**VULCAN IIL**:

THE WAVERIDER SUBORBITAL TRANSPORT AIRCRAFT.

by Kay Runne,

Independent Aerospace Researcher and Consultant.

[www.ramwave.eu](http://www.ramwave.eu)

KayRunne@msn.com

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Summary.

As an offspring of a pre-preliminary project VULCAN II for a SSTO spaceplane, also on the same level a project for a suborbital transport aircraft is proposed. It consists also of an integrated wing-body structure of silicon-carbon fiber, (SiC), with a double delta plan form developed as a waverider. It is therefore designated here as the VULCAN IIL, (L for Light).

Its propulsion is also conceived as a pre-cooled turbo-ramjet system like the SABRE engine under development by *Reaction* *Engines Ltd.*, by using also nanotube SiC heat exchanger technology, with a further development by extending the ramjet flight phase until M = 8, but without rocket propulsion. The helium heat exchanger technology is also applied for active skin cooling at the leading edges and the non-retractable air intakes of the vehicle.

The vehicle is to be designed for the transport of 300 passengers over a maximum range of 20 000 km. Based on the preliminary pre-project of the VULCAN II waverider spaceplane it is showed here with an assessment of masses, aerodynamics, propulsion and resulting performance, that the realization of such a project is feasible, provided that the assumptions, made in this assessment, like for the VULCAN II, are verified with appropriate technology research programs in the coming years, for which it serves as a focal point.

A comparison of the required fuel, depending of the range, is made with that of a hypothetical advanced subsonic airliner.

It would open up a new traffic market in addition of that of subsonic long range air traffic. In order to get a first quantitative idea a transportation efficiency is defined and compared for both systems.

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1. Introduction.

As an offspring of possible spaceplane development and enhanced by recent space tourism projects for short time parabolic weightless flight to high altitude, the idea of very high speed suborbital transport flight finds a renewed interest. The main first hand arguments supporting this interest, beside the high speed, are low drag at very high altitude and the partial offset of the weight by the centrifugal force. However the question is if a sufficient market can be found with an economically interesting balance between the fuel quantity needed to obtain the high installed thrust required for acceleration climb to cruise altitude and that of the low required thrust during cruise flight. It is a challenge to find a confirmative answer on this question. Let’s take it.

Based largely on the VULCAN II Waverider Spaceplane preliminary project proposal, ref.[1], and being a “light” derivative design of it, a preliminary project proposal for a suborbital waverider transport aircraft is presented here, accordingly branded here with the designation VULCAN IIL.

The vehicle is to be designed for the transport of 300 passengers over a maximum range of 20 000 km, corresponding to a trip from Europe to New Zealand.

Under the same conditions, assumptions and with the same context and references as for the VULCAN II an assessment of masses, aerodynamics, propulsion and resulting performance is made here in order to show that the realization of such a project could be feasible, provided that these assumptions are verified with appropriate technology research programs in the coming years, for which it serves together with the VULCAN II as a focal point.

2. General Description.

Like the VULCAN II it consists of an integrated wing-body but only partially active cooled structure of silicon-carbon fiber, (SiC), with a double delta wing-body plan form developed as a waverider. The main difference from the VULCAN II consists of the much larger passenger cabin with an accommodation for 300 passengers and the subsequent reduction of the fuel tank volume, corresponding to the required liquid hydrogen capacity only. In figs.1 and 2 simplified transparent views of the VULCAN IIL are presented.

Also its propulsion system, differing from that for the VULCAN II in the geometrical arrangement of the air breathing intake part and without rocket propulsion, is conceived as a further development off-spring of the SABRE engine under development by *Reaction* *Engines Ltd.*, ref.[2], by using also nanotube SiC heat exchanger technology with helium coolant, but extending the ramjet flight phase until cruise flight at M = 8, as with the VULCAN II.

The heat exchanger technology with helium coolant in a close circuit is also applied for active skin cooling, but only at the leading edges and the non-retractable air intakes, so at a much lower extent as with the VULCAN II.

Since the VULCAN IIL only flies in the atmosphere, a reaction control system is not required.

LH2

He

Propulsion System

Double Wall Pressure Cabin

**Top View**

**Side View**

Fig.1:

Cutaway system sketch of top & side view of VULCAN IIL.

 **Front View**

**Rear View**

 Fig.2: Cutaway system sketch of front & rear view of VULCAN IIL.

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Prandtl-Meyer axis

tanks

Prandtl-Meyer flow (M > 1)

straight shock

M < 1

to and from structure cooling

ø

□

M > 1

central body with ejector nozzle

To other engine

Fig. 3: Schematic functional view of the propulsion system.

Fig. 3: Schematic functional view of the propulsion system

Explication of Symbols in fig. 3:

= liquid pumps

 = valves

= heat exchangers

= air compressor

= helium compressor

= helium turbine

~

= starter/generator

 = helium cooled structure

 = flexible ramp (morphing)

3. Dimensions.

In table 1 the estimation of the to be expected main dimensions is presented.

Table 1: Main Dimensions

|  |  |  |  |
| --- | --- | --- | --- |
| Length over all |  l | 100 |  m |
| Span |  b | 75 |  m |
| Height over ground |  Hog | 37 |  m |
| Aerodynamic reference area |  Aref | 2800 |  m2 |
| Wetted Surface reference area |  WSref | 7200 |  m2 |
| Aspect ratio | =b2/Aref | 2 |   |
| Engine main frame diameter |  demf | 5 |  m |
| Diameter of pressure cabin |  dcab | 6,25 |  m |
| Length of pressure cabin |  lcab | 61 |  m |

The aerodynamic reference area is defined as the total vertical projected wing-body area.

The pressure cabin contains the passenger cabin, cockpit, kitchen, toilets, emergency equipment, luggage and cargo bay.

4. Masses.

The mass estimation, presented in table 2, is based on the already mentioned integrated SiC wing-body structure with partial active cooling, including the propulsion system, H2 and He tanks, landing gear and further equipment. The structure mass estimation is made with ref.[3], taking into account a 25% reduction of the unity mass data for aluminum, (kg/m2), indicated by ref.[3], corresponding to the use of SiC instead.

It is assumed, that the estimated quantity of helium covers the requirements of the turbines, the heat exchangers and the partial structure cooling, taking into account their only partial simultaneous operation during the different flight phases.

The estimated mass data of table 2 for the hydrogen mass, depending of the range, have been assessed and verified with the flight performance calculation of chapter 7.

Table 2: Mass estimation.

|  |  |  |  |  |  |  |
| --- | --- | --- | --- | --- | --- | --- |
|  |  | R [km] = | 20000 | 15000 | 10000 | 5000 |
| Item | parameter | mass relation | mass [103kg] | mass [103kg] | mass [103kg] | mass [103kg] |
| Wing-body structure | unit mass |  4,75 kg/m2 | 19,95 | 19,95 | 19,95 | 19,95 |
| Pressure cabin structure | unit mass |  3,5 kg/m2 | 6,98 | 6,98 | 6,98 | 6,98 |
| Tank structure | unit mass |  3,5 kg/m2 | 2,64 | 2,64 | 2,64 | 2,64 |
| Fin structure | unit mass |  4,3 kg/m2 | 1,29 | 1,29 | 1,29 | 1,29 |
| Propulsion system |   |  estimated | 1,50 | 1,50 | 1,50 | 1,50 |
| Heat exchangers |  |  estimated | 1,00 | 1,00 | 1,00 | 1,00 |
| He |   |  estimated | 1,00 | 1,00 | 1,00 | 1,00 |
| Landing gear |   | 4% of MLm | 2,00 | 2,00 | 2,00 | 2,00 |
| Pumps and other equipment |  |  estimated | 1,00 | 1,00 | 1,00 | 1,00 |
| Crew & cabin equipment |   |  estimated | 3,00 | 3,00 | 3,00 | 3,00 |
| Operating Empty mass  |   |   | 38,36 | 38,36 | 38,36 | 38,36 |
| 300 Pax Pay Load, (PL) |   |   | 31,50 | 31,50 | 31,50 | 31,50 |
| Zero Fuel mass, (ZFm) |   |   | 69,86 | 69,86 | 69,86 | 69,86 |
| Fuel mass, (Fm = mLH2) | density |  71 kg/m3 | 55,00 | 42,90 | 32,45 | 15,95 |
| Take-Off mass, (TOm) |   |   | 124,86 | 112,76 | 102,31 | 85,81 |
| Max. Landing mass, (MLm) | estimated |   | 60,00 | 60,00 | 60,00 | 60,00 |
| Wing loading at TOm |   |   | 44,28 kg/m2 |  40,27 kg/m2  | 36,54 kg/m2 | 30,65 kg/m2 |
| Wing loading at MLm |   |   | 21,43 kg/m2 | 21,43kg/m2 | 21,43 kg/m2 | 21,43 kg/m2 |

5. Aerodynamics.

For this preliminary project proposal it is estimated, that the aerodynamic characteristics of the Vulcan IIL are identical to those of the Vulcan II, ref.[1], concerning both the performance calculation and the boundary layer temperature.

Let us review and summarize the definitions of the relevant coefficients from ref.[1[:

Lift coefficient  , Drag coefficient ,

with L = Lift [kN], D = Drag [kN], q = dynamic pressure [bar],

 = Angle of Attack [º], (AoA), and

Aref = Aerodynamic reference area [m2].

We can split up cD in a lift independent term and in a lift induced term:

  eq.[1], with:

  eq.[2],

With cF = Friction coefficient from ref.[9], (fig.4b with Re ≈ 108),

=  = Aspect ratio with b = wing-body span [m],

WSref = Wetted Surface area [m2],

cDintb = interfer. & base Drag coeffient = 0,1\* cF\*WSref/Aref estimated.

Let us consider again only small values for , so we assume here

sin  =  and cos  = 1.

*5.1. Subsonic Flow.*

  [1/º] eq.[3]

  eq.[4]

*5.2. Supersonic Flow.*

Leading Edge parameter LEp:  eq.[5]

 eq.[6] with  [1/º] eq.[7].

  eq.[8]

  eq.[9].

Leading Edge parameter of the front part:  eq.[10]  [1/º] eq.[11]

*5.3. Maximum Lift-to-Drag Ratio.*

For the maximum L/D = cL/cD ratio:  eq.[12]

M < 1: , M > 1:  eq.[13]

*5.4. Compilation of Aerodynamic Data for Performance Calculation.*

The compilation the aerodynamic data required for the performance calculation and identical to those of ref.[1] is presented in Table 3:

Table 3: Aerodynamic data required for performance calculation.

|  |  |  |  |  |  |  |  |
| --- | --- | --- | --- | --- | --- | --- | --- |
|  M |  cLº] |  cF |  cD0 |  cD\* |  \* [º] |  cL\* |  L/Dmax |
| 0,25 | 0,0568 | 0,00295 | 0,0083 | 0,0167 | 2,90 | 0,1647 | 9,871 |
| 0,5 | 0,0635 | 0,00290 | 0,0082 | 0,0164 | 2,72 | 0,1727 | 10,526 |
| 0,8 | 0,0916 | 0,00282 | 0,0080 | 0,0160 | 2,23 | 0,2046 | 12,825 |
| 2 | 0,0363 | 0,00232 | 0,0066 | 0,0131 | 3,05 | 0,1108 | 8,442 |
| 2,5 | 0,0281 | 0,00220 | 0,0062 | 0,0124 | 3,42 | 0,0962 | 7,726 |
| 3 | 0,0233 | 0,00190 | 0,0054 | 0,0107 | 3,53 | 0,0824 | 7,665 |
| 3,5 | 0,0203 | 0,00174 | 0,0049 | 0,0098 | 3,68 | 0,0747 | 7,589 |
| 4 | 0,0178 | 0,00160 | 0,0045 | 0,0091 | 3,79 | 0,0675 | 7,459 |
| 4,5 | 0,0159 | 0,00135 | 0,0038 | 0,0076 | 3,71 | 0,0590 | 7,726 |
| 5 | 0,0143 | 0,00131 | 0,0037 | 0,0074 | 3,86 | 0,0550 | 7,422 |
| 5,5 | 0,0129 | 0,00120 | 0,0034 | 0,0068 | 3,88 | 0,0501 | 7,381 |
| 6 | 0,0118 | 0,00115 | 0,0033 | 0,0065 | 3,97 | 0,0469 | 7,209 |
| 6,5 | 0,0109 | 0,00106 | 0,0030 | 0,0060 | 3,98 | 0,0432 | 7,206 |
| 7 | 0,0101 | 0,00099 | 0,0028 | 0,0056 | 3,98 | 0,0401 | 7,198 |
| 7,5 | 0,0094 | 0,00095 | 0,0027 | 0,0054 | 4,05 | 0,0380 | 7,076 |
| 8 | 0,0088 | 0,00089 | 0,0025 | 0,0050 | 4,05 | 0,0356 | 7,074 |
| 8,5 | 0,0083 | 0,00084 | 0,0024 | 0,0048 | 4,06 | 0,0336 | 7,061 |

*5.5. Boundary Layer Temperature.*

The VULCAN IIL is also subjected to aerodynamic heating, but in a different extent as the VULCAN II, since she is subjected to lower skin temperatures, but during the largest part of her flight for a much longer exposure time. So also for the VULCAN IIL the temperature of the boundary layer is a crucial parameter for structural design,

We obtain for the boundary layer wall temperature Tw, according to ref.[4]:

  eq.[14],

With T0 = free stream air temperature and Pr = Prandtl number, which is defined as the heat conductivity ratio of the air at the boundary layer wall. As in ref.[1] we estimate in accordance with ref.[5]: Pr = 0,7. We obtain then with eq.[14]:

 eq.[15].

For M = 8 and  = 1,3, (see under *6.1.*), we find: Tw = 2005 ºK.

6. Propulsion System Characteristics and Performance.

The characteristics of the propulsion system of the Vulcan IIL are identical for the air breathing part with pre-cooling of those of the Vulcan II., since it is a simplified version with no rocket propulsion and a non-retractable intake.

*6.1. Pre-Cooling Heat Exchanger Characteristics.*

As with the VULCAN II the characteristics of the pre-cooling heat exchanger are determined by the requirement to realize a more efficient compression in the intake and to reduce its static temperature to an acceptable level for efficient combustion.

As with the VULCAN II we accept the theoretical stagnation temperature Tts as a design parameter:

  eq.[16]

Considering again real gas conditions, according to ref.[5] determined by increasing molecule vibration excitation and transition to dissociation above a temperature T = 2000 ºK, we are taking into account accordingly a dependence = f(T).

As a result of considering real gas effects and based on quantum mechanics theory, as explained in ref.[1], we obtain fig. 4 for = f(T):

Fig.4: Specific heat ratio  for air depending from temperature T.



p = 1 bar

As in ref.[1] we take partly into account on a preliminary and only quality basis the pressure increase, resulting in a lower average value of. A verification of this crude assumption can only be made by experiment. We assume again on a preliminary basis:

  eq.[17] and  eq.[18].

*6.2. Estimated Engine Performance.*

Accordingly the main engine performance parameters used are its propellant specific impulse Isp and its total efficiency  They are independent of the dimensions of the engine.

Definition of specific impulse Isp, according to ref.[1]:

 eq.[19]

With considering the 2 engines together:

 total thrust [kN] and  = total propellant mass flow.

 = gravity acceleration 

We obtain with the law of conservation of momentum:

  eq.[20],

with:  = total air mass flow ,

  = theoretical exhaust velocity ,

  = flight speed

 = Mach number ,  = speed of sound 

As with ref.[1] the theoretical exit velocity vthex would be attained in a nozzle with a complete expansion to ambient pressure. In practice the flow expands in the nozzle incompletely to a higher than the outside pressure. which would result theoretically in obtaining the same momentum. However the pressure difference will lead to a diverging of the flow at the nozzle exit, causing a momentum loss. Film cooling of the nozzle by air or hydrogen results in an additional exit velocity and momentum loss. These losses are included in vthex and documented in ref.[6].

We can transform eq.[20] to eq.[21] in order to obtain:

  eq.[21]

With the law of conservation of energy and taking into account the engine internal efficiency int we obtain:

  eq.[22]

and by transforming:  eq.[23]

with for hydrogen the specific combustion heat hc = 121 106 [J/kg].

Like in ref.[1] stochiometric combustion:  and a constant value for the internal efficiency int = 0,7 is assumed.

For the total engine efficiency  we can write again, by taking into account the law of energy conservation:

 eq.[24] and with eq.[19]:  eq.[25]

*6.3. Engine Performance Table and Diagrams.*

We assume the same performance characteristics for the pre-cooled turbo-ramjet as displayed in chapter 6 with figs.[6] and [7] of ref.[1] and presented in table 4:

Table 4: Estimated performance of the pre-cooled ramjet propulsion system:

|  |  |  |  |
| --- | --- | --- | --- |
|  M |  Vthex [m/s] |  Isp [s] |  |
| 0 |  | 7877 |  |
| 0,8 |  | 7084 | 0,136 |
| 2 |  | 6009 | 0,287 |
| 2,5 |  | 5600 | 0,335 |
| 3 |  | 5212 | 0,374 |
| 3,5 |  | 4846 | 0,406 |
| 4 |  | 4499 | 0,430 |
| 4,5 |  | 4170 | 0,449 |
| 5 |  | 3859 | 0,461 |
| 5,5 |  | 3564 | 0,469 |
| 6 |  | 3284 | 0,471 |
| 6,5 |  | 3018 | 0,469 |
| 7 |  | 2765 | 0,463 |
| 7,5 |  | 2524 | 0,453 |
| 8 |  | 2294 | 0,439 |
| 8,5 |  | 2074 | 0,422 |

7. Flight Performance.

For this first and preliminary assessment of the flight performance for the Vulcan IIL we follow for the take-off and accelerated climb phase the method and assumptions used for the Vulcan II, ref.[1]. But we must take account into our assessment the important differences, since the Vulcan IIL has no rocket propulsion, a non-retractable variable-geometry Prandtl-Meyer intake with a slightly larger capture area, subsequently main frame diameter. The thrust Ft has to be limited at take-off for noise reduction at civil airports.

For the cruise flight until destination, including the descent flight phase and landing, we assume flight at constant Mach number and angle of incidence, but with a slightly increasing attitude in agreement with the conditions for the validity of the Bréguet range equation. Included in our case in this assumption are the effect of the centrifugal force and in accordance with refs.[7] and [8] a 5% trip fuel reserve to the destination under cruise conditions, but with no allowance for descend distance, by not considering a lower fuel flow for a steep spiral descend and landing in order to keep flight time at a minimum. With this definition we will keep us at the safe side with an inherent additional spare fuel quantity.

*7.1. Take-off and Accelerated Climb.*

From fig.[8] of ref.[1] we copy fig.[4].: Arrangement of forces and climb angle:

L

D

Ft

W = m  g

Fc

c

c

+xa

(flight direction)

+za

For the performance calculations for the Mach numbers of table 4 we need the value of the thrust force Ft instead of the specific impulse Isp. We use therefore eq.[19] and write:

  eq.[26],

and with continuity law for the 2 engines together, with p in [bar]:

  eq.[27],

with Ac = engine capture area [m2], v = M  a [m/s] and with

 eq.[28].

Within the waverider configuration presented in fig.1 and 2 we can determine such an engine-airframe arrangement, that:

  eq.[29],

with demf [m] = engine main frame diameter.

Ac will be determined according to eq.[27] by the mass flow requirement  at the operational ceiling altitude HaltC with M = 8, at which the climb angle c = 0 and the acceleration ac = 0, (see also fig. 8) to obtain the total thrust at ceiling FtC with eq.[30] according to eq.[28], with q in [bar]:

 eq.[30]

If we assume, like in ref.[1]. for reasons of optimal structural utilization by more or less constant aerodynamic loads, as shown in table 3, a climb with constant dynamic pressure q, we obtain again:

  eq.[31]

with palt, being the static pressure at an altitude Halt, determined according to the ISA standard atmosphere.

With eq.[28], [29], [33] or [34] we can calculate the thrust, depending from Mach number, alternatively speed, and altitude.

From fig.8 we find in flight direction the following eq.[34] with the general movement equation and Newton’s law, with ac = acceleration in [m/s2]:

  eq.[34],

with  eq.[35]

And for the required Lift Lreq:

  eq.[36]

To find a suitable expression for the centrifugal force Fc we consider that at circular satellite speed vs = 8000 [m/s] we would obtain the corresponding centrifugal force Fcs = m  g. We can then formulate the following relation:

  eq.[37]

With eq.[37] we can transform rel.[34], respectively rel.[36] to:

  eq.[38]

  eq.[39]

Since we assume maximum L/D and q = constant for climb conditions we can write with eq.[40] the condition for the effective aerodynamic Lift Leff:

  eq.[40]

As in ref.[1] climb conditions with matching of L with Lreq will be left open here for assessment by follow-on investigations and research.

For the start of Cruise flight conditions, ( c = 0), we obtain with eqs.[39] and [40] for the Lift force at the start of Cruise flight conditions LC:

 eq.[41],

with mC being the start of Cruise flight mass, being thus a design criterion directly determined by the lift force LC.

Since air breathing propulsion thrust is depending from the air and fuel mass flow we can, like in ref.[1], only make a stepwise performance calculation with, additional to q = constant, the assumption of a constant acceleration ac for each step. In this document this will be done with a limited number of steps and therefore in a more or less crude way. For the next iterations in follow-up project studies an elaborated computer program can be set up for this purpose.

With the stepwise constant acceleration ac [m/s2] we can calculate stepwise the time lapse t:

 eq.[42], with v [m/s] being the speed difference,

and with it the mass dissipation m, determined by fuel consumption:

 eq.[43].

The traveled distance with each step s we can calculate with the general equation of movement:

 eq.[44]

And for the corresponding stepwise range Rs we make the approximation:

 eq.[45].

*7.2. Cruise, Descent and Landing.*

As it has already been indicated we use for the calculation of the cruise flight phase the Bréguet equation and assume here that the required fuel for the descent and landing phase is included.

A special application of the Bréguet range equation by incorporating the influence of the centrifugal Force and a 5% spare of the trip fuel, in accordance with JAR-OPS 1, (refs.[7] and [8]), needs some in-depth explication. This we present here briefly by undertaking a re-assessment of its fundamental deduction including the effect of the centrifugal force. Concerning the requirement of 30 min. holding at 1500 ft as final reserve fuel, ref.[7] and [8], we assume, that this is covered by a possible additional range in case of an aerodynamically optimized descent flight, instead of a time optimized one, as assumed here.

We start with the general differential equation, assuming a constant speed v:

  with dR = range element and dt = time element.

We can also write, with dm = mass element: 

Now we extend and obtain, with c≈ 0, Ft = D, L/D = cL/cD

L = Leff =Lreq, ,  and

 

As for the classic Bréguet range equation we must now integrate this differential range equation between the boundaries between the mass mC at the start of cruise decent and landing flight phase and the zero fuel mass + 5% trip fuel mass, (ZFm +5% mLH2), in accordance with flight safety requirements.

We obtain then for the cruise Range RC , including descent and landing:



 eq.[46],

(Integration constant = 0).

Resuming we can calculate with eq.[46] the Cruise Range at constant speed to the airport of destination, including descent, landing and 5% spare, depending on the fuel consumption with taking into account the corresponding mass dissipation with it.

In the tables 5.1, 5.2 and 5.3 the flight performance calculation is presented. For format reasons it is, like in ref.[1], split up in 3 parts with taking into account the differences from the spaceplane Vulcan II, ref.[1], a slightly increased capture area Ac = 13,939 m2 and at take-off a combustion ratio  = 0,2, ( = 1 for stochiometric combustion), for environment noise reduction.

The tables 5.1, 5.2 and 5.3 represent only the calculation for the design range Rd = 20000 km. For the ranges, 15000 km and 10000 km the calculation is done on the same way, reminding the assumption of constant dynamic pressure q in climb conditions, but within a range of 5000 km Mach 8 and the corresponding cruise altitude Halt cannot be attained within that range. For that range the cruise flight is assumed at Mach 7.

Table 3.1.: Speed, altitude and propulsion air mass flow, R = Rd = 20000 km

|  |  |  |  |  |  |  |  |
| --- | --- | --- | --- | --- | --- | --- | --- |
|  M |  p [bar] |  a [m/s] |  v [m/s] |  q [bar] |  Halt [km] |

|  |
| --- |
|  [kg/s] |

 |
| 0 | 1,0315 | 340,293 | 0 | 0 | 0 | 1187,28 |
| 0,25 | 1,0315 | 340,293 | 85,073 | 0,0453 | 0 | 2968,21 |
| 0,5 | 1,0315 | 340,293 | 170,146 | 0,1811 | 0 | 5936,41 |
| 0,8 | 0,0473 | 295,702 | 236,562 | 0,0212 | 23,332 | 500,78 |
| 2 | 0,0309 | 297,549 | 595,098 | 0,0867 | 23,657 | 812,87 |
| 2,5 | 0,0198 | 299,494 | 748,735 | 0,0867 | 26,546 | 646,07 |
| 3 | 0,0137 | 301,113 | 903,339 | 0,0867 | 28,965 | 535,50 |
| 3,5 | 0,0101 | 302,46 | 1058,61 | 0,0867 | 30,988 | 456,95 |
| 4 | 0,0077 | 304,624 | 1218,496 | 0,0867 | 32,806 | 396,99 |
| 4,5 | 0,0061 | 307,544 | 1383,948 | 0,0867 | 34,395 | 349,53 |
| 5 | 0,0049 | 310,579 | 1552,895 | 0,0867 | 36,062 | 311,51 |
| 5,5 | 0,0041 | 312,592 | 1719,256 | 0,0867 | 37,177 | 281,36 |
| 6 | 0,0034 | 315 | 1890 | 0,0867 | 38,52 | 255,94 |
| 6,5 | 0,0029 | 317,06 | 2060,89 | 0,0867 | 39,677 | 234,72 |
|  7 | 0,0025 | 322,601 | 2258,207 |  0,0867 | 40,769 | 214,21 |
| 7,5 | 0,0022 | 320,67 | 2405,025 |  0,0867 | 41,723 | 201,13 |
| 8 | 0,0018 | 323,217 | 2585,736 | 0,0829 | 42,238 | 178,80 |

According to table 3.1. the flight profile for R = Rd = 20000 km is displayed in fig. 9:



**R = Rd = 20000km**

Table 3.2: Forces, mass, climb angle and acceleration, R = Rd = 20000 km.

|  |  |  |  |  |  |  |  |
| --- | --- | --- | --- | --- | --- | --- | --- |
|  M |  Ft [kN] |  D [kN] |  Leff [kN] |  Lreq [kN] | m [103 ٭ kg] |  c [ º] |  ac[m/s2] |
| 0 | 535,7745 | 0 | 1224,832 | 1224,832 | 124,86 | 0 | 4,291 |
| 0,25 | 1289,619 | 211,628 | 2088,880 | 1223,346 | 124,72 | 0 | 8,643 |
| 0,5 | 2482,343 | 832,164 | 8759,693 | 863,755 | 124,55 | 45 | 6,314 |
| 0,8 | 1016,178 | 94,909 | 1217,168 | 1217,168 | 124,18 | 0 | 7,419 |
| 2 | 1399,119 | 318,831 | 2691,503 | 1200,064 | 123,48 | 5 | 7,899 |
| 2,5 | 1036,319 | 302,340 | 2335,751 | 1191,728 | 123,01 | 5 | 5,119 |
| 3 | 799,532 | 261,111 | 2001,358 | 1181,470 | 122,45 | 5 | 3,553 |
| 3,5 | 634,264 | 239,123 | 1814,595 | 1171,452 | 121,76 | 3,5 | 2,657 |
| 4 | 511,573 | 219,883 | 1640,072 | 1158,022 | 120,96 | 2,5 | 1,993 |
| 4,5 | 417,520 | 185,527 | 1433,336 | 1141,281 | 120,0 | 2 | 1,601 |
| 5 | 344,330 | 180,029 | 1336,200 | 1122,011 | 118,92 | 2 | 1,052 |
| 5,5 | 287,240 | 164,912 | 1217,161 | 1098,772 | 117,48 | 1,8 | 0,747 |
| 6 | 240,772 | 158,041 | 1139,254 | 1070,454 | 115,60 | 1,5 | 0,473 |
| 6,5 | 202,930 | 145,673 | 1049,749 | 1033,864 | 112,90 | 1,3 | 0,299 |
| 7 | 169,678 | 135,366 | 1049,749 | 978,474 | 108,39 | 0,8 | 0,190 |
| 7,5 | 145,436 | 130,556 | 974,309 | 924,161 | 103,57 | 0,2 | 0,112 |
| 8 | 117,509 | 116,899 | 826,987 | 826,987 | 94,13 | 0 | 0,006 |

Table 3.3.: Time lapse, fuel consumption, flown distance, range,

 R= Rd = 20000 km.

|  |  |  |  |  |
| --- | --- | --- | --- | --- |
|  M |  t [s] | m [kg] |  s [km] | Rs [km] |
| 0 | 19,82 | 137,463 | 0,843 | 0 |
| 0,25 | 9,84 | 170,614 | 1,256 | 0 |
| 0,5 | 10,52 | 364,654 | 2,139 | 1,512 |
| 0,8 | 48,33 | 706,708 | 20,097 | 20,097 |
| 2 | 19,45 | 461,684 | 13,069 | 13,020 |
| 2,5 | 30,20 | 569,764 | 24,948 | 24,853 |
| 3 | 43,70 | 683,323 | 42,869 | 42,706 |
| 3,5 | 60,18 | 802,992 | 68,519 | 68,391 |
| 4 | 83,0 | 962,111 | 107,996 | 107,893 |
| 4,5 | 105,51 | 1076,896 | 154,936 | 154,842 |
| 5 | 158,12 | 1438,230 | 258,691 | 258,534 |
| 5,5 | 228,47 | 1877,087 | 412,307 | 412,104 |
| 6 | 361,16 | 2699,187 | 713,457 | 713,213 |
| 6,5 | 659,19 | 4518,019 | 1423,555 | 1423,189 |
| 7 | 770,62 | 4820,246 | 1796,794 | 1796,619 |
| 7,5 | 1605,95 | 9431,985 | 4007,448 | 4007,424Breguet |
| 8 | 4295,59 | 22427,490 | 42772,213 | 11107,275 |
| Total flight | 8509,66 |   |   | 20151,672 |

 2,36 hrs

For the ranges of 15000, 10000 and 5000 km the performance calculation is done exactly on the same way and not displayed here in order to keep a clear overview of the document. With the assumption of climb with q = const., we obtain only relatively slight differences in respect to the accuracy of the flight profile for the other ranges. However within the range of 5000 km we cannot attain M = 8 under these climb conditions. Therefore for that range M = 7 then assumed here for cruise flight.

8. Required Fuel Mass and Payload Transport Efficiency.

To evaluate a possible economic interest of hypersonic suborbital transport flight an assessment of the fuel consumption in relation to the estimated masses of the VULCAN IIL is made and presented here for the ranges 20000, 15000, 10000 and 5000 km and compared with that of a future advanced subsonic airliner with the same payload for the same ranges.

Table 2 presented already the mass estimation and the required fuel mass, (mLH2), depending from the ranges of 20000, 15000. 10000 and 5000 km.

Let us review and assume from ref.[7] for a future advanced subsonic long-range airliner with a very advanced propulsion system, (geared fan or ducted propeller), the following data, presented in tables 4 and 5:

|  |
| --- |
| Table 4: Basic data of future Advanced Long-Range Subsonic airliner, ref.[7]. |
| Lift-to-drag ratio |  L/D | 25 |   |
| Operating Empty Mass / Payload | OEm/PL | 3,25 |   |
| Lower specific heating value (kerosene) |  hl | 43,2 |  MJ/kg |
| Proulsion efficiency |   | 0,45 |   |
| Gravity constant |  g | 9,81 |  m/s2 |

In fig. 10 we then compare the fuel mass ratio ZFm/Fm depending from the range of the VULCAN IIL with that of a very advanced future airliner, calculated in ref.[7], which could be developed in the same time. However we must consider that the range calculations of the VULCAN IIL and the subsonic airliner of ref.[7] are based on different assumptions, but let us accept this for this first preliminary investigation.

We see then from fig.10 a substantial larger fuel mass ratio for the VULCAN IIL, but especially for the longer ranges the difference might be compatible with that between first class and economy class fares for a subsonic airliner.

Further we must consider that with the VULCAN IIL liquid hydrogen, (LH2), is required and no hydrocarbon fuel. The cost of which is depending directly from that of electricity generation and not from world market crude oil prices.

Fig. 10: Fuel mass ratio Fm/ZFm depending from range R.



In Chapter 6 of the SBAC Aviation and Environment Briefing Papers, ref.[9], the parameter Payload Fuel Efficiency = Payload  Range / Fuel Mass is introduced, which is not an efficiency according to its usual definition, but merely a quality number for the Range factored with the Payload Fuel Mass Ratio, which is commonly used in aircraft performance evaluations. However this parameter doesn’t take into account the energy produced by burning the fuel in the propulsion system.

Let us define here, like in ref.[9] a dimensionless Payload Transport Efficiency PTE as a quality number by extending the above defined ratio with the gravity acceleration g and the Lower Heating Value hl as written in eq.[47], assuming that in accordance with JAR – OPS 1, (ref.[8]), only 95% of the stored Fuel Mass Fm is used:

  eq.[47].

The PTE may well exceed unity, which doesn’t correspond to the definition of a thermodynamic efficiency. In figure 11 the calculation of the PTE, depending from the range R is presented for the VULCAN IIL and the Advanced Subsonic Airliner.

Figure 11: Payload Transport Efficiency PTE depending from Range R

**Payload Transport Efficiency PTE**

0,0

0,5

1,0

1,5

2,0

2,5

3,0

3,5

5000

7000

9000

11000

13000

15000

17000

19000

**Range [km]**

**PTE**

VULCAN IIL

Adv.Subsonic

We see, apart from the much lower values of the here defined PTE for the VULCAN IIL, a remarkable difference with the Advanced Subsonic Airliner, since for the VULCAN IIL the PTE is more or less independent of the range R. This is characteristic for a suborbital transport vehicle, for which most of the energy is spent for the acceleration flight phase.

9. Conclusions.

With the realization of a SSTO Spaceplane also its suborbital transportation variant comes into reach within the next 10 – 15 years, provided the same main technology research and development items, in addition to those, which are generally required with civil transport aircraft development, will be executed:

 - Further experimental development of heat exchangers, based on nanotube

 SiC technology with helium in close circuit as coolant, pioneered and

 already successfully realized by REL. The aim is to obtain a heat

 exchanger, which can sustain stagnation air temperatures of about 3000ºK,

 but with a high heat transfer capability of the nanotechnology tubes.

- Research and development of actively cooled lightweight SiC structures,

 also with helium coolant in close circuit. Structure tests and heat flow tests

 under simulated flight at M = 8 conditions, but less severe, as reentry is not

 required for the VULCAN IIL.

- Research and development of the turbo-ramjet propulsion with heat

 exchanger system. Tests of its components, subsystems, like the helium

 closed circuit, and the complete system. For this it makes sense to perform

 separate wind tunnel test with the variable and retractable near-isentropic

 PRANDTL-MEYER intake, including the subsonic diffuser. For testing the

 part of the system behind the diffuser pre-heated air to simulate the effect

 of the intake can be used.

- Design, development and test of remotely controlled free flight models on

 reduced scale to identify and solve handling and control problems, leading

 to an early verification of flight characteristics in all flight phases.

Also the design and development of a suborbital transportation system, like the VULCAN IIL, is a major challenge, but in a lesser extend than an SSTO system. The challenge however is finding a market large enough to justify to develop and produce such a system, even it is derived from an SSTO space transportation system. On the other hand every new technical system generates it own market and not the other way round. Real progress is only achieved with an intrepid mind, So let’s take that challenge.

 Kay Runne, September 2017.

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List of Symbols.

 = Engine air intake capture area in [m2]

Aref = Aerodynamic reference Area in [m2]

a = speed of sound in [m/s]

ac = acceleration in [m/s2]

b = wing-body span in [m]

 = Drag coefficient

cDintb = interference & base Drag coeffient

cD\* = Drag coefficient at maximum L/D

cF = Friction coefficient

 = Lift coefficient

 = Lift slope in [1/º]

cLf = front part Lift slope in [1/º]

cL\* = Lift coefficient at maximum L/D

D = aerodynamic Drag in [kN]

demf = engine main frame diameter [m]

dcab = diameter of pressure cabin [m]

Fc = centrifugal Force in [kN]

Fcs = centrifugal Force at circular satellite speed in [kN]

Ft = total Thrust in [kN]

g = gravity acceleration = 9,81 [m/s2]

Halt = altitude in [km]

HaltC = start of Cruise flight phase altitude in [km]

 = specific propellant Impulse in [s]

L = aerodynamic Lift in [kN]

LEp  = leading edge sweep ratio parameter

LEpf = front part leading edge sweep ratio parameter

Lreq = required Lift in [kN]

lcab = length of pressure cabin [m]

M = = Mach number

MLm = Maximum Landing mass in [103kg]

m = vehicle mass in [103kg]

mC = vehicle mass at start of Cruise flight phase [103kg]

mLH2 = fuel mass in [103kg]

 = propellant mass flow in [kg/s]

 = air mass flow in [kg/s]

 = air mass flow at HaltC in [kg/s]

nx = acceleration load factor in flight direction

PL = Pay Load in [103kg]

Pr = Prandtl number

PTE = Payload Transport Efficiency

p = static pressure in [bar]

palt = static pressure at Halt, (ISA), in [bar]

R = Range in [km]

RC = Cruise Range in [km]

Rd = design Range in [km]

T = Temperature in [ºK]

TOm = Take-Off mass in [103kg]

Tts = theoretical stagnation Temperature in [ºK]

T0 = free stream air Temperature in [ºK]

Tw = wall boundary layer air Temperature in [ºK]

v = flight speed in [m/s], in figs.9 and 10 in [km/s]

vs = circular satellite speed in [m/s]

vthex = theoretical exhaust velocity in [m/s]

W = m  g

 = Weight in [kN]

WSref = Wetted Surface area in [m2]

ZFm = Zero Fuel mass in [103kg]

= Angle of Attack, (AoA), in [º]

 = Angle of Attack, (AoA), in [º] for maximum L/D

hc = specific combustion heat in [J/kg]

m = mass dissipation per step in [kg]

Rs = stepwise range in [km]

s = traveled distance per step in [km]

t = time lapse per step in [s]

v = speed difference per step in [m/s]

 = total propulsion efficiency

int = internal thermal efficiency

 = specific heat ratio



= average specific heat ratio

c = climb angle in [º]

s = leading edge sweep angle in [º]

sf = front part leading edge sweep angle in [º]

 = = Aspect Ratio

f = front part Aspect Ratio

 = half Mach cone angle in [º]

 = waverider inverted V-angle in [º]

f = waverider front part inverted V-angle in [º]

 = density in [kg/m3]

alt = density at Halt, (ISA), in [kg/m3]