the current flight envelope ranges from take-off at static sea-level conditions to subsonic cruise at 65,000 feet/Mach 0.6, the new engine should be able to operate at altitude of 70,000 feet at the same Mach number. It is hoped that the endurance might also be extended by reducing the fuel consumption.

- In order to allow more surveillance equipment to be carried and provision for in-flight refueling to be included, reduced engine mass is also an objective.

- A different engine architecture is permitted, but accommodation within the existing envelope is preferred (See *Figure 1*).

- Assume entry into service for the new engine to be 2025. Based on this, the development of new materials and an increase in design limits such as for the turbine entry temperature may be assumed. The development and potential application of carbon matrix composites is of particular interest (*Reference 2*). 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.

- Design proposals must include engine mass, engine dimensions, net thrust values, specific fuel consumption, thermal and propulsive efficiencies. 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.

- Define a mission that exemplifies that of the *Global Hawk*. "Fly" the mission with both the baseline engine and your new candidate engine and estimate the overall reduction in fuel burn.

3. Baseline Engine Model

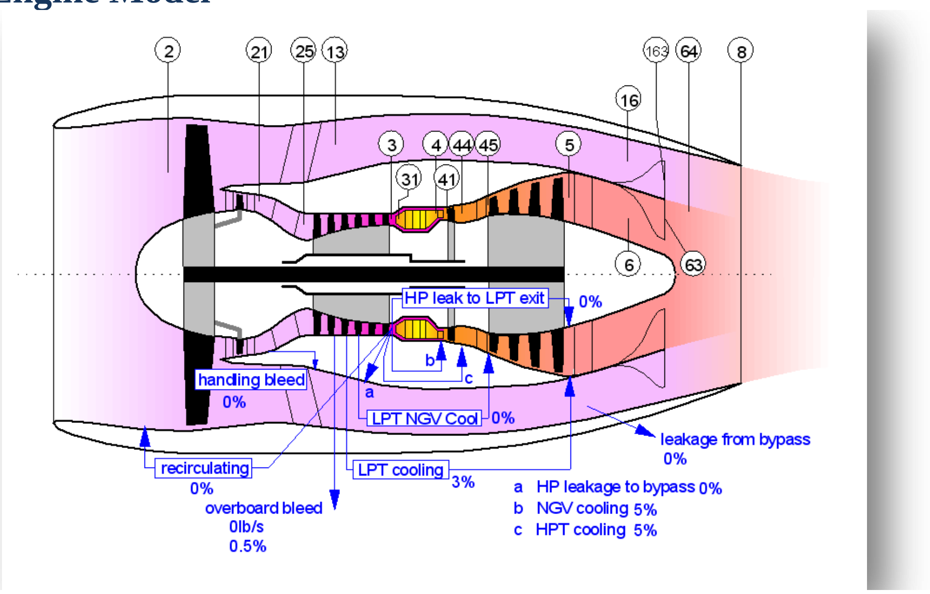

Figure 2: A Mixed Flow, High Bypass Ratio, Turbofan Engine Schematic with Calculation Stations & Cooling Flows

As stated previously, the baseline engine is a *Rolls-Royce AE3007H* or *F137*. A generic model has been generated from publically-available information *(Reference 3)* 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.

The *F137* is a high bypass ratio, twin-spool, mixed, turbofan and *Figure 2* contains a schematic with relevant station numbers.

3.1 Overall Characteristics

Table 2 contains a summary of basic engine characteristics, taken directly from *Reference 3*.

Design Features: AE3007H	
Engine Type	**Axial, turbofan**
Number of fan/compressor stages	**1, 14**
Number of HP/LP turbine stages	**2, 3**
Combustor type	**Annular**
Maximum net thrust at sea level	**8,917 lbf**
Specific fuel consumption at max. power	**0.64 lbm/hr/lbf**
Overall pressure ratio at max. power	**23.0**
Max. envelope diameter	**43.5 inches**
Max. envelope length	**106.5 inches**
Dry weight less tail-pipe	**1,581 lbm**
Current Applications	***Global Hawk UAV***

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 *AE3007H* engine.

Property	Unit	Value	Comment
Intake Pressure Ratio		-0.99	
No (0) or Average (1) Core dP/P		1	
Inner Fan Pressure Ratio		1.4	
Booster Map Type (0/1/2)		0	used for off-design only
Outer Fan Pressure Ratio		1.4	
Compr. Interduct Press. Ratio		0.99	
HP Compressor Pressure Ratio		16.5945	
Bypass Duct Pressure Ratio		0.97	
Turb. Interd. Ref. Press. Ratio		0.98	
Design Bypass Ratio		4.8	
Burner Exit Temperature	R	2880	
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	67.0511	
HP Spool Mechanical Efficiency		0.99	
LP Spool Mechanical Efficiency		0.99	
Burner Pressure Ratio		0.96	
Turbine Exit Duct Press Ratio		0.98	
Hot Stream Mixer Press Ratio		0.99	
Cold Stream Mixer Press Ratio		0.99	
Mixed Stream Pressure Ratio		1	
Mixer Efficiency		0.6	
Design Mixer Mach Number		0.5	
Design Mixer Area	in²	0	

Table 3: Basic Input

Of the four primary design variables, only the overall pressure ratio is given 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 2* allows for a 1% total pressure loss between the inlet leading edge and the fan face. The inner and outer fan pressure ratios are then selected separately; even though there is more blade speed at the fan tip than at its hub, the inner & outer fan pressure ratios have both been set at 1.4 – easily provided by a single-stage machine. A 1% total pressure loss is then accounted for in the duct between the fan and the HP compressor. Knowing that the required overall pressure ratio is 23.0 results in a pressure ratio across the “rear block” of 16.5945, allowing for losses and this also is an input value. Next a 3% total pressure loss is assumed in the bypass duct, followed by an inter-turbine duct loss of 2%. The 16.5945 pressure ratio on the rear block corresponds adequately to the fourteen-stage HP compressor mandated in *Table 2*.

Continuing with the input description, the design bypass ratio was set at 4.8. A value of 2880˚R for the turbine entry temperature was guessed as being reasonable for a military engine of this vintage with limited cooling capacity and an expected long life for the HP turbine (say 5,000 hours). 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 specifically listed 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 *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 5)*, 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 military application currently under consideration, a great deal of auxiliary power is needed for weaponry, avionics & surveillance equipment and this usage is growing rapidly in modern aircraft. We have selected a fairly low power off-take of 67 hp from our engine. You may choose to increase it for your chosen 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 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 8917 lbf given in *Table 2* once the engine scale has been determined. We also have a dimensional envelope to fit into, namely a maximum casing diameter of 43.5 inches and a maximum length of 106.5 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 Mach number of 0.55 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.

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 *Global Hawk* applications, 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.

Station	W lb/s	T R	P psia	WRstd lb/s
amb		518.67	14.696	
1	296.611	518.67	14.696	
2	296.611	518.67	14.549	299.607
13	245.471	574.88	20.369	186.457
21	51.140	578.73	20.369	38.975
25	51.140	578.73	20.165	39.369
3	49.606	1428.18	334.626	3.615
31	44.236	1428.18	334.626	
4	45.299	2880.00	321.241	4.883
41	49.135	2775.49	321.241	5.200
43	49.135	2027.31	68.844	
44	50.413	2013.09	68.844	
45	51.436	1996.37	66.988	22.138
49	51.436	1714.70	33.009	
5	51.947	1709.01	33.009	41.981
6	51.947	1709.01	32.349	
16	245.471	574.88	19.758	
64	297.418	787.56	21.082	
8	297.418	787.56	21.082	255.481
Bleed	0.256	1428.18	334.628	

Efficiencies:	isentr	polytr	RNI	P/P
Outer LPC	0.9297	0.9329	0.990	1.400
Inner LPC	0.8700	0.8760	0.990	1.400
HP Compressor	0.8013	0.8602	1.205	16.595
Burner	0.9995			0.960
HP Turbine	0.8965	0.8780	3.081	4.666
LP Turbine	0.8943	0.8857	0.938	2.029
Mixer	0.6000			

HP Spool mech Eff 0.9900 Nom Spd 16569 rpm
LP Spool mech Eff 0.9900 Nom Spd 6569 rpm

P2/P1= 0.9900 P25/P21=0.9900 P45/P44= 0.9730

hum [%]	war0	FHV	Fuel
0.0	0.00000	18552.4	Generic

FN	=	8918.93	lb
TSFC	=	0.4292	lb/(lb*h)
WF Burner	=	1.06335	lb/s
s NOX	=	0.8599	
BPR	=	4.8000	
Core Eff	=	0.4121	
Prop Eff	=	0.0000	
P3/P2	=	23.000	
P16/P6	=	0.61077	
A63	=	128.30	in²
A163	=	871.94	in²
A64	=	1000.24	in²
XM63	=	0.99381	
XM163	=	0.41600	
XM64	=	0.50000	
P63/P6	=	0.99000	
P163/P16	=	0.99000	
A8	=	876.60	in²
CD8	=	0.91003	
Ang8	=	25.00	°
P8/Pamb	=	1.43452	
WLkBy/W25	=	0.00000	
WCHN/W25	=	0.07500	
WCHR/W25	=	0.02500	
Loading	=	100.00	%
WCLN/W25	=	0.02000	
WCLR/W25	=	0.01000	
WBHD/W21	=	0.00000	
far7	=	0.00359	
WBLD/W25	=	0.00500	
PWX	=	67.1	hp
P16/P13	=	0.9700	
P6/P5	=	0.9800	

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 reduced this to a "*net mass factor*" of 1.2 in *Table 5*, 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* (1679.23 lbm) corresponds reasonably closely to the 1581 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 *AE3007H* baseline engine.

Table 5 is a more detailed "overall output table" from *Gasturb12* for the *RR 3007H* engine.

LP Shaft Thickness	in	0.1
HP Shaft Thickness	in	0.19685
Shaft Material Density	lb/ft³	600
LP Spool Design Spd Incr [%]		0
HP Spool Design Spd Incr [%]		0
Net Mass Factor		1.2
Net Mass Adder	lbm	0

Front LP Shaft Cone Length	in	8.93608
Middle LP Shaft Length	in	45.9499
Middle LP Shaft Radius	in	0.0904418
Rear LP Shaft Cone Length	in	5.2393
Front HP Shaft Cone Length	in	0
Rear HP Shaft Cone Length	in	-1.50113
Rear HP Shaft Length	in	10.1221
Rear HP Shaft Radius	in	1.69422
Engine Length	in	114.912
Max Engine Diameter	in	52.9302
LP Shaft Mass	lbm	9.83435
HP Shaft Mass	lbm	7.63595
Net Mass	lbm	1399.36
Total Mass	lbm	1679.23
LP Spool Inertia	lb*in²	33289.5
HP Spool Inertia	lb*in²	6125.57

	Units	St 2	St 21	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	296.611	51.1398	51.1398	49.6056	45.2993	50.4133	51.4361	51.9474	51.9474	245.471	245.471	297.418	297.4
Total Temperature	R	518.67	578.725	578.725	1428.18	2880	2013.09	1996.37	1709.01	1709.01	574.878	574.878	787.561	787.5
Static Temperature	R	489.035	551.274	551.274	1418.03	2842.64	1932.52	1947.1	1692.21	1675.48	557.125	561.187	750.976	711.7
Total Pressure	psia	14.549	20.3686	20.1649	334.626	321.241	68.8439	66.9877	33.009	32.3488	20.3686	19.7575	21.0816	21.08
Static Pressure	psia	11.8447	17.1729	17.0012	325.714	303.206	58.057	60.3674	31.7097	29.8456	18.2439	18.154	17.7886	14.69
Velocity	ft/s	596.307	575.42	575.419	363.209	753.364	1074.28	839.27	484.32	684.193	462.747	406.362	669.729	963.3
Area	in²	1095.67	152.211	153.749	31.7229	30.0756	83.3377	105.462	305.379	227.4	864.253	996.261	1000.24	797.7
Mach Number		0.55	0.5	0.5	0.2	0.3	0.513877	0.4	0.246597	0.35	0.4	0.35	0.5	0.738
Density	lb/ft³	0.065373	0.084079	0.083239	0.61996	0.287895	0.081086	0.083682	0.050577	0.048079	0.088385	0.087313	0.063933	0.055
Spec Heat @ T	BTU/(lb*R)	0.240085	0.2406	0.2406	0.262222	0.304863	0.287499	0.286909	0.278961	0.278961	0.240556	0.240556	0.244625	0.244
Spec Heat @ Ts	BTU/(lb*R)	0.239991	0.240283	0.240283	0.261901	0.304312	0.285458	0.285651	0.278465	0.277971	0.240351	0.240398	0.243818	0.243
Enthalpy @ T	BTU/lb	-4.31602	10.1327	10.1327	222.261	650.661	391.502	386.45	305.091	305.091	9.20543	9.20537	60.885	60.88
Enthalpy @ Ts	BTU/lb	-11.4219	3.51582	3.51585	219.625	639.319	368.439	372.374	300.403	295.736	4.92618	5.90542	51.9215	42.33
Entropy Function @ T		-0.11924	0.264842	0.264842	3.53056	6.57998	5.01444	4.97663	4.33518	4.33518	0.241421	0.241419	1.35557	1.355
Entropy Function @ Ts		-0.324877	0.094181	0.094182	3.50356	6.5222	4.84403	4.87257	4.29502	4.25464	0.131259	0.156774	1.18573	0.994
Exergy	BTU/lb	-0.357631	12.3969	12.0393	207.918	526.357	268.094	263.415	179.698	178.979	12.3031	11.2192	25.5611	25.56
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	0.068607	0.068
Fuel-Air-Ratio		0	0	0	0	0.024038	0.021547	0.02111	0.020897	0.020897	0	0	3.5881E-3	3.588
Water-Air-Ratio		0	0	0	0	0	0	0	0	0	0	0	0	0
Inner Radius	in	8.15053	8.97187	4.03897	7.55236	8.37794	8.37794	8.79683	8.79683	0	11.8361	8.7047	0	0

Table 5: Baseline Engine Detailed Output

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

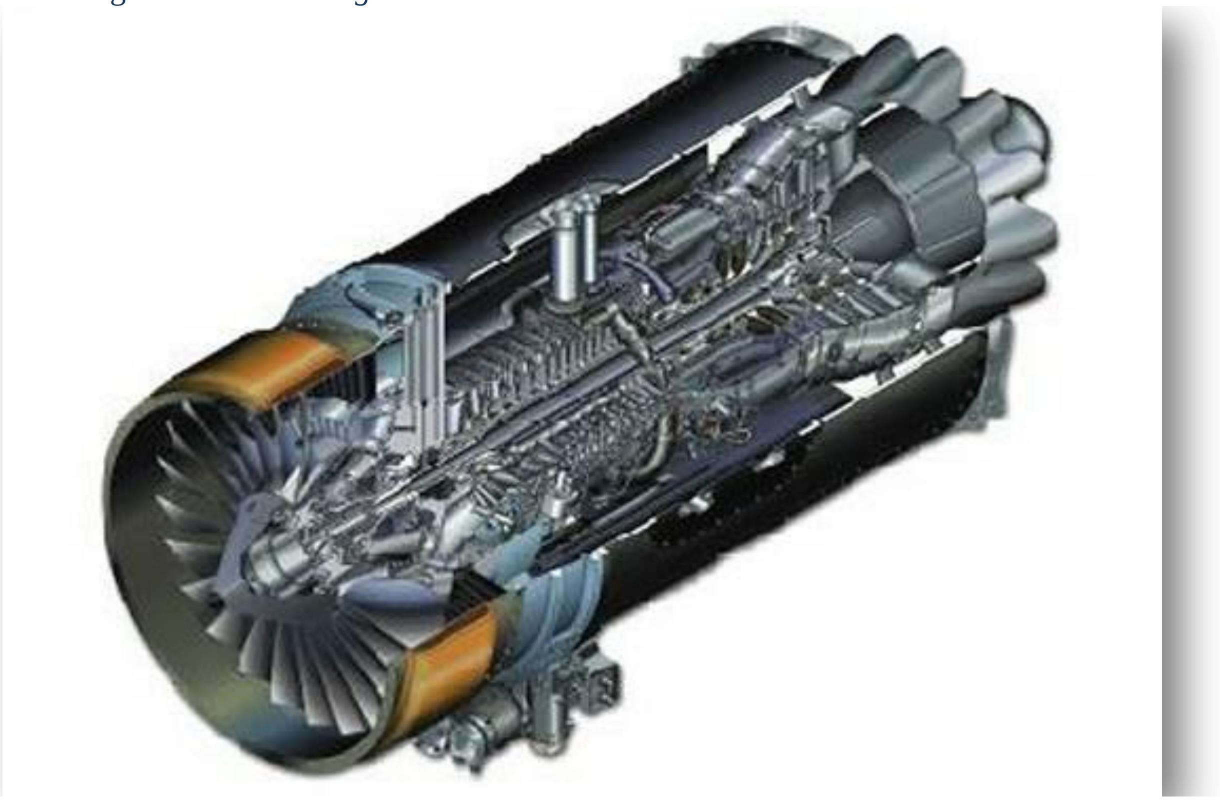

Figure 3: The AE3007H Baseline Engine

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

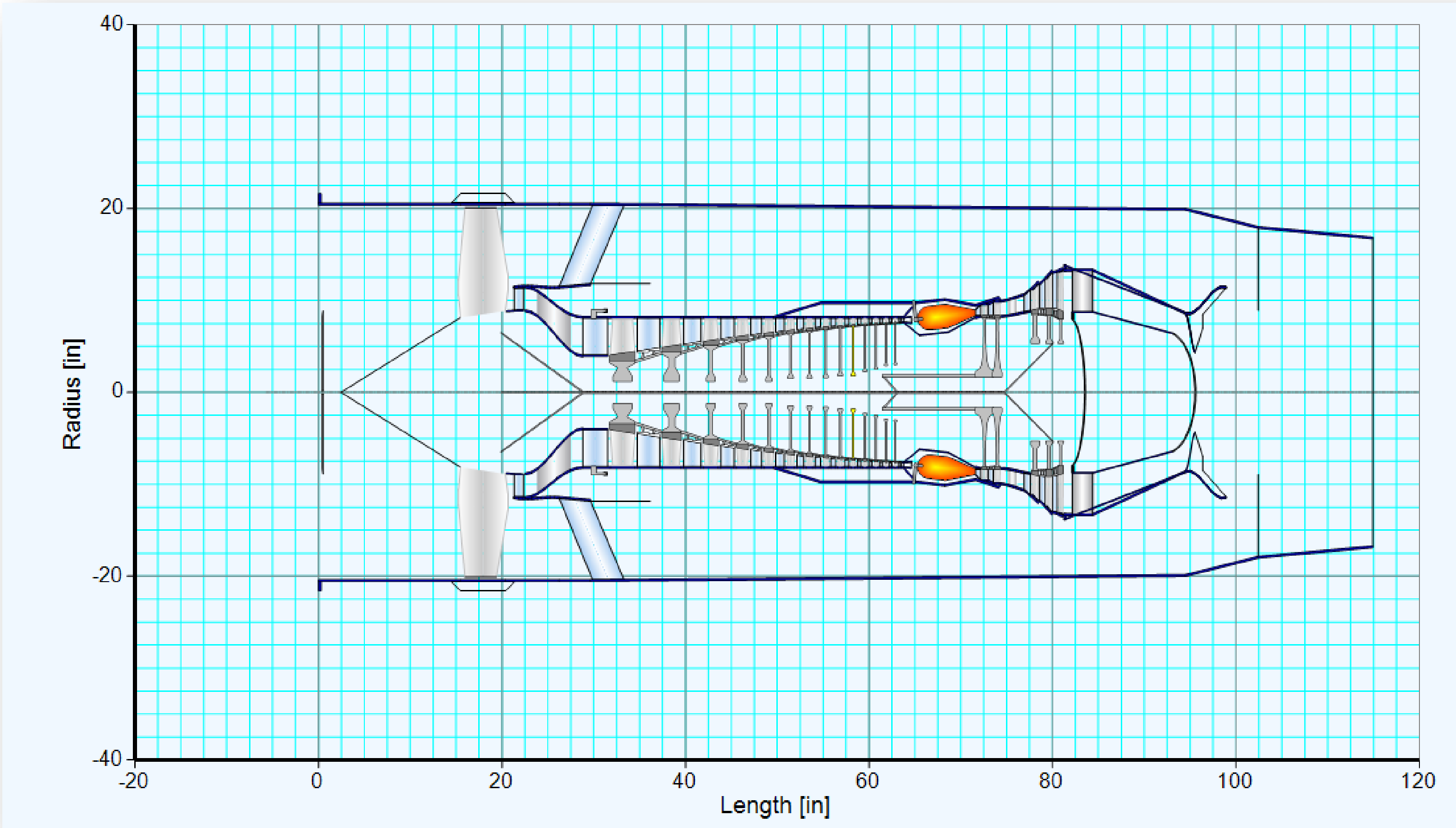

Figure 4: GasTurb12 Model of the Baseline Engine

Some details of the component models now follow.

3.2 Inlet

The inlet is designed with a conic 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		8
Strut Chord/Height		0
Gap Width/Height		0.2
Cone Length/Radius		1.6
Cone Angle [deg]		50
Casing Length/Radius		0.3
Casing Thickness	in	0.19685
Casing Material Density	lb/ft³	167.616
Inlet Mass Factor		1

Length	in	15.486
Cone Length	in	13.0408
Cone Mass	lbm	7.5189
Casing Mass	lbm	37.8576
Strut Mass	lbm	0
Total Mass	lbm	45.3765

Table 6: Inlet Design

Pertinent characteristics of the inlet are shown in *Table 6*. At 45.4 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

Input:		
LPC Tip Speed	ft/s	1168.00
LPC Inlet Radius Ratio		0.40000
LPC Inlet Mach Number		0.55000
Engine Inl/Fan Tip Diam Ratio		1.00000
min LPC Inlet Hub Diameter	in	0.00000
Output:		
LPC Tip circumf. Mach No		1.07730
LPC Tip relative Mach No		1.20957
Design LP Spool Speed	[RPM]	6568.55
LPC Inlet Tip Diameter	in	40.75264
LPC Inlet Hub Diameter	in	16.30105
Calculated LPC Radius Ratio		0.40000
LP Spool Torque	lb*ft	4634.11
Aerodynamic Interface Plane	in²	1304.37
Corr.Flow/Area LPC	lb/(s*ft²)	39.37619

Table 7: Fan: Detailed Overview

Note that the value of the tip speed at the top of *Table 7*, in conjunction with the fan tip radius, that sets the rotational speed of the LP spool.

Number of Stages		1
Inlet Guide Vanes (IGV) 0/1		0
IGV Profile Thickness [%]		5
IGV Material Density	lb/ft³	249.712
Annulus Shape Descriptor 0...1		1
Inlet Radius Ratio		0.3
Core Aspect Ratio Span/Chord		2.5
Bypass Vane Aspect Ratio		2.5
Core Vane Gap/Chord Ratio		0.2
Bypass Gap/Chord Ratio		1.2
Rotor Pitch/Chord Ratio		1
Core Vane Pitch/Chord Ratio		1
Bypass Vane Pitch/Chord Ratio		1
Disk Bore / Inner Inlet Radius		0.8
Rel Thickness Inner Air Seal		0.04
Casing Thickness	in	0.19685
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

Length	in	7.06532
Number of Inlet Guide Vanes		0
Number of Bypass Stream Vanes		8
Number of Core Stream Vanes		64
Total Number of Blade and Vanes		90
Outer Casing Mass	lbm	47.6465
Containment Ring Mass	lbm	18.4359
Splitter Mass	lbm	17.2445
Bypass Vane Mass	lbm	5.76072
Vane Mass	lbm	1.14725
Blade Mass	lbm	125.435
Inner Air Seal Mass	lbm	0
Rotating Mass	lbm	218.902
IGV Mass	lbm	0
Total Mass	lbm	309.329
Polar Moment of Inertia	lb*in²	25973

Table 8: Fan General Output

3.4 Inter-Compressor Duct

Number of Struts		8
Length/Inlet Inner Radius		0.7
Inner Annulus Slope@Exit [deg]		0
Relative Strut Length [%]		60
Casing Thickness	in	0.19685
Casing Material Density	lb/ft³	499.424
Compr Interduct Mass Factor		1

Length	in	6.28031
Outer Casing Mass	lbm	24.606
Strut Mass	lbm	11.0151
Inner Casing Mass	lbm	18.5714
Total Mass	lbm	54.1925

Table 9: Inter-Compressor Duct

Notice that instead of 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.

3.5 High Pressure Compressor

Input:		
HPC Tip Speed	ft/s	1168.00
HPC Inlet Radius Ratio		0.50000
HPC Inlet Mach Number		0.50000
min HPC Inlet Hub Diameter	in	0.00000
Output:		
HPC Tip circumf. Mach No		1.01491
HPC Tip relative Mach No		1.13139
Design HP Spool Speed	[RPM]	16568.94
HPC Inlet Tip Diameter	in	16.15588
HPC Inlet Hub Diameter	in	8.07794
Calculated HPC Radius Ratio		0.50000
HP Spool Torque	lb*ft	4873.68
Corr.Flow/Area HPC	lb/(s*ft²)	36.87239

Table 10: High Pressure Compressor - Detailed Overview

Again, we set the speed of the HP spool via the tip speed and the corresponding radius. The definition of the HP compressor annulus and its general characteristics are given in *Tables 11* and *12*.

		IGV	Stage 1	Stage 2	Stage 3	Stage 4	Stage 5	Stage 6	Stage 7	Stage 8	Stage 9	Stage 10	Stage 11	Stage 12	Stage 13	Stage 1
ri Rotor Inlet	in		4.039	4.854	5.513	6.015	6.396	6.686	6.91	7.085	7.224	7.342	7.431	7.504	7.564	7.614
ri Rotor Exit	in		4.446	5.183	5.764	6.205	6.541	6.798	6.998	7.155	7.283	7.387	7.468	7.534	7.589	7.636
ri Stator Exit	in	4.039	4.854	5.513	6.015	6.396	6.686	6.91	7.085	7.224	7.342	7.431	7.504	7.564	7.614	7.657
ro Rotor Inlet	in		8.078	8.078	8.078	8.078	8.078	8.078	8.078	8.078	8.078	8.078	8.078	8.078	8.078	8.078
ro Rotor Exit	in		8.078	8.078	8.078	8.078	8.078	8.078	8.078	8.078	8.078	8.078	8.078	8.078	8.078	8.078
ro Stator Exit	in	8.078	8.078	8.078	8.078	8.078	8.078	8.078	8.078	8.078	8.078	8.078	8.078	8.078	8.078	8.078
ri/ro Rotor Inlet			0.5	0.6009	0.6824	0.7446	0.7918	0.8277	0.8554	0.8771	0.8943	0.9089	0.92	0.929	0.9364	0.9425
ri/ro Stator Exit		0.5	0.6009	0.6824	0.7446	0.7918	0.8277	0.8554	0.8771	0.8943	0.9089	0.92	0.929	0.9364	0.9425	0.9479
x Rotor Inlet	in		31.79	37.45	41.96	45.59	48.57	51.05	53.17	55.01	56.63	58.08	59.39	60.6	61.72	62.77
x Rotor Exit	in		34.49	39.54	43.59	46.87	49.59	51.88	53.85	55.57	57.1	58.48	59.74	60.9	61.98	63
x Stator Inlet	in	28.83	34.76	39.81	43.86	47.14	49.86	52.15	54.12	55.84	57.37	58.75	60	61.17	62.25	63.27
x Stator Exit	in	31.52	37.18	41.69	45.33	48.3	50.78	52.9	54.74	56.36	57.81	59.12	60.33	61.45	62.5	63.49
No of Blades			28	39	52	69	89	113	140	170	203	243	284	328	375	425

Table 11: High Pressure Compressor - Annulus

Number of Stages		14
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		1
Given Radius Rat: Inl/Exit 0/1		0
Inlet Radius Ratio		0.5
Exit Radius Ratio		0.9
First Stage Aspect Ratio		1.5
Last Stage Aspect Ratio		2
Blade Gapping: Gap/Chord		0.1
Pitch/Chord Ratio		0.5
Disk Bore / Inner Inlet Radius		0.3
Diffuser Area Ratio		1.5
Rel Thickness Inner Air Seal		0.04
Compressor Mass Factor		1
Outer Casing Thickness	in	0.19685
Outer Casing Material Density	lb/ft³	249.712
Casing Thickness	in	0.19685
Casing Material Density	lb/ft³	249.712
Rel Work of Radial End Stage		0.3
Rad Diffusor/Rotor Blade Length		0.5
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	34.9274
Number of Inlet Guide Vanes		28
Total Number of Blade and Vanes		5326
Diffusor Length	in	0.856098
Casing Mass	lbm	50.4288
Outer Casing Mass	lbm	26.4438
Total Vane Mass	lbm	35.7742
Total Blade Mass	lbm	65.8972
Inner Air Seal Mass	lbm	8.22284
Rotating Mass	lbm	150.158
IGV Mass	lbm	5.92452
Exit Diffusor Mass	lbm	2.42584
Total Mass	lbm	271.155
Polar Moment of Inertia	lb*in²	4742.79

Table 12: High Pressure Compressor - General Output

3.6 Combustor

A fairly conventional annular combustor is used and details are given in *Table 13*. 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		2
Exit/Inlet Radius		1.1
Length/Inlet Radius		1
Can Width/Can Length		0.4
Inner Casing Thickness	in	0.0787402
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*R)	0.119503
Can Time Constant		1
Mass of Fuel Inj. / Fuel Flow		2
Burner Mass Factor		1

Mean Radius, Exit	in	8.65448
Length	in	7.86771
Can Volume	in³	638.009
Can Mass	lbm	41.8417
Can Surface Area / Mass	in²/lbm	35.1535
Fuel Injector Mass	lbm	2.1267
Inner Casing Mass	lbm	8.85193
Outer Casing Mass	lbm	29.6473
Total Mass	lbm	82.4677
Can Heat Soakage	hp	0

Table 13: Combustor

3.7 High-Pressure Turbine

Property	*Unit*	*Value*
1. HPT Rotor Inlet Dia	in	15.748
Last HPT Rotor Exit Dia	in	15.748
HPT Exit Radius Ratio		0.8
HPT Vax.exit / Vax.average		1.05
HPT Loss Factor [0.3...0.4]		0.35
HPT 1. Rotor Cooling Constant		0
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 14*, which is made available once that efficiency option is selected.

The useful summary of the HP turbine presented in *Table 15* also appears as a result of that selection.

Input:			
Number of Stages			2
1. HPT Rotor Inlet Dia	in		15.74803
Last HPT Rotor Exit Dia	in		15.74803
HPT Exit Radius Ratio			0.80000
HPT Vax.exit / Vax.average			1.05000
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			0.94509
HPT First Stator Exit Angle			60.39524
HPT Exit Mach Number			0.60118
HPT Exit Angle			-31.26498
HPT Last Rotor abs Inl Temp	R		2400.77
HPT First Rotor rel Inl Temp	R		2588.13
HPT First Stage H/T	BTU/(lb*R)		0.03987
HPT First Stage Loading			2.13595
HPT First Stage Vax/u			0.93545
HPT Exit Tip Speed	ft/s		1265.02
HPT Exit A*N*N	in²*RPM²*E-6		23765.64
HPT 1.Rotor Cool.Effectiveness			0.00000
HPT 1.Rotor Bld Metal Temp	R		2588.13
Velocities:			
1st Stage Inlet Absolute Velocity	V	ft/s	2078.72
1st Stage Inlet Axial Velocity	Vax	ft/s	1065.02
1st Stage Inlet Relative Velocity	W	ft/s	1245.97
1st Circumferential Velocity	U	ft/s	1138.51
1st Stage Exit Absolute Velocity	V	ft/s	1245.97
1st Stage Exit Axial Velocity	Vax	ft/s	1065.02
1st Stage Exit Relative Velocity	W	ft/s	2078.72
Last Stage Inlet Absolute Velocity	V	ft/s	2053.19
Last Stage Inlet Axial Velocity	Vax	ft/s	1014.30
Last Stage Inlet Relative Velocity	W	ft/s	1202.89
Last Circumferential Velocity	U	ft/s	1138.51
Last Stage Exit Absolute Velocity	V	ft/s	1245.96
Last Stage Exit Axial Velocity	Vax	ft/s	1065.01
Last Stage Exit Relative Velocity	W	ft/s	2078.71

Table 15: HPT Summary

A general summary of the HP turbine is given in *Table 16*, followed by the velocity diagrams and *Smith Chart* in *Figure 5*.

Number of Stages = 2		no input
Unshrouded/Shrouded Blades 0/1		0
Inner Radius: R,exit / R,inlet		1
Inner Annulus Slope@Inlet[deg]		0
Inner Annulus Slope@Exit [deg]		0
First Stage Aspect Ratio		1
Last Stage Aspect Ratio		1
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
Outer Casing Thickness	in	0.19685
Outer Casing Material Density	lb/ft³	499.424
Casing Thickness	in	0.19685
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	2.76544
Total Number of Blade and Vanes		405
Casing Mass	lbm	9.75791
Outer Casing Mass	lbm	10.945
Total Vane Mass	lbm	1.88549
Total Blade Mass	lbm	5.90699
Inner Air Seal Mass	lbm	0.0484538
Rotating Mass	lbm	41.1271
Total Mass	lbm	63.7155
Polar Moment of Inertia	lb*in²	1382.78

Table 16: High Pressure Turbine – General Output

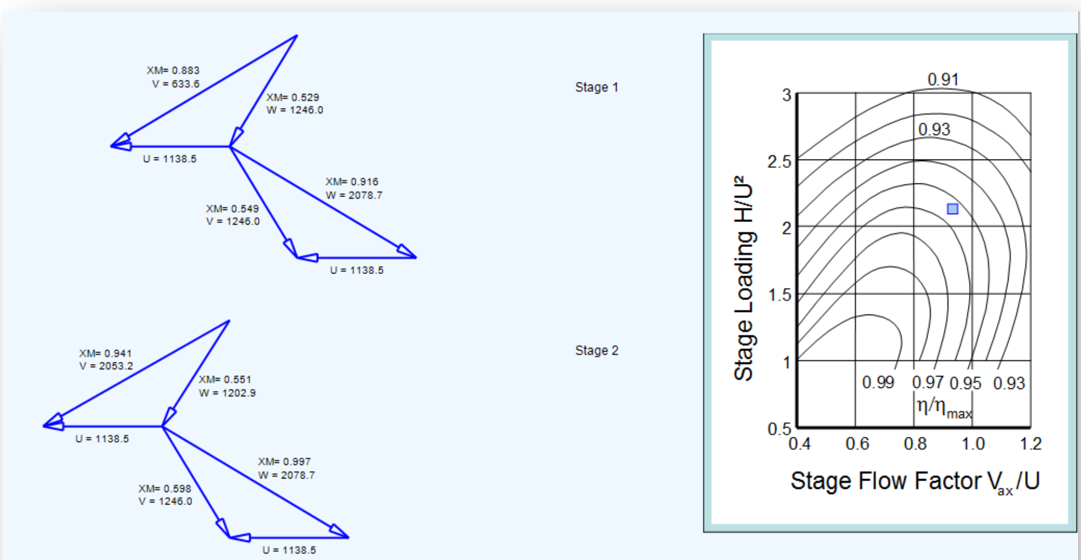

Figure 5: High Pressure Turbine Velocity Diagrams & Smith Chart

3.8 Inter-Turbine Duct

Table 17 contains details of the inter-turbine duct. Its relatively short length allows the two turbines to be close-coupled but the exit-to-inlet radius ratio of 1.05 moves the LP turbine outwards and generally reduces the LPT stage loading coefficients and increases efficiency.

Number of Struts		8
Exit/Inlet Inner Radius		1.05
Length/Inlet Inner Radius		0.3
Inner Annulus Slope@Inlet[deg]		0
Inner Annulus Slope@Exit [deg]		10
Relative Strut Length [%]		40
Casing Thickness	in	0.19685
Casing Material Density	lb/ft³	499.424
Turbine Interduct Mass Factor		1

Length	in	2.51338
Outer Casing Mass	lbm	9.49716
Strut Mass	lbm	1.46114
Inner Casing Mass	lbm	7.82187
Total Mass	lbm	18.7802

Table 17: Inter-Turbine Duct

3.9 Low-Pressure Turbine

Characteristics of the low pressure turbine are presented in *Tables 18* - 20 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*. 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 89.43%, which corresponds to the value in the engine performance summary in *Table 4*.

Property	*Unit*	*Value*
LPT with EGV's [0/1]		1
1. LPT Rotor Inlet Dia	in	19.685
Last LPT Rotor Exit Dia	in	19.685
LPT Exit Radius Ratio		0.6
LPT Vax.exit / Vax.average		1
LPT Loss Factor [0.3...0.4]		0.35
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	19.68504
Last LPT Rotor Exit Dia		in	19.68504
LPT Exit Radius Ratio			0.60000
LPT Vax.exit / Vax.average			1.00000
LPT Loss Factor [0.3...0.4]			0.35000
LPT 1. Rotor Cooling Constant			0.00000
Output:			
LPT Inlet Radius Ratio			0.75325
LPT First Stator Exit Angle			61.00125
LPT Exit Mach Number			0.29220
LPT Exit Angle			-32.44261
LPT Last Rotor abs Inl Temp		R	1808.33
LPT First Rotor rel Inl Temp		R	1949.36
LPT First Stage H/T		BTU/(lb*R)	0.01331
LPT First Stage Loading			2.08802
LPT First Stage Vax/u			0.85581
LPT Exit Tip Speed		ft/s	705.23430
LPT Exit A*N*N		$in^{2}*RPM^{2}*E\text{-}6$	13131.13
LPT 1.Rotor Cool.Effectiveness			0.00000
LPT 1.Rotor Bld Metal Temp		R	1949.36
LPT Torque		lb*ft	4634.11
Velocities:			
1st Stage Inlet Absolute Velocity	V	ft/s	995.98
1st Stage Inlet Axial Velocity	Vax	ft/s	482.84
1st Stage Inlet Relative Velocity	W	ft/s	572.13
1st Circumferential Velocity	U	ft/s	564.19
1st Stage Exit Absolute Velocity	V	ft/s	572.13
1st Stage Exit Axial Velocity	Vax	ft/s	482.84
1st Stage Exit Relative Velocity	W	ft/s	995.98
Last Stage Inlet Absolute Velocity	V	ft/s	995.98
Last Stage Inlet Axial Velocity	Vax	ft/s	482.84
Last Stage Inlet Relative Velocity	W	ft/s	572.13
Last Circumferential Velocity	U	ft/s	564.19
Last Stage Exit Absolute Velocity	V	ft/s	572.13
Last Stage Exit Axial Velocity	Vax	ft/s	482.84

Table 19: LPT Summary

Number of Stages = 3		no input
Unshrouded/Shrouded Blades 0/1		1
Inner Radius: R,exit / R,inlet		1
Inner Annulus Slope@Inlet[deg]		0
Inner Annulus Slope@Exit [deg]		-26.3
First Stage Aspect Ratio		2.5
Last Stage Aspect Ratio		3.5
Blade Gapping: Gap/Chord		0.25
Pitch/Chord Ratio		0.5
Disk Bore / Inner Inlet Radius		0.6
Rel Thickness Inner Air Seal		0.04
LP Turbine Mass Factor		1
Casing Thickness	in	0.19685
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	4.37465
Total Number of Blade and Vanes		1592
Casing Mass	lbm	21.7744
Total Vane Mass	lbm	21.836
Total Blade Mass	lbm	51.8832
Inner Air Seal Mass	lbm	0.0941532
Rotating Mass	lbm	92.9942
Total Mass	lbm	136.605
Polar Moment of Inertia	lb*in²	7316.51

Table 20: Low Pressure Turbine: General Output

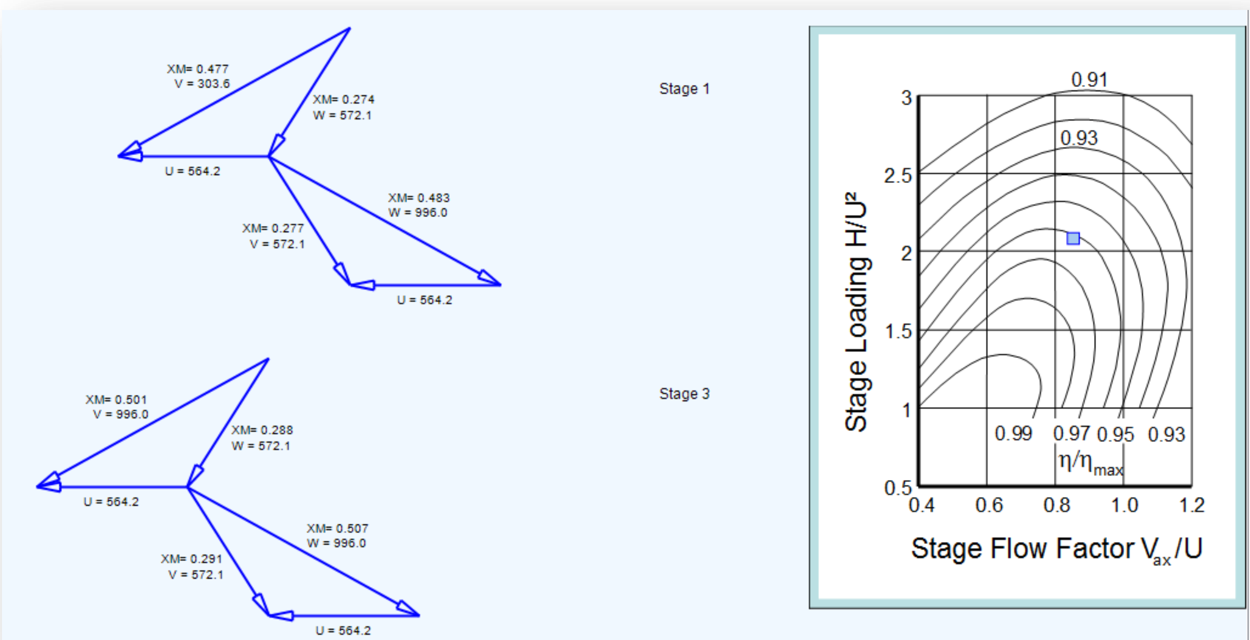

Figure 6: Low Pressure Turbine Velocity Diagrams & Smith Chart

3.10 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 14, the core exhaust extends to about 95 inches. It is important to note that in *GasTurb12* the core exhaust does not include the convergent portion or core nozzle. *Table 21* below contains the input and output details of the core exhaust.

Number of Struts		8
Strut Chord/Height		0.5
Strut Lean Angle		0
Gap Width/Height		0.2
Cone Angle [deg]		15
Cone Length/Inlet Radius		1
Casing Length/Inlet Radius		1
Inner Casing Thickness	in	0.0787402
Outer Casing Thickness	in	0.19685
Casing Material Density	lb/ft³	499.424
Exhaust Duct Mass Factor		1

Length	in	13.2132
Cone Length	in	8.79683
Outer Casing Mass	lbm	53.7648
Strut Mass	lbm	8.87745
Cone Mass	lbm	12.6983
Front Cover Mass	lbm	4.98574
Total Mass	lbm	80.3262

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 115 inches in *Figure 4*. The casing material density in the core nozzle is significantly less than that for the core exhaust owing to the prevailing temperatures; the values can be found in *Tables 21* and *22*. By the time the air flow reaches the core nozzle, the hot and cold flows have mixed and the resulting lower temperature can be sustained with a less dense material.

Std/Plug/Power Gen Exh 1/2/3		1
Inl Section Length/Outer Radius		inactive
Conv Length/Inl Section Radius		0.7
Cone Angle [deg]		15
Cone Length/Inlet Radius		0.4
Inlet Section Area Ratio		inactive
Divergent Length/Throat Radius		inactive
Inner Casing Thickness	in	0.0787402
Outer Casing Thickness	in	0.19685
Casing Material Density	lb/ft³	499.424
Nozzle Mass Factor		1

Overall Length	in	12.4904
Inlet Section Length	in	0
Convergent Length	in	12.4904
Divergent Length	in	0
Convergent Cone Angle [deg]		5.21112
Divergent Cone Angle [deg]		0
Inlet Section Mass	lbm	0
Convergent Section Mass	lbm	77.4466
Divergent Section Mass	lbm	0
Inner Casing Mass	lbm	0
Outer Casing Mass	lbm	77.4466
Total Mass	lbm	77.4466

Table 22: Core Nozzle

3.11 Bypass Duct & Mixer

Tables 23 and *24* describe 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		70
Flat Point Radius/Inlet Radius		1
Strut Inlet Pos in % of Length		50
Relative Strut Length [%]		20
Mean Strut Thickness	in	0.0787402
Strut Material Density	lb/ft³	249.712
Inner Casing Thickness	in	0
Outer Casing Thickness	in	0.19685
Casing Material Density	lb/ft³	249.712
Bypass Duct Mass Factor		1

Outer Casing Length	in	51.6194
Inner Casing Length	in	51.6194
Outer Casing Mass	lbm	185.448
Inner Casing Mass	lbm	0
Strut Mass	lbm	8.10122
Total Mass	lbm	193.549

Table 23: Bypass Duct

Length/Diameter		0.2
Number of Chutes		12
Chute Height [%]		40
Chute Thickness	in	0.15
Chute Material Density	lb/ft³	499.424
Casing Thickness	in	0.19685
Casing Material Density	lb/ft³	249.712
Mixer Mass Factor		1

Length	in	7.9286
Chute Mass	lbm	21.4381
Casing Mass	lbm	27.506
Cone Mass	lbm	0
Total Mass	lbm	48.944

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.

- The efficiencies of the turbomachinery components may be assumed to be the same as those of the baseline engine, and 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. If you have them, use them but reasonable assumptions regarding thrust requirements of the Global Hawk throughout the mission are quite acceptable.

- The use of design codes from industrial 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, 2012 on pages 31 and 33, it is recommended that teams who know that they will enter the competition inform either AIAA, ASME-IGTI or 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 30).

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

5. Competition Expectations

The existing rules and guidelines for the Joint *AIAA Foundation/ASME.IGTI Student Design Competition* should be observed and these are provided in *Appendix 2*. In addition, the following specific suggestions are offered for the event.

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 critical mission point(s) that drive the candidate propulsion system design(s).

- 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 counts.

- 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.

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

References

1. *<http://en.wikipedia.org/wiki/Northrop_Grumman_RQ-4_Global_Hawk>*

2. "*GE Tests CMCs for Future Engine*"
Aviation Week & Space Technology. July 30, 2012.

3. "*Aerospace Source Book.*"
Aviation Week & Space Technology. January 15, 2007.

4. *"GasTurb 11: A Design & Off-Design Performance Program for Gas Turbines"*
<http://www.gasturb.de>
Joachim Kurzke, 2007.

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

6. *"Aeronautical Vest Pocket Handbook". Pratt & Whitney Aircraft.* Circa 1980

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.

Available Software & Additional Reference Material

GasTurb 12 is a comprehensive code for the preliminary design of propulsion and industrial gas turbine engines. It encompasses design point and off-design performance, based on extensive libraries of engine architectures and component performance maps, all coupled to impressive graphics. A materials database and plotting capabilities enable a detailed engine model to be generated, with stressed disks and component weights. A student license for this code is available at a very low price directly from the author (*Reference 4*) strictly for academic work only.

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

Once a *Letter of Intent* has been received, the names of team members will be recognized as being eligible to be granted access to the materials above. Students must then approach *SoftInWay Inc*. to obtain an *AxSTREAM* license. *SoftInWay Inc* will not contact team members.

All the offers above are subject to *ITAR* restrictions.

Appendix 1. Letter of Intent

2012/2013
Joint AIAA–IGTI Undergraduate Team Engine Design Competition

Request for Proposal:
An Improved Engine for a High Altitude Long Endurance Unmanned Air Vehicle

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 2012/2013 Joint AIAA-IGTI Undergraduate Team Engine Design Competition, you must complete this form and return it to the AIAA Director of Student Programs ***before November 1, 2012***, at AIAA Headquarters, along with a one-page "Letter of Intent", 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 or ASME branches or at-large Student Members are eligible and encouraged to participate.

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

3. An electronic copy of the report in MS Word or Adobe PDF format must be submitted on a CD or DVD to AIAA Student Programs. Total size of the file(s) cannot exceed 60 MB, which must also fit on 50 pages when printed. **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 direct 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 and IGTI organizing committees from the 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 *ASME TurboExpo Turbine Technical Conference & Exposition* in San Antonio, Texas on Wednesday, June 5, 2013. Airfare and lodging expenses will be partially covered for the invited teams and their advisors. The results of the presentations will be combined with the earlier scores to determine first, second and third places.

8. The prizes shall be: First place-$2,500; Second place-$1,500; Third place-$1,000 (US dollars). 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 three competing teams will be introduced and acknowledged at the *TurboExpo2013* closing ceremony on Thursday, June 6, 2013.

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 & Activity Sequences

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

A. Letter of Intent – November 1, 2012
B. Receipt of Proposal – April 1, 2013
C. Proposal evaluations completed - April 30, 2013
D. Proposal presentations & Announcement of Winners at ASME TurboExpo in San Antonio, TX; June 5, 2013

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, to the following address:

AIAA Student Programs
1801 Alexander Bell Drive
Suite 500
Reston, VA 20191-4344

The CD containing the finished 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 maximum presentation score will be *50 points*.

6. The top 3 finalists will present their work at the 2013 TURBO EXPO the week of June 3 – 7 at the San Antonio Convention Center in San Antonio, Texas

The student fee to attend Turbo Expo is $220. This fee covers the following:

- Access to EVERY session in the TURBO EXPO Technical Conference
- DVD Proceedings comprised of all papers published for TURBO EXPO 2013
- Certificate of Completion for Professional Development Hours (PDHs)
- Admission to the Opening Session, IGTI Awards Program, Keynote Session and Welcome Reception
- Complimentary daily lunches
- Unlimited access to the Exhibit Hall during the 3-day Exposition (June 4-6, 2013)
- Two Exhibit Hall receptions with complimentary refreshments
- Opportunity to attend Facility Tours

To register for Turbo Expo 2013 please visit:
http://www.asmeconferences.org/TE2013//